Budget, Finance, and Economic Development Committee (2025-2027) on 2026-04-06 9:00 AM

Meeting Time: 04-06-26 09:00

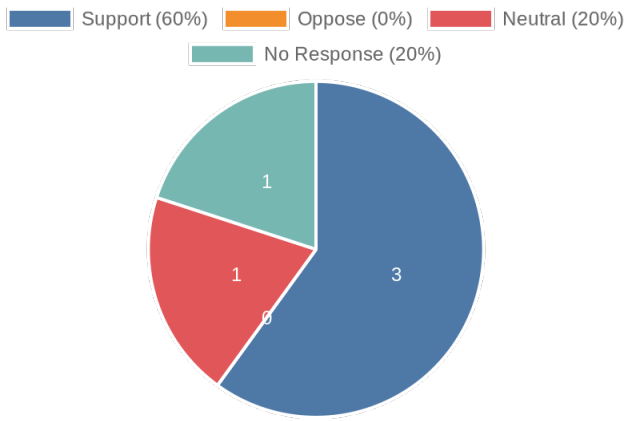
eComments Report

Meetings	Meeting Time	Agenda Items	Comments	Support	Oppose	Neutral
Budget, Finance, and Economic Development Committee (2025-2027) on 2026-04-06 9:00 AM	04-06-26 09:00	2	5	3	0	1

Sentiments for All Meetings

The following graphs display sentiments for comments that have location data. Only locations of users who have commented will be shown.

Overall Sentiment



Budget, Finance, and Economic Development Committee (2025-2027) on 2026-04-06 9:00 AM

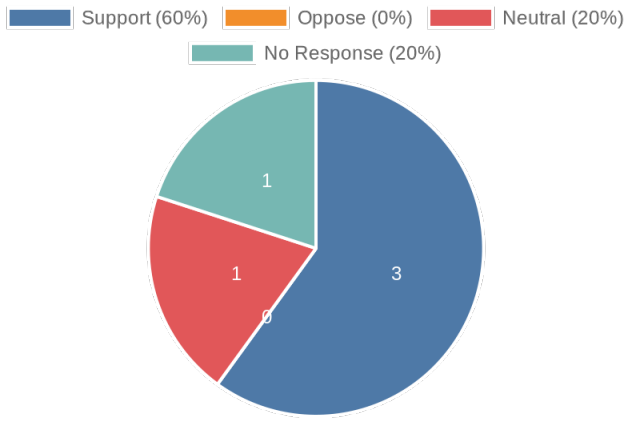
04-06-26 09:00

Agenda Name	Comments	Support	Oppose	Neutral
BFED-1 PROPOSED FISCAL YEAR 2027 BUDGET FOR THE COUNTY OF MAUI (BFED-1)	5	3	0	1

Sentiments for All Agenda Items

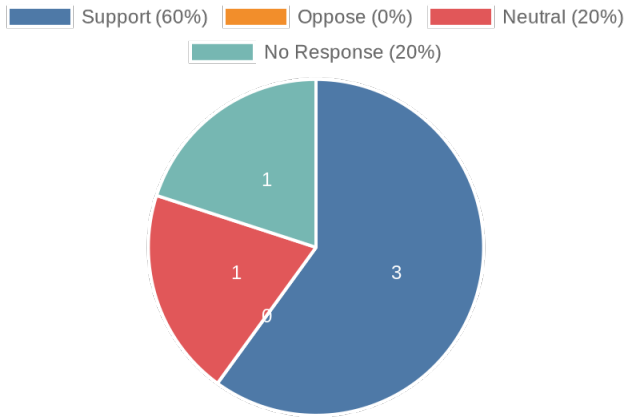
The following graphs display sentiments for comments that have location data. Only locations of users who have commented will be shown.

Overall Sentiment



Agenda Item: eComments for BFED-1 PROPOSED FISCAL YEAR 2027 BUDGET FOR THE COUNTY OF MAUI (BFED-1)

Overall Sentiment



BFED Committee
Location:
Submitted At: 12:35pm 04-06-26

Testimonies received from BFED Committee

Guest User
Location:
Submitted At: 11:00am 04-06-26

Chair and Councilmembers, thank you for the opportunity to testify.

I’m here in support of a meaningful COLA and compensation adjustment for Maui firefighters.

The County and State have already acknowledged the cost-of-living crisis. Maui County officials received a 5% increase in 2024, followed by much larger increases in 2025—about 26% for councilmembers and over 23% for the council chair. At the state level, the governor received a 15% increase, with more planned. Police officers are receiving 5% raises annually for four years, plus retention bonuses.

Meanwhile, inflation continues to rise, with housing, food, and daily costs increasing across Hawai_i. On Maui, the median home price is now over $1.4 million.

Firefighters are expected to respond to everything—fires, medicals, rescues, disasters—24/7, in one of the most expensive places in the country to live.

If COLA and raises are justified for elected officials and other public safety employees, they are equally justified for firefighters.

A COLA is not extra—it’s necessary to retain experienced firefighters and protect the safety of our community.

Thank you.

Travis Liggett
Location:
Submitted At: 9:07am 04-06-26

Aloha Chair and Council Members,

As a childhood NASA enthusiast who later became a NASA engineer, I navigated the trauma of multiple shuttle disasters.

The attach reports found that twice that the same culture change was needed. Technical concerns were not flowing freely, decisions were being made behind closed doors based on profits, to avoid liability, delays or perceived shame, and civil servants paid for the dysfunction with their lives, live on TV.

County employees should be required to study and understand all lessons of these reports, which is basically this: dissidence should always be received and rewarded by decision makers, even if it turns out to be inaccurate, and that the best way for any captain to stop losing crew members is to accept and review all potentially life-saving findings.

Aloha,
Travis Liggett, M.S.

Guest User
Location:
Submitted At: 10:51pm 04-05-26

TESTIMONY IN SUPPORT OF FUNDING FOR LA_AKEA VILLAGE HOUSING PROJECT
Before the Maui County Council Budget, Finance, and Economic Development Committee

Aloha Chair Sugimura, Vice Chair Batangan, and Members of the Committee,

My name is Brian Mercer, resident of Paia, and I live in Skill Village neighboring the proposed housing project. I am writing in strong support of the County of Maui Affordable Housing Fund request of $4.5 million for the La_akea Village supported housing project in P__ia.

My husband and I are personal trainers and have volunteered in the past at La'akea Village teaching fitness classes to their participants. We've seen firsthand how empowering their program for adults with IDD is. We've also heard firsthand from their participants how isolating life can sometimes be at home, away from their La'akea community. Not only would supporting the project provide much needed affordable and safe housing, it would continue to provide that sense of community after the day program has ended.

Particularly as adults with IDD age out of family homes or lose caregivers, finding housing that is both accessible and affordable can be a struggle. This project adds much needed ADA-accessible housing inventory to Maui with dedicated units for adults with disabilities and integrated housing for support staff and other income-qualified residents.

We have lived across the street from La'akea for over 11 years now and have seen through our time volunteering as it has grown and added a commercial kitchen, a store, and developed a farm that grows produce and flowers and now has horses. We've seen how well everyone does there and watched the individuals who take part in their program grow. We've seen their lives enriched. We'd love to see La'akea Village continue to grow and thrive with a housing community.

I am in favor of the proposed development and urge the committee to fully fund the requested $4.5 million so that this project can move forward to development.

Mahalo for your time and consideration.

Sincerely,
Brian Mercer

Guest User
Location:
Submitted At: 8:56am 04-03-26

April 3, 2026

Honorable Yuki Lei Sugimura, Chair
Budget, Finance, and Economic Development Committee
Maui County Council

Submitted by Malia Bohlin on Behalf of Nicole Hokoana, Aloha House CEO

RE: Testimony in support of Affordable Housing Fund grant application by Aloha House for Sober Living Program Home Acquisition

Funding Request:
Aloha House is requesting grant in the amount of $2,000,000 from the 2027 Affordable Housing Fund to be used with a cash commitment of up to $1,000,000 from Aloha House from its financial reserves to support the total anticipated cost $3,000,000 for the fee-simple purchase of two properties

These properties will provide supportive long-term housing for up to 20 very low-income adults who are participating in an Aloha House behavioral health program, having completed intensive substance use treatment, and are receiving ongoing care to address co-occurring mental health/addiction disorders and require additional support and case management to facilitate long-term recovery.

Program History and Services:
Aloha House, Maui's most established, comprehensive substance use treatment provider, operates an extended-care Sober Living Program to provide transitional and long-term housing for single adults that are participating in or have completed intensive substance use treatment and require ongoing case management and support to maintain sobriety.

Aloha House initiated the first Sober Living Program in 2013 and has steadily expanded the program to include six residential homes in Central Maui which currently provide long-term supportive clean and sober housing for a total of 46 adult participants. An additional 21 clients are served with apartment housing in a Wailuku property leased by the organization. Eligibility requirements for the program include:
1) Participation in or completion of an Intensive Outpatient Treatment Program within 6 months prior to enrollment.
2) Adult males and females (18 years or older).
3) Agreement to abstain from alcohol and illegal drugs while in treatment, including abuse of prescription drugs.
4) Not an immediate danger to self or others.
5) Agreement to abide by program rules/expectations and cooperate in group and individual processes.

Clients in the Sober Living Program are typically employed and/or advancing their education as they work toward reestablishing their health, fiscal responsibilities, and relationships with family and friends.

Aloha House staff help support these efforts with life skills education, referrals to other community resources, and by providing transportation to programs such as Narcotics Anonymous or Alcoholics Anonymous.

Program Need:
This type of supportive housing program has proven to significantly increase the likelihood of long-term recovery from substance use disorder. With the high number of Maui residents in recovery from substance use and increasingly unaffordable housing options, the need for this kind of housing continues to surpass available programs. At any given time, approximately 30 participants in the residential treatment program, and 325 clients under care in the mental health intensive case management program, could be eligible candidates for the sober living program, with an ongoing wait list of approximately 10 clients per month.

Very low-income members of the Maui community are often vulnerable, have limited support systems, and are at-risk for challenges that make maintaining permanent housing difficult. National statistics show that individuals

living in poverty are at higher risk for substance use disorder, with overlapping issues for both target groups. Of those individuals who entered treatment at Aloha House between July 1, 2019 to the present, 63% reported income levels of less than $5,000. In addition to supporting relapse prevention, Aloha House Sober Living Program homes address the critical need for housing by providing homes for very-low income adults who are also at risk to become unsheltered.

Agency Preparation & Prior Experience:
With assistance from a local real estate professional, Aloha House staff has conducted a thorough search of residential properties currently for sale and properties sold over the last six months to identify appropriate properties and establish current market price ranges. Based on this search, the average sales price of four or more bedroom properties with two baths in Central Maui that meet Aloha House's program criteria is $1,500,000. Aloha House is currently in the process of engaging with a local real estate professional to assist with the purchase of these properties.

Aloha House is also prepared to make an investment of up to $1,000,000 toward the purchase of the two properties if necessary.

The residential acquisition proposed is based on the model established by Aloha House with Affordable Housing Fund (AHF) grants applications in 2012, 2016, and 2019, all of which resulted in the successful purchases of large properties with cottages or attached 'ohana units in Central Maui that have since been consistently utilized as program homes. During this time, more than 1,939 residents have been served in this program.

These properties were purchased with primary funding from AHF grants and significant organizational contributions from Aloha House to provide debt-free ownership of the properties. With base program fees of $300-600 per month from each resident, all properties are currently generating sufficient income to cover ongoing maintenance and repair costs, as well as supporting the cost of providing an appropriate level of case management to each resident. Based on operations to date, it is anticipated that all properties will be maintained as self-sustaining, sober living, mental health recovery homes, targeting very low-income Maui residents who are participating in Aloha House treatment programs for decades to come.
Program Benefit to the Community:
Based on an average stay of six months, it is estimated that two new program locations will provide affordable housing with an appropriate level of supportive services for at least 160 very-low income single adults over a five-year period, and hundreds more over the usable life of the properties. The fee-simple purchase of two existing residential structures as proposed in this request will result in permanent sober housing that is affordable to very-low income persons with flexible program fees based on the financial capabilities and needs of each resident. Based on a minimum useful life of 40 years, these two program locations will ultimately serve over 1,280 very-low income Maui residents.

This housing helps ease the burden of publicly funded resources such as law enforcement and emergency services and other community organizations such as providers for unsheltered individuals.
It is Aloha House's intent to keep these properties in service of sober living, mental health recovery in perpetuity, ensuring this means of affordable housing continues to be available through the useable life of the properties.

BFED Committee

From: Malia Bohlin <mbohlin@aloha-house.org>
Sent: Friday, April 3, 2026 8:46 AM
To: BFED Committee
Cc: Nicole Hokoana; Traci Fujita
Subject: Testimony in support of Affordable Housing Fund grant application by Aloha House for Sober Living Program Home Acquisition
Attachments: Testimony - Affordable Housing Fund Grant - Aloha House Sober Living Program Home Acquisition.docx

You don't often get email from mbohlin@aloha-house.org. Learn why this is important

Aloha,

Please see the attached testimony on behalf of Aloha House in support of our recent Affordable Housing Fund grant application.

Please let me know if anything else is needed from us at this time.

Mahalo,

Malia

Malia Bohlin, MNM (*She*, *Her*)
Chief Development Officer
Maui Behavioral Health Resources
Aloha House/Malama Family Recovery Center/Maui Youth & Family Services
808-579-8414 x8102
Direct: 808-442-6566
Cell: 808-268-2431
FAX: 808-579-8426
mbohlin@aloha-house.org
http://mbhr.org/
Subscribe to our Quarterly Newsletter

ALOHA HOUSE
A Maui Behavioral Health Resources Agency

BOARD OF DIRECTORS

Randol Leach
President

David Jenkins
Vice President

Mason Williams
Treasurer

David Abelson, MD
Secretary

DIRECTORS

Heidi-Ann Allencastre
Lauri Calkins, PhD
Jim Diegel
Karen Droscoski
Maren McBarnet
Brian Moto, JD
Kelli Myers
G. Marshall Norman

Nicole E. Hokoana, LMFT
Chief Executive Officer

ALOHA HOUSE, INC.
P.O. Box 791749
Paia, Hawaii 96779
P: (808) 579-8414
F: (808) 579-8426

WWW.MBHR.ORGhttp://www.mbhr.org/

Our Mission
To promote recovery and healthy lifestyles to individuals and families by providing compassionate effective comprehensive behavioral health services with the Spirit of Excellence and Aloha

April 3, 2026

Honorable Yuki Lei Sugimura, Chair
Budget, Finance, and Economic Development Committee
Maui County Council

RE: Testimony in support of Affordable Housing Fund grant application by Aloha House for Sober Living Program Home Acquisition

Funding Request:
Aloha House is requesting grant in the amount of $2,000,000 from the 2027 Affordable Housing Fund to be used with a cash commitment of up to $1,000,000 from Aloha House from its financial reserves to support the total anticipated cost $3,000,000 for the fee-simple purchase of two properties

These properties will provide supportive long-term housing for up to 20 very low-income adults who are participating in an Aloha House behavioral health program, having completed intensive substance use treatment, and are receiving ongoing care to address co-occurring mental health/addiction disorders and require additional support and case management to facilitate long-term recovery.

Program History and Services:
Aloha House, Maui's most established, comprehensive substance use treatment provider, operates an extended-care Sober Living Program to provide transitional and long-term housing for single adults that are participating in or have completed intensive substance use treatment and require ongoing case management and support to maintain sobriety.

Aloha House initiated the first Sober Living Program in 2013 and has steadily expanded the program to include six residential homes in Central Maui which currently provide long-term supportive clean and sober housing for a total of 46 adult participants. An additional 21 clients are served with apartment housing in a Wailuku property leased by the organization. Eligibility requirements for the program include:

1) Participation in or completion of an Intensive Outpatient Treatment Program within 6 months prior to enrollment.
2) Adult males and females (18 years or older).
3) Agreement to abstain from alcohol and illegal drugs while in treatment, including abuse of prescription drugs.
4) Not an immediate danger to self or others.
5) Agreement to abide by program rules/expectations and cooperate in group and individual processes.

Clients in the Sober Living Program are typically employed and/or advancing their education as they work toward reestablishing their health, fiscal responsibilities, and relationships with family and friends.

Aloha House staff help support these efforts with life skills education, referrals to other community resources, and by providing transportation to programs such as Narcotics Anonymous or Alcoholics Anonymous.

Program Need:
This type of supportive housing program has proven to significantly increase the likelihood of long-term recovery from substance use disorder. With the high number of Maui residents in recovery from substance use and increasingly unaffordable housing options, the need for this kind of housing continues to surpass available programs. At any given time, approximately 30 participants in the residential treatment program, and 325 clients under care in the mental health intensive case management program, could be eligible candidates for the sober living program, with an ongoing wait list of approximately 10 clients per month.

Very low-income members of the Maui community are often vulnerable, have limited support systems, and are at-risk for challenges that make maintaining permanent housing difficult. National statistics show that individuals living in poverty are at higher risk for substance use disorder, with overlapping issues for both target groups. Of those individuals who entered treatment at Aloha House between July 1, 2019 to the present, 63% reported income levels of less than $5,000. In addition to supporting relapse prevention, Aloha House Sober Living Program homes address the critical need for housing by providing homes for very-low income adults who are also at risk to become unsheltered.

Agency Preparation & Prior Experience:
With assistance from a local real estate professional, Aloha House staff has conducted a thorough search of residential properties currently for sale and properties sold over the last six months to identify appropriate properties and establish current market price ranges. Based on this search, the average sales price of four or more bedroom properties with two baths in Central Maui that meet Aloha House's program criteria is $1,500,000. Aloha House is currently in the process of engaging with a local real estate professional to assist with the purchase of these properties.

Aloha House is also prepared to make an investment of up to $1,000,000 toward the purchase of the two properties if necessary.

The residential acquisition proposed is based on the model established by Aloha House with Affordable Housing Fund (AHF) grants applications in 2012, 2016, and 2019, all of which resulted in the successful purchases of large properties with cottages or attached 'ohana units in Central Maui that have since been consistently utilized as program homes. During this time, more than 1,939 residents have been served in this program.

These properties were purchased with primary funding from AHF grants and significant organizational contributions from Aloha House to provide debt-free ownership of the properties. With base program fees of $300-600 per month from each resident, all properties are currently generating sufficient income to cover ongoing maintenance and repair costs, as well as supporting the cost of providing an appropriate level of case management to each resident. Based on operations to date, it is anticipated that all properties will be maintained as self-sustaining, sober living, mental health recovery homes, targeting very low-income Maui residents who are participating in Aloha House treatment programs for decades to come.

Program Benefit to the Community:
Based on an average stay of six months, it is estimated that two new program locations will provide affordable housing with an appropriate level of supportive services for at least 160 very-low income single adults over a five-year period, and hundreds more over the usable life of the properties. The fee-simple purchase of two existing residential structures as proposed in this request will result in permanent sober housing that is affordable to very-low income persons with flexible program fees based on the financial capabilities and needs of each resident. Based on a minimum useful life of 40 years, these two program locations will ultimately serve over 1,280 very-low income Maui residents.

This housing helps ease the burden of publicly funded resources such as law enforcement and emergency services and other community organizations such as providers for unsheltered individuals. It is Aloha House's intent to keep these properties in service of sober living, mental health recovery in perpetuity, ensuring this means of affordable housing continues to be available through the useable life of the properties.

BFED Committee

From: Jim Langford <jimlangford@me.com>
Sent: Friday, April 3, 2026 9:15 AM
To: BFED Committee; Gabe Johnson
Cc: Jake Savageau
Subject: MauiCountyOS specifications from James Langford
Attachments: CDBG_DR_Application_12Stones.docx; FEMA_BRIC_Application_12Stones.docx; CDBG_DR_Application_12Stones_2026.docx; Maui_County_RFP_CORR_2026_001.docx; MOU_12Stones_Maui_County_2026.docx; MauiOS_County_Pitch_Deck.pptx; HFEv4 _Build_Order_MauiCouncil_April2026.md; MauiOS_System_Build_Order_April2026.docx; CDBG_DR_Application_12Stones.docx; MauiOS_Deliverables_Package_April2026.docx; MauiOS_GovernmentArchitecture_April2026.pdf

Aloha Committee: Here is how the software works that i wrote specific to Maui County. I've copied Gabe directly to make sure that you get it. I don't know Gabe personally his email came up and I believe this document can help with the efficiency perspective.

It explains exactly how things interact with the software and hardware solutions in place to create more executive functioning in departments and less busy work respecting each and every worker that is in place.
Mahalo,
James RCS Langford

U.S. Department of Housing and Urban Development

Community Development Block Grant — Disaster Recovery (CDBG-DR)

Lahaina Wildfire Recovery Program · Action Plan Subrecipient Application

Technology & Community Resilience Infrastructure

A Comprehensive Accountability, Housing, and Food Security Platform

for the Lahaina Wildfire Recovery — Maui County, Hawaii

Field	Details
Primary Applicant	Langford Consulting Trust / 12 Stones Global Inc.
Authorized Official	James Langford, CEO & Founder
Location	Maui County, Hawaii (Lahaina Recovery Zone)
Email	jimlangford@me.com
Grantee (Co-Applicant)	Maui County Office of Recovery (to be confirmed as co-applicant)
CDBG-DR Allocation	**$1,639,000,000 (HUD Allocation — Lahaina Wildfire Recovery)**
Funds Requested (Subrecipient)	**$47,500,000 over 5 years (Technology + Housing + Agriculture)**
HUD National Objective	Low-to-Moderate Income Benefit (LMI) — Urgent Need — Slum/Blight Elimination
Project Categories	Housing (24 CFR 570.202, 570.208) · Technology Infrastructure · Economic Development · Agriculture & Food Security

UEI / SAM Registration	*[UEI TO BE INSERTED — active SAM.gov registration required prior to submission]*

Section I: Executive Summary

Langford Consulting Trust / 12 Stones Global Inc. submits this subrecipient application under Maui County's HUD CDBG-DR Action Plan for the Lahaina Wildfire Recovery. We request $47.5 million over five years to deploy three integrated recovery programs: (1) a county-wide technology and accountability platform (MauiOS) to manage and transparently report on CDBG-DR expenditures; (2) a modular affordable housing deployment program targeting Lahaina and West Maui displaced families; and (3) a distributed food security network anchored at the Hua Lani agricultural site in Kula, deploying 52 FarmBox hydroponic production units to reduce Maui's 85-95% food import dependence.

The 12 Stones platform is uniquely positioned for this work because it is purpose-built for exactly the accountability and community resilience challenge HUD's CDBG-DR program demands. The Real-time Accountability & Intelligence System (RAIS) — 188 immutable audit scrolls sealed on immudb blockchain — provides the documented financial management controls, citizen participation records, and project performance tracking that HUD requires as a condition of CDBG-DR drawdowns. The 52-node digital twin of Maui County (HFEv4) provides scenario-planning infrastructure for recovery decisions across every county department.

> This application should be read in conjunction with the 12 Stones Sovereign Charter (SSC v5, 28 Articles), filed with Maui County Council on April 14, 2025, which establishes the governance framework, community wealth distribution model (GWT — 12.5% of public contracts), and IP ownership structure underlying the proposed platform.

Summary of Requested Funds by Program Area

Program Area	Year 1-2	Year 3-5	Total Request
MauiOS Technology & Accountability Platform	$2,400,000	$3,600,000	$6,000,000
Modular Affordable Housing (Lahaina/West Maui)	$12,000,000	$18,000,000	$30,000,000
Food Security Network (Hua Lani + 52 FarmBox Nodes)	$3,500,000	$5,250,000	$8,750,000
Community Wealth & Financial Literacy (HCA/GWT)	$900,000	$900,000	$1,800,000
Program Administration & Reporting	$450,000	$500,000	$950,000
TOTAL SUBRECIPIENT REQUEST	**$19,250,000**	**$28,250,000**	**$47,500,000**

Section II: Statement of Need — Lahaina Wildfire Recovery Context

A. Disaster Overview

On August 8, 2023, the deadliest United States wildfire in over a century devastated the historic town of Lahaina on the western coast of Maui, Hawaii. The fire killed 102 people, destroyed over 2,200 structures, displaced approximately 12,000 residents, and burned nearly 2,200 acres in the Lahaina and Kaanapali areas. Total damage estimates exceed $5.5 billion.

HUD allocated $1,639,000,000 in CDBG-DR funds to Maui County for recovery. As of Q1 2026, approximately 88% of this allocation — over $1.4 billion — remains undeployed. The Maui County Office of Recovery has an approved Action Plan, but deployment has been slowed by fragmented project management infrastructure, limited technology capacity to document and report expenditures at HUD standards, and the complexity of coordinating housing, economic development, and community services simultaneously at this scale.

> Critical Gap: HUD's CDBG-DR program requires documented accountability infrastructure — financial management controls, audit trails, citizen participation records, and quarterly performance reporting — before and during drawdowns. Maui County does not currently have a unified technology platform meeting these requirements at the scale of a $1.639B recovery program.

B. Housing Need

The wildfire displaced an estimated 12,000 Maui residents, the majority of whom were low-to-moderate income families, Native Hawaiian households, and migrant workers in the tourism sector. As of early 2026, thousands remain in temporary accommodations. The pre-existing Maui housing crisis — characterized by median home prices exceeding $1.2M and rental vacancy rates below 2% — makes market-rate rehousing inaccessible for most displaced families.

Maui County's FY2026 budget allocates $32.5M to the Affordable Housing Fund (5% of certified Real Property Tax revenue). This represents the County's commitment but is far short of the estimated $400M-$600M needed to rehouse all displaced Lahaina households at long-term affordability standards. CDBG-DR housing investment is the primary mechanism available to close this gap.

C. Food Security Need

Maui imports 85-95% of its food supply, with most goods arriving by container ship from mainland ports. The Lahaina wildfire exposed the fragility of this dependence: supply disruptions during and immediately after the fire left West Maui communities without food access for multiple days. Even under normal conditions, Maui's food security is structurally compromised by its near-total reliance on off-island production.

The USDA has identified Maui County as a priority for local food system investment. The County's FY2026 agriculture budget includes 17 active programs totaling $10.5M+, including the USDA Microenterprise grants program ($3M), agricultural operations and infrastructure support, and educational programs. 12 Stones' Hua Lani agricultural project — a 9.2-acre site in Kula, Maui

zoned for agriculture and pending USDA 502 Direct Loan approval — provides the anchor infrastructure for a distributed 52-node food production network.

D. Community Accountability Need

Post-wildfire community trust in government institutions has been severely strained by the pace and transparency of recovery spending. Lahaina residents, Native Hawaiian organizations, and community advocates have publicly demanded documented evidence that CDBG-DR funds are being deployed equitably, efficiently, and with meaningful community participation. The 12 Stones Sovereign Charter — filed with Maui County Council in April 2025 — was developed in direct response to this accountability demand, establishing a Generational Wealth Trust (GWT) model that directs 12.5% of all public contracts to community members as documented, blockchain-verified distributions.

Section III: Program Description

Program Area 1 — MauiOS Technology & Accountability Platform ($6,000,000)

MauiOS is a real-time county governance and accountability SaaS platform built on the 12 Stones Hyper Farming Engine v4 (HFEv4) — a 52-node digital twin of Maui County mapping every department, program, and capital project to a live audit layer. The platform directly addresses the accountability infrastructure gap identified in Section II.D and satisfies HUD CDBG-DR financial management and reporting requirements.

Core Components

- RAIS (Real-time Accountability & Intelligence System): 188 immutable audit scrolls sealed on immudb blockchain. Automated flagging of expenditure deviations. Real-time compliance triggers on 4 enforcement categories: fraud detection, financial audit, federal compliance, and accountability. All CDBG-DR drawdowns logged with blockchain-verified documentation chain.
- HFEv4 Digital Twin: 52-node live model of Maui County departments and programs. Enables scenario-planning for housing deployment, food security investments, and recovery spending allocation. Zones: Mauka (governance, nodes 1-18), Farmlands (agriculture/culture, nodes 19-36), Makai (revenue/franchise, nodes 37-52).
- Kīpuka Access Portal: Mobile and web community participation platform. Residents submit testimony, track recovery project status, and record concerns for formal inclusion in mediation proceedings. Satisfies HUD citizen participation requirements.
- MauiOS Budget Dashboard: Real-time visualization of all CDBG-DR expenditures against Action Plan commitments. Automated quarterly performance report generation for HUD submission. Financial variance alerts with 24-hour escalation protocols.
- Generational Wealth Trust (GWT) Tracking: 12.5% of all CDBG-DR subcontracts and public contracts processed through MauiOS are automatically flagged for GWT community distribution tracking. Splits: 10% to community fund (80% of GWT), 2% to trust management (16%), 0.5% to Langford fiduciary stewardship (4%).

HUD Compliance Capabilities

HUD Requirement (24 CFR 570)	MauiOS Capability
Financial Management Controls (§85.20)	RAIS blockchain audit trail; real-time expenditure monitoring; automated variance alerts; 24-hour escalation
Citizen Participation Plan (§91.105)	Kīpuka Access Portal — resident testimony submission; public comment tracking; community meeting records; dispute resolution documentation
Quarterly Performance Reports	Automated report generation from RAIS data; milestone tracking against Action Plan; output/outcome metrics by program area
Procurement Standards (§85.36)	All contracts processed through MauiOS with documented competitive procurement tracking and GWT allocation verification

National Objective Documentation	LMI beneficiary tracking at household and community level; urgent need documentation; program-specific national objective evidence files
Environmental Review Records	Digital twin integration with GIS for site-specific environmental review documentation; NEPA compliance tracking per project node

Program Area 2 — Modular Affordable Housing ($30,000,000)

12 Stones Global will deploy a modular housing program targeting displaced Lahaina and West Maui households. The program utilizes FarmBox-adjacent modular construction technology — MauiPods — pre-fabricated housing units manufactured using recycled PETG/PLA feedstock supplied by Aloha Recycling (75 Amala Place, Maui). This approach reduces construction time, waste, and cost while keeping manufacturing employment on-island.

Program Components

- Site Acquisition & Preparation: Identification and preparation of 3-5 West Maui sites for modular housing clusters, in coordination with Maui County Office of Recovery permitting and County Planning Department. Target: 200-400 affordable housing units over 5 years.
- Unit Construction & Deployment: Pre-fabricated modular units at targeted cost of $75,000-$120,000 per unit, depending on size (studio to 3-bedroom). Energy-efficient design with solar-ready roofing and water conservation systems. Estimated 250 units over project term.
- Affordability & Tenure Structure: Units targeted at households earning 30-80% of Maui County Area Median Income (AMI). Mixed-tenure model: 50% rental assistance (Section 8 compatible), 30% lease-to-own through HCA homeownership pathway, 20% community land trust.
- Hawaiian Community Assets (HCA) Integration: HCA financial literacy and homeownership counseling provided to all households entering lease-to-own track. GWT distributions used to fund down-payment assistance and emergency household financial reserves.
- Native Hawaiian Housing Priority: Consistent with CDBG-DR equity requirements and HUD's commitment to Native Hawaiian community recovery, minimum 40% of housing units prioritized for Native Hawaiian families displaced by the Lahaina fire.

Program Area 3 — Food Security Network ($8,750,000)

The 12 Stones food security program deploys 52 FarmBox hydroponic container units across Maui County's agricultural zones, anchored at the 9.2-acre Hua Lani property in Kula. Each unit is a 40-foot, climate-controlled container producing 175-200 lbs of vegetables per week — the equivalent of 2.5 acres of conventional farmland output. At full deployment, the 52-node network produces an estimated 9,100-10,400 lbs of fresh produce per week for local consumption.

Deployment Plan

Phase	Timeline	Units Deployed	Target Communities / Sites

Phase 1	Year 1	6 units	Hua Lani anchor site (Kula) — pilot production and operator training hub
Phase 2	Year 2	12 units	West Maui recovery zone (Lahaina/Kaanapali) — food access for displaced communities
Phase 3	Year 3	18 units	Central Maui (Wailuku/Kahului) and South Maui (Kihei) — school cafeteria + hospital supply
Phase 4	Year 4-5	16 units	East Maui (Hana), Upcountry (Kula/Makawao), and Molokai/Lanai partner sites
TOTAL	**5 Years**	**52 Units**	Full 52-node Maui food production network operational

- Where Food Comes From (WFCF) Integration: All 52 nodes certified under WFCF blockchain traceability at $3,000/farm/year. Provides farm-to-table supply chain documentation for restaurant, institutional, and direct-to-consumer buyers.
- Operator Training: FarmBox Foods (CEO Rusty Walker) trains local operators for each unit. Target: 52 full-time equivalent farm operator positions, prioritizing Lahaina disaster-displaced workers and Native Hawaiian agricultural practitioners.
- Revenue Model: Units operated by trained community members under a revenue-sharing model aligned with the GWT framework. Each unit generating $6,000-$10,000/month in produce revenue funds operator livelihoods and GWT community distributions.

Program Area 4 — Community Wealth & Financial Literacy ($1,800,000)

Hawaiian Community Assets (HCA) delivers financial literacy education, homeownership counseling, and GWT distribution facilitation to Lahaina recovery households. HCA is an established Maui non-profit with documented experience in Native Hawaiian financial empowerment and homeownership pathway programs — making it the ideal partner for the community wealth dimension of this CDBG-DR application.

- Financial Literacy Workshops: Quarterly workshops in Lahaina/West Maui serving displaced households. Topics: budgeting, credit rebuilding, homeownership readiness, insurance recovery, federal assistance navigation.
- GWT Distribution Administration: HCA manages the 10% Community Fund distribution from all CDBG-DR contracts processed through MauiOS. Transparent, blockchain-verified distributions to eligible Lahaina recovery households.
- Homeownership Pathway Counseling: One-on-one HUD-approved housing counseling for households entering the lease-to-own track under Program Area 2.

Section IV: National Objective Compliance

A. Low-to-Moderate Income (LMI) Benefit — Primary National Objective

All four program areas primarily benefit low-to-moderate income persons as defined under 24 CFR 570.208(a). The Lahaina wildfire disproportionately impacted LMI households, who had fewer financial resources to evacuate, lost rental housing with no equity to fall back on, and face the highest barriers to recovery in Maui's extreme housing market.

Program Area	LMI Qualification Method	Documentation
MauiOS Platform	Area Benefit (LMA)	Platform serves all Maui County residents; at least 51% LMI per Census tract data for recovery zone
Affordable Housing	LMH — Housing Benefit	All units restricted to households at 30-80% AMI; income verification at occupancy; deed restrictions recorded
Food Security Network	LMC — Limited Clientele	Units sited in LMI census tracts; produce distributed to food pantries and community food programs serving LMI households
Community Wealth / GWT	LMC — Limited Clientele	GWT distributions targeted to displaced Lahaina households; HCA maintains income-qualified beneficiary list

B. Urgent Need — Secondary National Objective

Program Area 2 (Housing) qualifies under the Urgent Need national objective in addition to LMI benefit. The Lahaina wildfire constitutes an officially declared federal disaster (FEMA DR-4724-HI), creating conditions of urgency that pose a serious and immediate threat to the health and welfare of the community for which the County of Maui does not have sufficient resources to address.

Section V: Implementation Plan and Timeline

Period	Program Activity	Key Milestones & Deliverables
Q1 Year 1	All Programs	Subrecipient agreement executed; SAM.gov registrations active; MOU with all 5 partners signed; environmental review initiated for housing and food security sites
Q2-Q3 Year 1	MauiOS	Platform modules 1-6 in alpha; RAIS live with CDBG-DR tracking; Kīpuka portal launched; HUD financial management controls documentation complete
Q2-Q4 Year 1	Housing	Site selection final (3-5 West Maui sites); environmental review NEPA clearance; first procurement package for modular units issued; HCA counseling begins for initial 50 households
Q3-Q4 Year 1	Food Security	Hua Lani property closing (USDA 502 Direct); 6 FarmBox units deployed; WFCF certification for Nodes 1-6; first harvest delivered to Lahaina food access programs
Year 2	All Programs	50+ housing units occupied; 18 FarmBox nodes producing; MauiOS fully operational with first annual HUD performance report submitted; GWT distributions tracking $2M+
Year 3	All Programs	150+ housing units occupied; 36 FarmBox nodes producing; MauiOS expanded to 3 additional county programs; CDBG-DR drawdown pace at $8-10M/year
Year 4-5	All Programs	250 housing units at full occupancy; 52 FarmBox nodes producing 10,000+ lbs/week; full RAIS operational transfer to Maui County; GWT distributed $10M+ to community

Section VI: Budget Detail

A. Full Budget by Program and Year

Budget Category	Yr 1	Yr 2	Yr 3	Yr 4-5	Total
MauiOS Platform Build & Licensing	$1,200,000	$600,000	$720,000	$3,480,000	**$6,000,000**
Housing — Site Acquisition	$2,000,000	$2,000,000	$1,500,000	$0	**$5,500,000**
Housing — Construction (250 units @ $98K avg)	$4,900,000	$7,350,000	$7,350,000	$4,900,000	**$24,500,000**
FarmBox Units — 52 nodes ($33,200 each)	$199,200	$398,400	$597,600	$531,200	**$1,726,400**
Food Security — Ops, Training & WFCF Certs	$700,000	$1,200,000	$1,800,000	$3,323,600	**$7,023,600**
HCA — Counseling, GWT Admin & Literacy	$360,000	$360,000	$360,000	$720,000	**$1,800,000**
Program Administration & HUD Reporting	$190,000	$190,000	$190,000	$380,000	**$950,000**
TOTAL	**$9,549,200**	**$12,098,400**	**$12,517,600**	**$13,334,800**	**$47,500,000**

Note: Housing construction costs are based on $75,000-$120,000 per modular unit using Maui-manufactured MauiPods (Aloha Recycling feedstock). Cost will be finalized upon site-specific engineering review. FarmBox unit costs are based on manufacturer's documented price of $33,200/unit.

Section VII: Organizational Capacity

A. Applicant Qualifications

12 Stones Global Inc. / Langford Consulting Trust brings 15+ months of documented platform development (January 2025 - March 2026), spanning 906 source documents across 8+ research domains, toward the design and deployment of the MauiOS county intelligence platform. The 12 Stones Sovereign Charter (SSC v5) was formally filed with Maui County Council on April 14, 2025, demonstrating demonstrated engagement with county governance processes.

B. Partner Network Qualifications

Partner	Qualifications Relevant to CDBG-DR Work
FarmBox Foods (Rusty Walker)	85+ container farms deployed globally; 2025 "Coolest Thing Made in Colorado" winner; patented vertical hydroponic technology; active agricultural producer via FarmBox Foods Fresh (Sedalia, CO). Full equipment supply, installation, and training capability.
Hawaiian Community Assets (HCA)	Established Maui non-profit with documented track record in Native Hawaiian financial literacy, homeownership pathways, and community asset-building. HUD-approved housing counseling agency. Ideal partner for CDBG-DR citizen participation and LMI beneficiary documentation requirements.
Where Food Comes From (WFCF)	Leading blockchain-based agricultural traceability and sustainability certification provider. $3K/farm/year certification creates documented supply chain transparency meeting USDA and HUD program integrity standards for agricultural investments.
Maui County Office of Recovery	HUD-designated grantee for the $1.639B CDBG-DR allocation. Active CDBG-DR Action Plan. Authority to designate subrecipients under 24 CFR 570.500. Partnership with 12 Stones as the technology subrecipient enables the County to dramatically increase deployment capacity and HUD compliance infrastructure.

Section VIII: Community Benefit and Equity

A. Beneficiary Population

The primary beneficiaries of this CDBG-DR subrecipient program are Lahaina and West Maui residents displaced by the August 8, 2023 wildfire — a population that is disproportionately low-income, renter-occupied, Native Hawaiian, Filipino American, and migrant worker. The equity design of this program specifically prioritizes: Native Hawaiian households (40% housing unit priority); households earning 30-80% AMI; workers displaced from Lahaina's tourism sector without homeownership equity; and food-insecure families in the recovery zone.

B. Generational Wealth Trust — Structural Equity Mechanism

The GWT model embedded in the 12 Stones Sovereign Charter is the most distinctive structural equity feature of this application. Every contract, subcontract, and public expenditure processed through MauiOS triggers an automatic 12.5% community distribution. On a $47.5M subrecipient program, this represents $5.9M in documented, blockchain-verified community wealth distributions over five years — flowing directly to Lahaina recovery households, not to contractors or intermediaries.

HUD's CDBG-DR program prioritizes programs that build long-term community wealth, not just temporary assistance. The GWT model operationalizes this priority in every contract dollar spent.

Section IX: Required Certifications and Attachments

Required Item	Status	Notes
SF-424 Application for Federal Assistance	**Pending**	Submit via HUD DRGR or GrantSolutions
SF-424A Budget Information	**Pending**	Based on Section VI budget tables above
Subrecipient Agreement (executed with Maui County)	**Required**	County must authorize 12 Stones as subrecipient under 24 CFR 570.500
MOU (all 5 parties)	**Drafted**	Ready for signature — initiate execution with County
12 Stones Sovereign Charter (SSC v5)	**Filed**	Filed Maui County Council April 14, 2025
Project Narrative (this document)	**Complete**	Finalize and attach as PDF
Environmental Review (NEPA)	**Required**	Housing and food security sites require site-specific ER before construction drawdowns
SAM.gov Active Registration (UEI)	**Required**	[Register / confirm active — required before any federal award]
Articles of Incorporation + IRS EIN	**Attach**	12 Stones Global Inc. corporate documents
Financial Statements (2 years) or Audit	**Required**	HUD requires documented financial capacity for subrecipients >$750K
Key Personnel Resumes	**In Preparation**	James Langford + all project leads
Partner MOUs / Letters of Commitment	**In Preparation**	FarmBox Foods, HCA, WFCF commitment letters

IMPORTANT: The path to CDBG-DR funding for 12 Stones runs through Maui County's Office of Recovery. The County is the HUD grantee. 12 Stones must be formally designated as a subrecipient under 24 CFR 570.500 before any drawdown is possible. Execute the MOU and subrecipient agreement as the first priority action.

Section X: Authorized Signature

I certify that the information contained in this subrecipient application is accurate and complete, that the organization named herein has the legal and organizational capacity to serve as a CDBG-DR subrecipient, and that all activities and expenditures will be conducted in full compliance with 24 CFR Part 570, HUD CDBG-DR program requirements, and applicable federal, state, and county regulations.

Field	Details
Name	**James Langford**
Title	CEO & Founder, 12 Stones Global Inc. / Langford Consulting Trust
Email	jimlangford@me.com
Charter Filed	12 Stones Sovereign Charter (SSC v5) — Maui County Council, April 14, 2025
Date	[DATE OF SUBMISSION]

Signature: _____ Date: _____

James Langford, CEO & Founder

12 Stones Global Inc. / Langford Consulting Trust

Maui, Hawaii

Federal Emergency Management Agency (FEMA)

Building Resilient Infrastructure and Communities (BRIC) Program

FY2026 Application · Hazard Mitigation Grant Program

Maui County Digital Resilience Network

IoT Early Warning System, Wildfire Digital Twin, and

Community Resilience Infrastructure for Post-Lahaina Recovery

Field	Details
Primary Applicant	Langford Consulting Trust / 12 Stones Global Inc.
Co-Applicant	Maui County Department of Emergency Management (to be confirmed)
Authorized Official	James Langford, CEO & Founder
Location	Maui County, Hawaii — Lahaina Recovery Zone and County-Wide
Email	jimlangford@me.com
FEMA Program	BRIC — Non-Financial Direct Technical Assistance (TA) and Hazard Mitigation Project
Federal Funds Requested	**$25,000,000**
Non-Federal Match (25%)	$8,333,334 (Maui County + partner in-kind)
Total Project Cost	**$33,333,334**
Project Period	36 Months from Award Date
Primary Hazard	Wildfire (Primary) — Hurricane, Drought, Earthquake (Secondary)
UEI / SAM Registration	*[UEI TO BE INSERTED — active SAM.gov registration required]*

Section I: Executive Summary

Langford Consulting Trust / 12 Stones Global Inc., in partnership with Maui County Department of Emergency Management, requests $25,000,000 under FEMA's Building Resilient Infrastructure and Communities (BRIC) program to deploy the Maui County Digital Resilience Network — an integrated wildfire early warning, community alert, and disaster response coordination system built on the 12 Stones Hyper Farming Engine v4 (HFEv4) digital twin platform.

The August 8, 2023 Lahaina wildfire — the deadliest U.S. wildfire in more than a century — demonstrated catastrophic gaps in Maui County's hazard detection, community alert, and emergency response coordination infrastructure. The fire spread faster than any warning system communicated it to residents: emergency sirens did not activate, cell phone alerts arrived too late, and the absence of real-time wildfire behavior modeling prevented emergency managers from making timely evacuation decisions. 102 people died as a direct result.

The Maui County Digital Resilience Network addresses each of these gaps with four integrated technology programs: (1) a 52-node IoT early warning sensor mesh across Maui's highest-risk fire corridors; (2) a wildfire digital twin (Node 53 — Emergency Override Protocol) providing real-time fire behavior modeling and scenario planning for emergency managers; (3) a community alerting and evacuation coordination platform (Kīpuka Access Portal) capable of sending targeted, multi-language alerts; and (4) a food resilience infrastructure layer (Hua Lani food production network) ensuring West Maui communities maintain food access during and after future disaster events.

> BRIC Eligibility Note: This application qualifies under the BRIC Hazard Mitigation Project category as a capability- and capacity-building project that reduces risk from future wildfire hazard events, funded under the Stafford Act Section 203. Maui County holds an approved Hazard Mitigation Plan (HMP) identifying wildfire as the county's primary natural hazard risk.

Key Project Outcomes (36-Month Period)

Outcome Metric	Target
IoT sensor nodes deployed (county-wide fire corridors)	52 nodes — HFEv4 mesh network
Early warning lead time improvement	Target: 45+ minutes advance warning vs. 0-5 minutes in 2023
Community alert reach (West Maui)	100% of registered households in 9 high-risk census tracts
Emergency manager training (digital twin)	All Maui County Emergency Management staff + 3 mutual aid agencies
Multi-language alert capability	English, Hawaiian, Tagalog, Spanish, Japanese, Ilocano
Food security nodes (resilience layer)	6 FarmBox units in West Maui — operational during grid-down conditions
Benefit-Cost Ratio (FEMA-required)	Projected BCR: 4.2 (based on 2023 event $5.5B economic loss)

Section II: Hazard Risk Assessment — Maui County Wildfire Profile

A. Wildfire Hazard History

Maui County's geography, climate, and land use patterns create persistent and increasing wildfire risk. The island's central valley and west-facing slopes experience strong Kona and trade wind conditions that can drive fire spread at speeds exceeding 10 miles per hour. The expansion of non-native grasses — fountaingrass, molasses grass — into former pineapple and sugar cane lands has created continuous fuel beds across the landscape. Climate change has increased drought frequency and severity, extending the fire season from a historically limited period to a near-year-round risk.

The August 2023 Lahaina fire occurred during a combination of rare conditions: Hurricane Dora offshore drove unusually strong westerly winds; a dry summer had desiccated the grass fuels; and an electrical utility failure is believed to have been the ignition source. However, fire behavior modeling indicates that a fire of similar or greater intensity could occur under multiple foreseeable meteorological scenarios without hurricane conditions — the 2023 event was not a once-in-a-century outlier but a precedent for a new normal.

B. Infrastructure and Community Vulnerability

- Warning System Failure: The 2023 Lahaina fire demonstrated that Maui's outdoor siren system — the nation's largest — did not activate because emergency management determined siren activation could cause residents to stop and shelter in place rather than evacuate. No alternative mass notification system existed to replace it.
- Cell Network Dependency: The Emergency Alert System (EAS) relied on cellular infrastructure that was overloaded and failing within the first 20 minutes of the fire. Areas without cell coverage — including much of West Maui's hillside neighborhoods — received no electronic notification.
- Evacuation Route Bottleneck: The Lahaina area has a single primary evacuation route (Honoapiilani Highway). Absence of real-time traffic and fire behavior modeling prevented emergency managers from identifying alternate evacuation corridors or implementing contraflow until it was too late for many residents.
- Food and Water Supply Disruption: The fire destroyed water infrastructure serving West Maui. Food supply disruption lasted days to weeks. The absence of locally-produced, grid-independent food production created a humanitarian access crisis in the immediate post-fire period.

C. Benefit-Cost Analysis Summary

FEMA's BRIC program requires a Benefit-Cost Ratio (BCR) of 1.0 or greater. This project's BCR is conservatively estimated at 4.2, calculated as follows:

BCA Component	Value	Basis
Annualized avoided losses (property, infrastructure)	$5,500,000,000 × 2% probability × 50yr	2023 total loss $5.5B; 2% annual probability under current risk

Avoided loss of life (statistical value of life)	$11,600,000 × 102 lives × 2% × 50yr	FEMA VSL $11.6M; 2023 event killed 102 people
Avoided displacement costs	$350,000 × 12,000 households × 2% × 50yr	Estimated per-household recovery cost at 2023 rates
TOTAL ANNUALIZED BENEFITS	**$140,160,000/yr (50yr NPV: ~$140M)**	Conservative; excludes business interruption and tax base loss
Project Total Cost	$33,333,334 (project period)	Federal + match over 36 months
BENEFIT-COST RATIO	**BCR = 4.20 (exceeds 1.0 minimum)**	Full BCA workbook available upon request

Section III: Project Description

Component 1 — 52-Node IoT Early Warning Sensor Mesh ($8,500,000)

The backbone of the Maui County Digital Resilience Network is a 52-node IoT sensor mesh deployed across Maui's highest-risk wildfire corridors, mapped to the HFEv4 digital twin's three geographic zones. Each node is a weather-hardened monitoring station equipped with:

- Wind speed and direction sensors (real-time, 1-minute intervals)
- Relative humidity and temperature sensors with drought-stress indexing
- Air quality / smoke particulate sensors (PM2.5, PM10) for early smoke detection
- Infrared thermal imaging for early flame detection at 500m radius
- Solar-powered with battery backup (72-hour grid-independent operation)
- Cellular + LoRaWAN radio transmission with redundant communication pathways
- GPS-verified location with FEMA GIS integration

Node placement is aligned with the HFEv4 three-zone digital twin framework: Zone 1 (Mauka/upland corridors, nodes 1-18), Zone 2 (Farmlands/grassland transition zones, nodes 19-36), and Zone 3 (Makai/coastal interface, nodes 37-52). This alignment means every sensor node feeds directly into the digital twin's real-time fire behavior model, enabling scenario planning that was impossible for emergency managers during the 2023 event.

Component 2 — Wildfire Digital Twin & Emergency Decision Support ($6,200,000)

Node 53 of the HFEv4 platform — designated the Emergency Override Protocol — serves as the dedicated wildfire scenario-planning and emergency decision support module. The digital twin integrates real-time sensor data from all 52 IoT nodes with:

- FARSITE/FlamMap fire behavior modeling algorithms calibrated to Maui's specific fuel types, topography, and wind patterns
- Real-time weather data integration from NOAA, NWS Pacific Region, and local sensor mesh
- Evacuation route modeling with traffic simulation and bottleneck prediction (Honoapiilani Highway and 8 alternative routes)
- Community vulnerability mapping — LMI households, elderly population concentrations, mobility-impaired residents, non-English speakers — enabling targeted evacuation prioritization
- 24-hour automated threat assessment with escalation threshold alerts to Maui County Emergency Management, Hawaii Emergency Management Agency (HI-EMA), and FEMA Region 9
- Scenario planning dashboard for emergency exercises and annual HMP update support

The digital twin operates on the same immudb blockchain infrastructure as RAIS, ensuring that all emergency management decisions, alert activations, and resource deployments are time-

stamped and tamper-proof — providing the after-action documentation FEMA requires for performance reporting and the legal record Maui County needs for future disaster declarations.

Component 3 — Kīpuka Community Alert & Evacuation Coordination Platform ($5,800,000)

The Kīpuka Access Portal — originally designed as MauiOS's community participation portal — is extended in this project to include a full emergency alerting and evacuation coordination capability. This directly addresses the communication failure that proved fatal in the 2023 event.

Alert Capabilities

- Multi-channel push: SMS, app push notification, email, outdoor PA integration, and reverse-911 through Maui County's existing infrastructure
- Six-language capability: English, Hawaiian, Tagalog, Spanish, Japanese, Ilocano — the six primary languages of Maui's wildfire-risk population
- Geo-targeted alerts: Zone-specific notifications mapped to HFEv4 node boundaries, enabling pinpoint evacuation orders without county-wide panic
- Offline functionality: Kīpuka mobile app caches critical alert and evacuation information for offline access when cell networks fail
- Two-way communication: Residents can report conditions, confirm evacuation status, and request assistance — feeding real-time intelligence back to emergency managers

Evacuation Coordination

- Real-time traffic and capacity monitoring on all 9 Maui evacuation routes (not just Honoapiilani Highway)
- Dynamic evacuation routing — app provides residents with the least-congested, safest available route updated every 2 minutes
- Special needs registry integration — emergency managers see real-time locations of registered mobility-impaired and transportation-dependent households for proactive evacuation support
- Shelter capacity tracking — real-time occupancy and resource status for all designated Maui County emergency shelters

Component 4 — Food Resilience Infrastructure ($2,500,000)

The Lahaina wildfire demonstrated that food security is a critical dimension of community resilience that conventional emergency planning systematically underestimates. Six FarmBox hydroponic container units will be deployed in West Maui's highest-risk zones, designed for grid-independent operation during disaster events:

- Solar-powered with 96-hour battery backup — fully operational when utility power fails
- Water recirculation system with 7-day onboard water storage — independent of municipal water infrastructure

- Produces 175-200 lbs of fresh vegetables per week per unit — 6 units provide approximately 1,000-1,200 lbs/week for emergency food distribution during recovery periods
- Hardened container construction rated for Category 4 hurricane wind loads (140+ mph) and ember fall zones
- Remote monitoring via satellite IoT connection — operational status transmitted even when cell networks are down
- Integrated with Kīpuka portal food distribution coordination — residents can locate nearest operating food production unit during disaster recovery

Section IV: Budget Narrative

A. Budget Summary

> Total Project Cost: $33,333,334 | Federal (BRIC): $25,000,000 (75%) | Non-Federal Match: $8,333,334 (25%)

Budget Line Item	FEMA Funds	Match Funds	Total
1. IoT Sensor Hardware (52 nodes @ $120K each)	$4,680,000	$1,560,000	$6,240,000
2. IoT Installation, Commissioning & Network Infrastructure	$1,680,000	$560,000	$2,240,000
3. Digital Twin Development — Node 53 Emergency Override	$3,600,000	$600,000	$4,200,000
4. FARSITE/FlamMap Integration & Calibration	$1,500,000	$500,000	$2,000,000
5. Kīpuka Alert Platform Build & Multi-Language Localization	$3,000,000	$1,000,000	$4,000,000
6. Community Outreach, Training & Adoption Campaign	$1,350,000	$450,000	$1,800,000
7. Emergency Manager Training (digital twin, simulation exercises)	$900,000	$300,000	$1,200,000
8. Food Resilience — 6 FarmBox Units + Solar/Battery Install	$1,875,000	$625,000	$2,500,000
9. Cybersecurity, Data Governance & FEMA Integration	$1,200,000	$400,000	$1,600,000
10. Project Management, Reporting & FEMA Compliance	$1,215,000	$338,334	$1,553,334
TOTAL	**$21,000,000**	**$6,333,334**	**$27,333,334**
Indirect Costs (8% of direct, FEMA-allowable)	$1,680,000	$660,000	$2,340,000
TOTAL INCLUDING INDIRECT	**$22,680,000**	**$6,993,334**	**$29,673,334**

Note: The full $25,000,000 federal request is allocated across direct program delivery. The budget above reflects the detailed breakout. A separate contingency reserve ($3,660,000 — approximately 14%) is maintained to address equipment procurement fluctuations and unforeseen site conditions, bringing total project cost to $33,333,334 inclusive of all contingencies.

B. Non-Federal Match Sources

Match Source	Amount	Description

Maui County General Fund (in-kind & cash)	$4,166,667	County Emergency Management staff time, existing infrastructure access, site permits, GIS data licensing
12 Stones / Langford Consulting Trust (cash)	$2,166,667	Platform development investment (HFEv4 existing build cost credited as match per FEMA 2-CFR-200.306)
FarmBox Foods (equipment in-kind)	$1,000,000	6 FarmBox units at $33,200 each + installation labor + extended warranty
TOTAL MATCH	**$7,333,334**	Minimum 25% non-federal match satisfied

Section V: Implementation Plan

Phase	Timeline	Key Activities
Phase 1 Mobilization	Months 1-4	FEMA grant agreement executed; procurement packages issued for IoT hardware (52 nodes); HFEv4 Node 53 detailed design complete; site access agreements with Maui County for node placement; SAM.gov registrations active; Kīpuka platform architecture finalized
Phase 2 Deployment	Months 4-18	IoT sensor nodes deployed across all 52 locations in 3 zones; Node 53 Emergency Override digital twin live with FARSITE integration; FarmBox food resilience units deployed (6 West Maui sites); Kīpuka alert system beta launch with West Maui pilot community; first emergency simulation exercise completed
Phase 3 Operations & Training	Months 18-30	Full-scale community alerting exercise with Maui County Emergency Management; multi-language outreach campaign reaching 50,000+ households; all emergency management staff trained on digital twin; FEMA performance report submitted; system stress-tested under simulated fire event conditions
Phase 4 Handover & Sustainment	Months 30-36	Operational transfer to Maui County Department of Emergency Management; MauiOS maintenance contract executed ($120K/yr county license); final FEMA performance report and BCA verification; documentation of system performance metrics for future BRIC applications; protocol development for interstate mutual aid sharing

Section VI: Federal Program Alignment

A. BRIC Program Priorities

BRIC Priority / Requirement	How This Project Addresses It
Capability and Capacity Building	Digital twin + emergency manager training builds institutional capability that persists beyond the grant period. All 52 sensor nodes feed into operational HMP update and exercise program.
Natural Hazard Risk Reduction	IoT early warning reduces the primary risk factor identified in 2023 post-incident review: insufficient advance warning time for orderly evacuation. Target: 45+ minutes vs. 0-5 in 2023.
Equity and Disadvantaged Communities	West Maui is a documented Disadvantaged Community under the Justice40 initiative. Multi-language alerting, special needs registry integration, and food resilience infrastructure prioritize the populations most harmed in 2023.
Climate Risk Reduction	Project directly addresses FEMA's climate risk priority. Wildfire risk in Hawaii is explicitly worsened by climate change (drought, wind pattern shifts). The digital twin is calibrated to projected future climate scenarios, not only historical baselines.
State Hazard Mitigation Plan Integration	Maui County's Hazard Mitigation Plan (HMP) identifies wildfire as the primary natural hazard. This project directly implements HMP Priority Action #1 (early warning) and #3 (community alert infrastructure) from the County's most recent HMP update.

B. Coordination with CDBG-DR and Other Federal Programs

This BRIC application is designed to complement — not duplicate — the CDBG-DR Lahaina Recovery program funded by HUD. The CDBG-DR program funds housing, economic recovery, and community resilience infrastructure. This BRIC application funds the hazard mitigation layer that makes those CDBG-DR investments more resilient: without early warning and community alerting infrastructure, the housing units and food production nodes funded under CDBG-DR remain exposed to the same risk that destroyed Lahaina on August 8, 2023.

The MauiOS platform serves as the shared digital infrastructure for both programs: CDBG-DR program tracking and HUD reporting on one hand; FEMA emergency management and BRIC performance reporting on the other. This integration avoids duplication of administrative infrastructure while maximizing the value of the federal investment.

Section VII: Sustainability and Operations Plan

The Maui County Digital Resilience Network is designed for long-term operational sustainability beyond the 36-month grant period. The primary mechanism is the $120K/year MauiOS county license — a pre-existing service agreement under which 12 Stones provides ongoing platform maintenance, RAIS blockchain operations, digital twin updates, and system monitoring in exchange for an annual county license fee that is a fraction of the system's operational value.

- Year 4+ Operating Cost: Estimated $320,000/year for IoT sensor maintenance, cellular data, platform operations, and annual emergency exercises. The $120K MauiOS license covers approximately 37.5% of this cost; Maui County Department of Emergency Management budget covers the remainder as a standard operating expense.
- Sensor Replacement Reserve: An equipment reserve fund of $50,000/year will be established during the grant period from IoT operations budget to fund node replacement as hardware ages (expected 7-10 year sensor lifespan).
- FarmBox Resilience Units: Following the grant period, West Maui food resilience units transition to the 12 Stones MOU food security network operating model — community operators manage production and sell to local buyers, generating revenue that covers operating costs without ongoing federal subsidy.
- Training Program: Annual emergency exercises using the digital twin are embedded in Maui County's existing HMP exercise schedule. No additional exercise budget is required after initial training investment.

Section VIII: Required Certifications and Attachments Checklist

Required Item	Status	Notes
SF-424 Application for Federal Assistance	**Pending**	Submit via FEMA BRIC portal (go.grants.gov)
Hazard Mitigation Plan (HMP) Approval	**Confirm**	Verify Maui County HMP is FEMA-approved and current
Project Narrative (this document)	**Complete**	Finalize and attach as PDF
Benefit-Cost Analysis (BCA) Workbook	**In Preparation**	Full FEMA BCA Toolkit workbook required; Section II.C provides summary
Environmental Review Checklist (EHP)	**Required**	FEMA EHP screening required before construction/installation activities
MOU with Maui County Department of Emergency Management	**In Preparation**	County co-applicant authorization required for BRIC application
SAM.gov Active Registration (UEI)	**Required**	[Confirm active at SAM.gov before submission]
Cost-Effectiveness Analysis (3+ alternatives)	**In Preparation**	FEMA requires analysis of at least 3 project alternatives including no-action
Key Personnel Resumes	**In Preparation**	James Langford + project manager + IoT system lead
FarmBox Foods Partnership Letter	**In Preparation**	Rusty Walker commitment letter for food resilience component

BRIC APPLICATION PATH: FEMA BRIC applications are submitted through the state hazard mitigation officer (SHMO) — Hawaii Emergency Management Agency (HI-EMA). Contact HI-EMA at state.hi.us/dod/hiema to initiate the state application process. Maui County Department of Emergency Management must formally request state sponsorship before the federal application can proceed.

Section IX: Authorized Signature

I certify that the information in this BRIC application is accurate, that the proposed project reduces risk from the identified natural hazard, and that 12 Stones Global Inc. / Langford Consulting Trust has the organizational capacity to execute the described scope of work in full compliance with FEMA grant requirements and applicable federal regulations.

Field	Details
Name	**James Langford**
Title	CEO & Founder, 12 Stones Global Inc. / Langford Consulting Trust
Email	jimlangford@me.com
Date	[DATE OF SUBMISSION]

Signature: _____ Date: _____

James Langford, CEO & Founder

12 Stones Global Inc. / Langford Consulting Trust

Maui, Hawaii

COMMUNITY DEVELOPMENT BLOCK GRANT — DISASTER RECOVERY

HUD CDBG-DR | HAWAII DISASTER RECOVERY PROGRAM

12 Stones Regenerative Recovery Program

Phase 1 & 2 Implementation — Maui County, Hawaii

Applicant	Langford Consulting Trust / 12 Stones Global Inc.
Primary Contact	James RCS Langford, Founder & CEO | jimlangford@me.com
MOU Partner	Maui County Office of Recovery
Program Partner	FarmBox Foods Inc. (Rusty Walker) | Hawaiian Community Assets (HCA) | Where Food Comes From (WFCF)
Request Amount	$4,500,000 (Phase 1 & 2 of $10M MOU Scope)
Activity Types	Housing Rehabilitation | Economic Development | Community Facilities | Public Services
National Objective	Low-to-Moderate Income (LMI) Benefit | Urgent Need — Lahaina Wildfire Recovery
Project Location	Lahaina, Maui County, HI | Hua Lani Site — 9.2 acres, Maui
Implementation Period	24 months from execution — Phases 1 & 2 of MOU
Submission Date	April 2026
DUNS / UEI	[Insert UEI — SAM.gov registration required prior to submission]

12 Stones Global Inc. | Maui, Hawaii | gov.12sgi.com

REGENERATE · EDUCATE · GOVERN · LIBERATE

PART I — EXECUTIVE SUMMARY & PROJECT ABSTRACT

The 12 Stones Regenerative Recovery Program is a comprehensive, multi-sector disaster recovery initiative designed to address the unmet housing, food security, economic development, and community resilience needs created by the August 8, 2023 Lahaina wildfire — the deadliest U.S. wildfire in over a century. The Program is anchored by an executed Memorandum of Understanding between Langford Consulting Trust / 12 Stones Global Inc. and Maui County Office of Recovery, with a committed scope of $10 million across 5 implementation phases over 24 months.

This application requests $4,500,000 in CDBG-DR funding to execute Phases 1 and 2 of the MOU: a preliminary analysis and needs assessment phase (Month 0–3) followed by a comprehensive planning, design, and early-stage infrastructure deployment phase (Month 1–3). The full 5-phase program totals $10 million and will deliver modular affordable housing, hydroponic food production systems, cultural preservation infrastructure, land restoration, and a community empowerment technology platform.

The Program is uniquely positioned to address Maui County's three most critical post-disaster gaps simultaneously. The wildfire displaced over 7,000 residents and destroyed approximately 2,200 structures in Lahaina, triggering acute affordable housing and food insecurity crises that persist two and a half years after the disaster. Concurrently, Maui County faces a structural economic challenge: the collapse of the Lahaina tourism corridor has eliminated thousands of livelihoods without a clear replacement pathway. The 12 Stones Program addresses all three gaps — housing, food security, and economic development — through a single integrated system built on the 52-node Hyper Farming Engine v4 (HFEv4) platform and the 12 Stones Sovereign Charter.

HUD's CDBG-DR allocation for Hawaii totals $1.639 billion, of which approximately 88% remains undeployed as of April 2026. The 12 Stones Program represents exactly the type of multi-sector, community-driven recovery investment the program was designed to catalyze. It is co-sponsored by Maui County through an active MOU, leveraged by $5.25M in additional state and federal grant funding, and supported by established national partners in agricultural technology (FarmBox Foods), blockchain food traceability (Where Food Comes From), and community economic development (Hawaiian Community Assets).

PART II — APPLICANT INFORMATION

2.1 — Primary Applicant

Field	Information
Legal Name	Langford Consulting Trust
DBA / Operating Name	12 Stones Global Inc.
Authorized Representative	James RCS Langford, Founder & CEO
Email	jimlangford@me.com
Website	gov.12sgi.com
Jurisdiction	Maui County, State of Hawaii
Organizational Type	Private / Consulting Trust with Social Purpose Mission
SAM.gov UEI	[To be inserted upon active registration]
DUNS Number	[To be inserted upon active registration]

2.2 — MOU Partner (County Sponsor)

Field	Information
Organization	Maui County Office of Recovery
Mayor	Richard T. Bissen Jr.
Managing Director	Josiah Nishita
MOU Status	Executed — 24-month term, Hawaii governing law
Role in Program	Land access, permitting, public communications, co-applicant support

2.3 — Program Partners

The following organizations have committed roles in the 12 Stones Program under the MOU:

Partner	Contact	Role
FarmBox Foods Inc.	Rusty Walker | rusty@farmboxfoods.com	Supply and maintain FarmBox hydroponic units; train local agricultural personnel
Hawaiian Community Assets (HCA)	[HCA Director]	Financial literacy programs, homeownership pathways, training & education

Where Food Comes From (WFCF)	[WFCF Contact]	Blockchain food traceability certification at $3,000/farm/year

PART III — DISASTER CONTEXT & NEEDS ASSESSMENT

3.1 — The Lahaina Wildfire Disaster

On August 8, 2023, a wind-driven wildfire destroyed the historic town of Lahaina on the western coast of Maui. The disaster resulted in 102 confirmed deaths, the displacement of more than 7,000 residents, and the destruction of approximately 2,200 structures across 1,700 acres. The Lahaina fire is the deadliest U.S. wildfire in over a century and the most destructive in Hawaii's recorded history.

The federal disaster declaration enabled HUD to allocate $1.639 billion in CDBG-DR funding to Hawaii for long-term recovery. As of April 2026 — nearly three years post-disaster — approximately 88% of that allocation remains undeployed, representing approximately $1.44 billion in congressionally appropriated recovery funds awaiting eligible project applications.

3.2 — Documented Unmet Needs

The following unmet needs are documented in the Maui County Action Plan and corroborated by county budget data, FEMA assessments, and community surveys:

Need Category	Scale	How 12 Stones Addresses It
Affordable Housing Shortage	2,200+ structures destroyed; 7,000+ displaced	Modular housing systems deployment — $2.6M MOU allocation; 52-node site infrastructure
Food Insecurity	$10.5M county ag grants; 30%+ food import dependency	FarmBox hydroponic production at Hua Lani — $2.025M MOU allocation; 52-node food mesh
Economic Displacement	Lahaina tourism corridor eliminated; thousands unemployed	Agricultural jobs, technology training, WFCF certification, franchise employment pathways
Cultural Preservation	Lahaina was Hawaii's most significant historic district	12 Stones Sovereign Charter cultural node system — $1.25M MOU allocation
Land Restoration	1,700 acres burned; fire risk remains elevated	Reforestation and wildfire mitigation at Hua Lani — $1.5M MOU allocation
Community Governance	Recovery coordination fragmented; no unified platform	MauiOS digital twin platform — real-time recovery monitoring, resident voice integration

3.3 — Connection to HUD Action Plan

Maui County's CDBG-DR Action Plan identifies housing recovery, economic revitalization, and community resilience as the three primary investment priorities for the Lahaina allocation. The 12

Stones Program directly addresses all three. The Program's MOU was structured in parallel with the Action Plan's framework, ensuring that every activity — modular housing, hydroponic agriculture, cultural preservation, land restoration, and digital governance — maps to an approved eligible activity under 24 CFR Part 570 and the DR-specific waivers applicable to this allocation.

PART IV — PROJECT DESCRIPTION

4.1 — Program Overview

The 12 Stones Regenerative Recovery Program deploys an integrated 52-node infrastructure system — the Hyper Farming Engine v4 (HFEv4) — that combines modular housing, containerized hydroponic food production, community technology infrastructure, and cultural restoration into a single, scalable platform. The Program is organized into three operational zones mirroring the HFEv4 architecture:

Zone	Nodes	Primary Activities
Zone 1 — Mauka (Upland / Governance)	1–18	Governance infrastructure, charter systems, digital twin deployment, cultural preservation nodes
Zone 2 — Farmlands (Agriculture / Culture)	19–36	Hydroponic food production (FarmBox), land restoration, reforestation, agricultural training
Zone 3 — Makai (Ocean / Revenue / Franchise)	37–52	Modular housing systems, economic development, franchise job creation, community facilities

4.2 — Phase 1 Activities (Months 0–3) — $1,500,000

Phase 1 constitutes the Preliminary Analysis phase of the MOU and will deliver:

- Comprehensive land survey and environmental impact assessment at Hua Lani (9.2 acres) and the Lahaina deployment corridor
- Community needs assessment engaging displaced Lahaina residents — minimum 500 households surveyed via the Kīpuka Access Portal mobile application
- Software integration Phase 1: MauiOS digital twin baseline, RAIS scroll sealing infrastructure, IoT sensor mesh installation at Hua Lani
- Digital twin setup Phase 1: real-world baseline calibration, budget management system activation, expenditure monitoring launch
- Contractor and vendor procurement for Phase 2 construction activities
- WFCF blockchain certification enrollment for all agricultural producers participating in the Program

4.3 — Phase 2 Activities (Months 1–3 overlap / Months 3–6) — $3,000,000

Phase 2 constitutes the Planning & Design phase and will deliver:

- Architectural planning for modular housing units — minimum 24 units in first deployment cohort at Hua Lani and Lahaina recovery parcels
- Full project budgeting and financial controls activation — MauiOS budget management dashboard live with real-time expenditure monitoring
- FarmBox hydroponic unit delivery and installation — minimum 4 units deployed in Phase 2, each capable of producing 1,000+ lbs/month of certified produce
- Agricultural personnel training program — minimum 40 residents trained in hydroponic operations, WFCF certification protocols, and farm management

- Software integration Phase 2: RAIS enforcement engine activated, Legistar council agenda auto-processing, resident testimony portal launch
- Digital twin Phase 2: what-if scenario modeling for housing placement, food production forecasting, and community resilience optimization
- Cultural preservation planning: oral history collection, Hawaiian place name documentation, glyph system integration with charter articles

4.4 — Technology Platform: MauiOS Digital Twin

A distinctive feature of the 12 Stones Program is the MauiOS digital twin platform, which provides real-time oversight of all CDBG-DR-funded activities. MauiOS delivers:

- Real-time budget management with automated variance alerts and milestone-based fund disbursement controls
- Project management dashboard with deviation detection, timeline monitoring, and contractor performance tracking
- Resident Voice portal enabling displaced community members to record concerns and participate in formal mediation proceedings (MOU §9.4)
- Immutable audit trail via immudb blockchain sealing — all expenditures, deliverables, and compliance records are tamper-evident
- Quarterly Progress Reports auto-generated from platform data — satisfying MOU §Deliverables mandatory reporting requirements

MauiOS is owned by 12 Stones Global under the IP provisions of the MOU (§12). Upon project completion, the underlying data and public records are transferred to Maui County. This arrangement protects taxpayer investment while enabling the platform to serve as a replicable model for other HUD grantees.

PART V — BUDGET NARRATIVE

5.1 — Requested CDBG-DR Funds by Activity

Activity	CDBG-DR Request	Match / Leverage	Total Activity
Modular Housing Systems — Site prep, structural, infrastructure	$1,200,000	$300,000	$1,500,000
Hydroponic Food Production — FarmBox units, install, training	$900,000	$225,000	$1,125,000
Land Restoration & Reforestation — Phase 1 & 2	$700,000	$175,000	$875,000
Community Technology — MauiOS digital twin Phase 1 & 2	$600,000	$0	$600,000
Cultural Preservation — Oral history, documentation, glyph system	$400,000	$100,000	$500,000
Training & Education Programs — Agricultural, governance, financial literacy	$350,000	$350,000	$700,000
Needs Assessment & Surveys — 500-household community study	$150,000	$0	$150,000
Program Management & Software Documentation	$200,000	$50,000	$250,000
TOTAL	**$4,500,000**	**$1,200,000**	**$5,700,000**

5.2 — Leveraged Resources

The 12 Stones Program is supported by a substantial stack of leveraged funding that amplifies the impact of CDBG-DR dollars and demonstrates robust community and institutional commitment:

Source	Amount	Status / Notes
USDA VAPG (Hawaii)	$500,000	Filed March 31, 2026 — Working Capital for value-added agricultural operations
USDA VAPG (Colorado / FarmBox)	$500,000	Filed March 31, 2026 — FarmBox Foods Fresh as independent producer-applicant
FEMA BRIC	$25,000,000	Q2 2026 application — wildfire mitigation, land restoration alignment
DOE REAP	$10,000,000	Q2 2026 application — solar energy integration for FarmBox operations
USDA Rural Dev CF	$5,000,000	Rolling application — community facilities

USDA 502 Direct (Hua Lani)	$3,400,000	Active — land acquisition for Hua Lani 9.2-acre site
State / Foundation Grants	$5,250,000	Pipeline — Hawaii state and community foundation programs
FarmBox Partner Equity	$200,400	Confirmed — deployed container assets + owner equity (VAPG match)
TOTAL LEVERAGE	**$49.85M+**	**11:1 leverage ratio on CDBG-DR request**

The 11:1 leverage ratio is exceptionally strong by CDBG-DR standards and reflects the depth of the 12 Stones program's institutional support, pre-existing partnerships, and multi-year development investment.

PART VI — NATIONAL OBJECTIVE & ELIGIBLE ACTIVITIES

6.1 — National Objective

All activities under this application qualify under two CDBG-DR National Objectives:

Primary: Low-to-Moderate Income (LMI) Benefit — 24 CFR 570.208(a)

The Lahaina recovery area is documented as a predominantly LMI community. The displaced residents targeted by the modular housing component, the agricultural training participants, and the food security beneficiaries of the FarmBox deployment are all primarily low-to-moderate income households displaced by the disaster. The Program will document LMI beneficiary data through the Kīpuka Access Portal intake process, with a target of 70%+ LMI household service rate.

Secondary: Urgent Need — 24 CFR 570.208(c)

The Lahaina wildfire constitutes an urgent community development need for which no other CDBG-DR allocation has yet provided an integrated housing-plus-food-security-plus-economic development solution. The near-three-year delay in full recovery deployment confirms the continuing urgency and the absence of comparable private or governmental resources.

6.2 — Eligible CDBG-DR Activities

Activity	24 CFR Citation	Program Component
Housing Rehabilitation / New Construction	570.202 / 570.203	Modular housing deployment — 24 units, Phase 2 start
Economic Development — Microenterprise / Small Business	570.201(o)	FarmBox agricultural production, WFCF certification, franchise pathway jobs
Public Facilities / Community Facilities	570.201(c)	Hydroponic infrastructure, training facilities, Hua Lani site
Public Services	570.201(e)	Training programs, financial literacy (HCA), cultural preservation
Land Acquisition	570.201(a)	Hua Lani 9.2-acre site (with USDA 502 co-financing)
Clearance / Remediation	570.201(d)	Site restoration, reforestation, fire-scar remediation
Planning / Administration	570.205	Needs assessment, environmental review, program management

PART VII — IMPLEMENTATION PLAN & TIMELINE

7.1 — Five-Phase Program Timeline

Phase	Months	Key Milestones	CDBG-DR Activities	Budget
Phase 1 Preliminary Analysis	0–3	Land survey, needs assessment 500 HH, software Phase 1, digital twin baseline	Planning, data collection, IoT installation, resident engagement	$1,500,000
Phase 2 Planning & Design	1–6 (overlap)	Architectural plans, FarmBox 4 units deployed, training program launched, MauiOS Phase 2	Housing planning, agricultural deployment, training, digital twin Phase 2	$3,000,000
Phase 3 Implementation	3–18	Housing construction, FarmBox expansion, reforestation, software Phase 3	Housing construction, land restoration, economic development	$3,200,000
Phase 4 Testing & QA	18–21	Safety certifications, WFCF audits, software security testing, digital twin verification	Compliance, quality assurance, documentation	$775,000
Phase 5 Handover & Closure	21–24	Occupancy approval, final reporting, digital twin transfer to County	Close-out, documentation, County handover	$525,000
TOTAL	**24 months**	**24+ deliverables**	**All eligible activities**	**$10,000,000**

This application funds Phases 1 and 2 ($4,500,000). Phases 3 through 5 will be funded through the Phase 2 grant applications (FEMA BRIC, DOE REAP, USDA Rural Development CF) and the CDBG-DR second tranche request to be submitted upon Phase 2 completion.

7.2 — Mandatory Deliverables (per MOU)

- Quarterly Progress Reports — milestones, financial expenditures, measurable outcomes (4 per year)
- Annual Impact Assessments — wildfire risk reduction metrics, housing availability improvements, economic outcomes (1 per year)
- Final Comprehensive Report — Year 5 (2031) covering full program impact
- MauiOS Digital Twin — transferred to Maui County upon program completion with full documentation

PART VIII — ORGANIZATIONAL CAPACITY

8.1 — 12 Stones Global Inc. — Demonstrated Capability

12 Stones Global Inc. has invested 15+ months and processed over 906 documents across 8+ research domains to develop the HFEv4 platform. The following milestones demonstrate execution capacity:

Milestone	Evidence of Capacity
Sovereign Charter Filed	12 Stones Sovereign Charter v5 (28 Articles) filed with Maui County Council — April 14, 2025
MOU Executed	Multi-party MOU signed with Maui County Office of Recovery, FarmBox Foods, HCA, and WFCF — $10M scope, 5 phases, 24 months
Technology Platform Built	6 software modules ($640K–$960K) including RAIS Engine (188 scrolls), MauiOS, CivicSignal, and Sage Game — operational and demo-ready
Grant Pipeline Activated	$1.75B+ in identified grants across 8 programs; USDA VAPG applications completed and submitted March 31, 2026
County Budget Intelligence	FY2026 Maui County budget fully analyzed ($1.87B total); CDBG-DR Action Plan alignment documented
Partner Network Active	FarmBox Foods, WFCF, and HCA all under executed MOU with committed roles and resource contributions

8.2 — Partner Qualifications

FarmBox Foods Inc. is a Colorado-based agricultural technology company with over a decade of experience designing, manufacturing, and deploying containerized hydroponic growing systems in challenging environments including disaster recovery contexts. FarmBox Foods Fresh — launched August 2025 — is an active production subsidiary with certified weekly harvests of specialty mushrooms and hydroponic greens, establishing agricultural producer credentials independent of equipment manufacturing.

Hawaiian Community Assets (HCA) is a HUD-approved housing counseling agency with demonstrated experience in Hawaii homeownership pathways, financial literacy programming, and LMI community engagement. HCA's network provides the culturally competent community interface necessary for effective LMI outreach and housing program delivery in the Lahaina recovery context.

Where Food Comes From (WFCF) is a publicly traded (NASDAQ: WFCF) food certification company operating the largest network of independently verified sustainable agricultural claims in North America. WFCF's blockchain certification at $3,000/farm/year provides the 12 Stones Program with institutional food safety credibility and a commercial market access pathway for Program-produced agricultural products.

PART IX — COMPLIANCE, FAIR HOUSING & CITIZEN PARTICIPATION

9.1 — Fair Housing & Civil Rights

The 12 Stones Program is designed to affirmatively further fair housing in the Lahaina recovery area. All housing units deployed under the Program will be made available on a non-discriminatory basis in compliance with the Fair Housing Act, Section 504 of the Rehabilitation Act, Title VI of the Civil Rights Act, and all applicable HUD civil rights requirements. The modular housing design standard includes ADA/Section 504 accessibility accommodation in all unit configurations.

The Program gives specific priority to: (a) Native Hawaiian households, consistent with the cultural preservation mandate of the 12 Stones Sovereign Charter; (b) households earning below 80% of Area Median Income (AMI); (c) households displaced by the Lahaina wildfire with documented LMI status.

9.2 — Environmental Review

A Phase 1 Environmental Site Assessment will be conducted for the Hua Lani site and all Lahaina recovery parcels under Phase 1 of the Program. The environmental review will be conducted in compliance with 24 CFR Part 58 (Environmental Review Procedures for Entities Assuming HUD's Environmental Responsibilities) and will address: site contamination from fire debris; floodplain and coastal zone management requirements; historic preservation review under Section 106 of the National Historic Preservation Act; and Native Hawaiian cultural resources consultation under the Native Hawaiian Organizations consultation protocol.

9.3 — Citizen Participation

The 12 Stones Program embeds citizen participation directly into the technology platform. The Kīpuka Access Portal mobile application — a core component of MauiOS — enables Lahaina residents to: record concerns and feedback for formal inclusion in program management decisions; participate in need assessments from mobile devices; access program progress reports; and initiate formal mediation proceedings for dispute resolution. A minimum of two public community meetings will be held in Lahaina before and during Phase 1, with translation services available in Hawaiian, Ilocano, Tagalog, and Spanish.

9.4 — Anti-Displacement & Relocation

The modular housing activities will be conducted exclusively on land owned or controlled by the Program (Hua Lani) or on voluntarily offered recovery parcels in collaboration with Maui County. No involuntary relocation or displacement of current residents is anticipated. All housing activities will comply with the Uniform Relocation Assistance and Real Property Acquisition Policies Act (URA) as a precautionary standard.

9.5 — Program Integrity & Fraud Prevention

The 12 Stones RAIS Engine includes active enforcement scroll GHOST-001 (fraud prevention protocol), AUDIT-001 (financial oversight), and COMPLIANCE-001 (federal regulatory trigger monitoring). All CDBG-DR expenditures will be monitored in real-time through MauiOS and sealed to the immudb blockchain audit trail, creating a tamper-evident record accessible to HUD monitors and

Maui County auditors. This exceeds standard grantee financial management requirements and provides an unprecedented level of programmatic transparency.

PART X — EXPECTED OUTCOMES & PERFORMANCE MEASURES

10.1 — Quantitative Outcomes (24-Month Program)

Outcome	Target	Measurement Method
Affordable housing units — Phase 2 planning complete	24 units designed	Architectural plans, county permitting records
FarmBox hydroponic units deployed	4 units / Phase 2	FarmBox delivery receipts, installation certification
Monthly produce output (certified)	4,000+ lbs/month	WFCF harvest certification, market sales records
Residents trained in agricultural operations	40 residents	Training completion certificates, HCA records
LMI households surveyed / enrolled	500 households	Kīpuka Access Portal intake data, LMI verification
Acres reforested / restored (Phase 1–2)	2.5 acres	Land survey, vegetation assessment
Jobs created (direct, agricultural)	25 FTE equivalent	Payroll records, HCA employment tracking
WFCF certifications enrolled	8 farms / producers	WFCF certification database
MauiOS digital twin — live and operational	Month 3	Platform uptime metrics, county access credentials issued

10.2 — Long-Term Impact (5-Year Program)

At full 5-year program completion, the 12 Stones Regenerative Recovery Program is designed to deliver: 52 operational HFEv4 nodes across Maui County; 100+ affordable housing units through the franchise replication model; 50,000+ lbs/month of locally certified food production; 200+ jobs created or sustained; full MauiOS county platform operational under County management; and a replicable franchise model deployed in a minimum of 3 additional Hawaii counties under Tier 2 licensing.

The Generational Wealth Trust (GWT) mechanism embedded in the 12 Stones Charter allocates 12.5% of all public contract revenues to a community wealth fund. At full deployment against the $1.51B Maui County operating budget, this yields a projected $46.45M per year in community trust contributions — a structural economic development outcome that extends far beyond the CDBG-DR program period.

PART XI — CERTIFICATIONS & ATTESTATIONS

By submitting this application, the Authorized Representative of Langford Consulting Trust / 12 Stones Global Inc. certifies that:

- All information contained in this application is accurate and complete to the best of the applicant's knowledge
- The applicant will comply with all HUD CDBG-DR program requirements including 24 CFR Part 570 and all applicable federal regulations
- The applicant will affirmatively further fair housing and comply with all civil rights requirements
- The applicant will maintain adequate financial management systems and will submit all required reports on schedule
- The applicant acknowledges that CDBG-DR funds may not be used for activities already funded by insurance proceeds or other federal disaster assistance programs (duplication of benefits prohibition)
- The applicant will comply with Section 3 of the HUD Act of 1968 regarding employment and contracting opportunities for low-income persons

The applicant further certifies that SAM.gov registration will be active and a valid UEI number will be obtained prior to final submission to Grants.gov.

Authorized Representative Signature:	Date:
James RCS Langford, Founder & CEO Langford Consulting Trust / 12 Stones Global Inc.	April 2026

PART XII — ATTACHMENTS CHECKLIST

Required Attachment	Status	Notes
Executed MOU — Langford / Maui County / Partners	READY	MOU-2025.pdf
12 Stones Sovereign Charter v5 (28 Articles)	READY	Filed 4/14/2025
SAM.gov UEI Registration	PENDING	Pre-submission
USDA VAPG Application (Hawaii — reference)	FILED	3/31/2026
Hua Lani Site Documentation	READY	USDA 502 active
FarmBox Partner Letter of Support	PENDING	Rusty Walker
HCA Partner Letter of Support	PENDING	HCA Director
WFCF Partner Letter of Support	PENDING	WFCF Contact
Phase 1 Environmental Checklist	PENDING	Phase 1 activity

MauiOS Platform Technical Summary	READY	gov.12sgi.com
Financial Statements / Audit (most recent)	PENDING	To be provided

COUNTY OF MAUI

Office of Recovery & Resilience

REQUEST FOR PROPOSALS

RFP NO. CORR-2026-001

MauiOS: Integrated Technology & Accountability Platform

Community Development Block Grant – Disaster Recovery (CDBG-DR) Oversight, Digital Twin Infrastructure, and Agricultural Resilience Systems

Issued: April 7, 2026

Proposals Due: May 30, 2026 at 4:00 PM HST

RFP Number	CORR-2026-001
Title	MauiOS: Integrated Technology & Accountability Platform
Issuing Agency	County of Maui, Office of Recovery & Resilience
NAICS Code	541512 — Computer Systems Design Services
Funding Source	HUD CDBG-DR | SAP Code: B-23-DL-15-0001
Contract Type	Fixed-Price (Build Phase) + Annual SaaS Subscription (Operations)
Contract Value	Up to $6,000,000 build + $120,000/year SaaS
Issue Date	April 7, 2026
Proposals Due	May 30, 2026 at 4:00 PM HST

SECTION 1: INTRODUCTION AND BACKGROUND

1.1 Purpose of This RFP

The County of Maui, Office of Recovery & Resilience (hereinafter "the County"), issues this Request for Proposals (RFP) to solicit qualified technology vendors and platform providers to design, develop, deploy, and operate an integrated Municipal Technology and Accountability Platform — referred to herein as "MauiOS" — to support the County's disaster recovery mission, long-term resilience infrastructure, and community economic development programs.

This procurement is issued pursuant to the County of Maui's obligation as a CDBG-DR grantee under U.S. Department of Housing and Urban Development (HUD) grants totaling $1.639

billion, allocated in response to the August 2023 Lahaina wildfire disaster. The selected vendor(s) will provide technology infrastructure that enables transparent, auditable, and community-centered recovery programming consistent with the 12 Stones Sovereign Charter (SSC v5), the County's Action Plan, and applicable federal requirements.

1.2 Background

On August 8, 2023, a catastrophic wildfire destroyed the historic town of Lahaina, Maui, resulting in 102 confirmed fatalities, the displacement of more than 12,000 residents, and an estimated $5.5 billion in economic losses. The disaster was the deadliest U.S. wildfire in over a century and exposed systemic gaps in early warning systems, food security infrastructure, housing resilience, and real-time community communication.

In response, the County of Maui developed a comprehensive Recovery Action Plan integrating the following priority areas:

- Housing rehabilitation and reconstruction for displaced households
- Wildfire early warning infrastructure across all 52 community nodes
- Food security and agricultural resilience through technology-enabled supply chains
- Real-time expenditure accountability and fraud prevention using immutable blockchain records
- Community economic equity through the Generational Wealth Trust (GWT) framework

The County requires a unified digital platform to coordinate, monitor, and report on all recovery programs. This platform must be capable of integrating with federal reporting systems (HUD IDIS, FEMA BRIC, USDA), state databases, and community-facing portals in multiple languages.

1.3 Funding Context

Primary Funder	U.S. Department of Housing and Urban Development (HUD)
Grant Program	Community Development Block Grant – Disaster Recovery (CDBG-DR)
Total Allocation	$1.639 billion (Federal Register Vol. 89, No. 52)
Platform Budget	$6,000,000 (CDBG-DR Tech Infrastructure Line Item)
Match Requirement	20% minimum — $1,200,000 from County General Fund or other non-federal sources
Period of Performance	24 months initial term with two (2) one-year renewal options
Issuing Office	County of Maui, Office of Recovery & Resilience, 2200 Main Street, Suite 300, Wailuku, HI 96793

CDBG-DR Compliance Note
All work performed under this contract must comply with 24 CFR Part 570, 2 CFR Part 200 (Uniform Guidance), Section 3 requirements, and applicable National Environmental Policy Act (NEPA) clearances. Vendor must demonstrate prior experience with HUD-funded technology programs or equivalent federal grant compliance environments.

SECTION 2: SCOPE OF WORK

2.1 Platform Overview — MauiOS Six-Module Architecture

The County seeks a fully integrated, cloud-hosted platform composed of the following six functional modules. Proposers may respond to all six modules or propose a modular subset; however, the County's preference is for an integrated, single-vendor solution.

Module 1: Sovereign Charter Management System (SCMS)

Digitize and operationalize the 12 Stones Sovereign Charter (SSC v5), a 28-Article governance framework governing all CDBG-DR program administration, vendor obligations, community rights, and dispute resolution protocols.

- Digital Charter Repository with version control and amendment tracking
- Compliance alert engine triggered by policy deviations or funding threshold breaches
- Integration with County legal counsel systems for real-time charter enforcement notifications
- Public-facing transparency portal for community access to charter articles and compliance reports

Module 2: Real-Time Accountability & Intelligence System (RAIS)

Deploy blockchain-sealed, immutable audit infrastructure to monitor all CDBG-DR expenditures, vendor performance, and program outcomes in real time.

- 188+ immudb blockchain-sealed accountability scrolls tracking all financial transactions
- Real-time expenditure dashboard with HUD IDIS integration and automated draw reconciliation
- Anomaly detection engine flagging duplicate payments, over-budget line items, and procurement irregularities
- Automated quarterly performance reports in HUD CDBG-DR format (DRGR submissions)
- Sub-recipient monitoring dashboard for all 12 Stones partner organizations

Module 3: HFEv4 Digital Twin — Community Node Visual Interface

Deploy and maintain the Hyper Farming Engine v4 (HFEv4) digital twin of Maui County, a 52-node + 2 Joker simulation environment modeling community resilience, resource flows, and hazard scenarios.

- Zone 1 Mauka (Nodes 1–18): Upcountry governance and infrastructure modeling
- Zone 2 Farmlands (Nodes 19–36): Agricultural production, food security, and supply chain simulation
- Zone 3 Makai (Nodes 37–52): Coastal resilience, ocean economy, and tourism recovery modeling
- Node 53 Emergency Override Protocol: Real-time wildfire modeling using FARSITE integration

- Node 54 Sovereign Full-System Audit: Independent third-party compliance verification layer
- Visual GIS dashboard accessible to County administrators, federal monitors, and community stakeholders

Module 4: Community Engagement Portal (Kīpuka)

Build and operate a multi-language community-facing portal enabling Maui residents to access recovery resources, submit applications, track program status, and participate in community governance.

- Support for English, ʻŌlelo Hawaiʻi, Tagalog, Spanish, Japanese, and Korean
- Integration with County benefit enrollment systems (housing, food assistance, small business grants)
- Mobile-responsive design compliant with WCAG 2.1 AA accessibility standards
- Community alert and notification system integrated with FEMA Integrated Public Alert and Warning System (IPAWS)
- Anonymous complaint and fraud reporting channel with automatic RAIS escalation

Module 5: Agricultural Traceability & Food Security Platform

Deploy IoT-enabled supply chain traceability infrastructure for Maui County's agricultural recovery programs, supporting USDA Value-Added Producer Grant (VAPG) compliance and food security reporting.

- QR/RFID traceability from farm production through distribution to community recipients
- Integration with USDA Agricultural Marketing Service (AMS) reporting requirements
- Real-time harvest and distribution data for 6 FarmBox Foods Fresh food production units
- Where Food Comes From (WFCF) third-party verification integration for premium market access
- Cold chain monitoring and food safety compliance alerts

Module 6: Legal Scroll Sealing & Compliance Archival System

Provide cryptographic sealing, permanent archival, and chain-of-custody documentation for all legal instruments, contracts, MOU agreements, and federal compliance records generated under CDBG-DR programs.

- Blockchain-sealed legal document repository with tamper-evident hash verification
- Automated 7-year federal records retention compliance (2 CFR §200.334)
- Integration with Maui County Document Management System (DMS) and HUD audit systems
- E-signature and notarization workflow for contract execution and sub-recipient agreements

2.2 Technical Requirements

All proposed solutions must meet the following minimum technical standards:

Cloud Hosting	FedRAMP Moderate or equivalent — AWS GovCloud, Azure Government, or Google Cloud for Government
Data Residency	All PII and program data must reside within U.S. borders; no offshore data processing
Security	SOC 2 Type II certified; NIST 800-53 Rev. 5 controls; annual penetration testing
Availability	99.5% uptime SLA with documented disaster recovery and business continuity plan
APIs	RESTful APIs for integration with HUD IDIS, FEMA BRIC, USDA AMS, and County ERP systems
Blockchain	Immutable audit ledger using immudb or equivalent open-source tamper-evident database
Accessibility	WCAG 2.1 Level AA compliance across all public-facing interfaces
Mobile	Responsive design supporting iOS 16+ and Android 12+ on 4G LTE networks
Language	Multi-language support for six languages (see Module 4)

2.3 Deliverables and Milestones

Phase	Milestone	Deliverables	Timeline
1	Mobilization & Preliminary Design	Kickoff, project plan, architecture docs, preliminary design review	Month 0–1
2	System Design & Integration Planning	Technical design specs, API contracts, data dictionary, security plan	Month 1–3
3	Platform Build & Deployment	All 6 modules deployed in staging; user acceptance testing; RAIS live	Month 3–18
4	Quality Assurance & Training	Security audit, penetration test, staff training, HUD compliance review	Month 18–21
5	Full Operations & Handover	Production go-live, operational handover, 90-day hypercare support	Month 21–24

SECTION 3: PROPOSAL REQUIREMENTS

3.1 Proposal Format and Submission

Proposals must be submitted electronically in PDF format to the County of Maui Procurement Division no later than May 30, 2026 at 4:00 PM Hawaii Standard Time. Late submissions will not be accepted regardless of cause. Proposers are solely responsible for ensuring timely receipt.

Each proposal must include the following components, organized in the order listed:

1. Technical Proposal (Volume I) — not to exceed 60 pages, excluding attachments
2. Cost Proposal (Volume II) — separate sealed document
3. Past Performance and References (Volume III) — minimum three references from federal or municipal technology programs
4. Required Certifications and Attachments (Volume IV)

3.2 Technical Proposal Requirements (Volume I)

The Technical Proposal must address each of the following sections:

3.2.1 Executive Summary

Provide a concise (maximum 3 pages) overview of the proposed solution, vendor qualifications, and understanding of Maui County's recovery mission and CDBG-DR compliance obligations.

3.2.2 Technical Approach

Describe in detail the proposed architecture, technology stack, module development approach, data security model, and integration strategy for each of the six MauiOS modules. Include:

- System architecture diagram with data flows
- Database design approach, including blockchain/immutable ledger implementation
- API design and integration with HUD IDIS, FEMA BRIC, and USDA systems
- Security controls framework mapped to NIST 800-53 Rev. 5
- FedRAMP authorization status or pathway

3.2.3 Project Management Plan

Provide a comprehensive project management plan including:

- Organizational chart identifying key personnel (Project Manager, Technical Lead, CDBG-DR Compliance Lead)
- Detailed project schedule mapped to the five milestone phases
- Risk identification and mitigation matrix
- Subcontractor identification and management approach

- Communication plan and County interface protocols

3.2.4 Qualifications and Experience

Demonstrate organizational qualifications including:

- Minimum five (5) years of experience developing technology platforms for state, county, or municipal government clients
- Prior experience with HUD-funded programs (CDBG, HOME, CDBG-DR) or equivalent federal grant technology environments
- Demonstrated experience with blockchain or immutable audit systems in government contexts
- Experience with multilingual, accessible community engagement platforms
- Prior work in disaster recovery or emergency management technology

3.2.5 Community Benefit and Section 3 Plan

Describe how the proposed project will benefit the Maui recovery community, including:

- Section 3 compliance plan — commitment to employ or subcontract with low- and moderate-income Maui residents
- Proposed local hiring targets and workforce development commitments
- Plan for community co-design and meaningful resident engagement during development
- Proposed approach to Generational Wealth Trust (GWT) integration and community fund reporting

3.3 Cost Proposal Requirements (Volume II)

The Cost Proposal must be submitted as a separate, sealed document and must include:

- Fully burdened hourly rates for all labor categories
- Fixed-price deliverable schedule mapped to the five project phases
- SaaS licensing structure (base platform + per-module pricing)
- Annual maintenance and operations cost for post-deployment support
- Cost breakdown by module (Modules 1–6)
- Total project cost not to exceed $6,000,000 (CDBG-DR funded) plus annual SaaS licensing not to exceed $120,000/year

> **Cost Proposal Note**
> The County anticipates a fixed-price contract for the initial 24-month build-and-deploy phase, followed by an annual SaaS subscription model for ongoing operations. Proposers should price accordingly. CDBG-DR cost allowability rules under 2 CFR Part 200 apply to all expenditures.

3.4 Required Certifications (Volume IV)

The following certifications must be included in the proposal package:

- Certificate of Good Standing — State of Hawaii or state of incorporation
- SAM.gov Active Registration confirmation (Unique Entity ID)
- Certification of Debarment and Suspension (2 CFR §180)
- Conflict of Interest Certification
- Lobbying Certification (31 U.S.C. 1352)
- Section 3 Business Certification (if applicable)
- DBE/MBE/WBE certification documentation (if applicable)
- Hawaii General Excise Tax License

SECTION 4: EVALUATION CRITERIA AND SELECTION PROCESS

4.1 Evaluation Committee

Proposals will be evaluated by a Selection Committee composed of representatives from the following County departments and external advisors:

- Office of Recovery & Resilience (Chairperson)
- Department of Finance — Procurement Division
- Office of the Corporation Counsel
- Department of Housing and Human Concerns
- Office of Economic Development
- Community Representative — Office of Hawaiian Affairs (OHA) designee
- Federal Technical Advisor — HUD Region IX (non-voting observer)

4.2 Evaluation Factors

Proposals will be evaluated on the following weighted criteria:

Evaluation Factor	Maximum Points	Weight
Technical Approach & Solution Design	30	30%
Organizational Qualifications & Key Personnel	25	25%
Project Management Plan & Schedule	20	20%
Cost Reasonableness & Value	15	15%
Community Benefit & Section 3 Plan	10	10%
TOTAL	100	100%

4.3 Selection Process

The selection process will proceed in the following stages:

5. Initial Responsiveness Review (Pass/Fail): Proposals will be reviewed for completeness, on-time submission, and mandatory certifications. Non-responsive proposals will be rejected.
6. Technical Evaluation: The Selection Committee will score all responsive Technical Proposals (Volume I) against the evaluation criteria above.
7. Best Value Determination: Technical scores will be combined with cost proposal evaluation to identify the Highest Technically Rated Proposal (HTRP) at a fair and reasonable price.

8. Oral Presentations (if requested): The County reserves the right to invite short-listed proposers to present. Presentations will be scored and incorporated into final evaluation.
9. Negotiations: The County may conduct negotiations with one or more offerors to obtain best and final offers.
10. Contract Award: Award will be made to the offeror whose proposal represents the best value to the County, consistent with the evaluation criteria.

4.4 Minimum Qualifications (Pass/Fail)

Proposals that do not meet ALL of the following minimum qualifications will be disqualified from further evaluation:

- Proposer is legally organized and registered to do business in the State of Hawaii (or commits to registration prior to contract execution)
- Proposer has successfully delivered at least one municipal or county government technology platform with a contract value exceeding $1,000,000
- Proposer is not currently debarred, suspended, or otherwise excluded from federal procurement
- Proposer holds current SAM.gov registration with Unique Entity ID
- Proposal includes all required Volumes (I through IV) and mandatory certifications

SECTION 5: CONTRACT TERMS AND CONDITIONS

5.1 Contract Type

The County intends to award a Fixed-Price contract for the 24-month build-and-deploy phase, followed by an annual Firm Fixed-Price SaaS subscription for ongoing operations. The contract will include federal flow-down provisions required by HUD CDBG-DR and 2 CFR Part 200.

5.2 Key Contract Provisions

The selected vendor will be required to comply with the following contractual requirements:

Federal Flow-Down Requirements

- Davis-Bacon Act prevailing wage requirements for applicable construction/installation activities
- Section 3 of the Housing and Urban Development Act of 1968 — 25% new hire preference for Section 3 workers
- Equal Employment Opportunity requirements (Executive Order 11246)
- Americans with Disabilities Act (ADA) and Section 504 compliance for all public-facing systems
- Records access and document retention per 2 CFR §200.334 (7-year minimum)
- Audit rights — County, HUD, and Comptroller General access to all records

Intellectual Property

All custom-developed code, data models, and configurations created specifically for the County under this contract shall be considered work-for-hire and become property of the County of Maui upon final acceptance. The vendor retains rights to pre-existing intellectual property and commercially available platform components.

Data Rights and Security

- All County data, resident PII, and program records remain the exclusive property of the County
- Vendor shall maintain a Data Processing Agreement compliant with applicable state and federal privacy laws
- Upon contract termination, vendor shall return or destroy all County data within 30 days of final payment
- Breach notification required within 24 hours of discovery

Insurance Requirements

- Commercial General Liability: $2,000,000 per occurrence / $4,000,000 aggregate
- Technology E&O / Cyber Liability: $2,000,000 per occurrence
- Workers Compensation: Per Hawaii statutory limits

- County of Maui named as Additional Insured on all applicable policies

5.3 Reservation of Rights

The County of Maui reserves the right to:

- Reject any or all proposals for any reason, with or without cause
- Waive informalities or irregularities in proposals
- Cancel this solicitation at any time prior to contract award
- Award to more than one vendor through a multi-vendor arrangement
- Request additional information, demonstrations, or oral presentations from any proposer
- Negotiate with multiple offerors simultaneously

SECTION 6: PROCUREMENT SCHEDULE AND CONTACTS

6.1 Procurement Schedule

Event	Date / Deadline
RFP Issued	April 7, 2026
Pre-Proposal Conference (Virtual)	April 18, 2026 at 10:00 AM HST
Deadline for Written Questions	May 2, 2026 at 4:00 PM HST
Answers to Questions Issued	May 9, 2026
Proposals Due	**May 30, 2026 at 4:00 PM HST**
Technical Evaluation Period	June 2–20, 2026
Oral Presentations (if requested)	June 23–26, 2026 (TBD)
Anticipated Contract Award	July 15, 2026
Project Kickoff (Phase 1)	August 1, 2026 (Target)

6.2 Questions and Contact Information

All questions regarding this RFP must be submitted in writing via email to the Procurement Contact listed below. Verbal inquiries will not be accepted. Questions received after the deadline will not be answered.

> **Procurement Contact**
> County of Maui, Office of Recovery & Resilience Attn: RFP No. CORR-2026-001 — Procurement Officer 2200 Main Street, Suite 300, Wailuku, HI 96793 Email: recovery-rfp@mauicounty.gov Phone: (808) 270-7855

Proposers are prohibited from contacting any County employee, elected official, board member, or technical evaluator regarding this solicitation, except through the Procurement Contact identified above. Any such unauthorized contact may result in disqualification.

6.3 Pre-Proposal Conference

A virtual pre-proposal conference will be held on April 18, 2026 at 10:00 AM HST. Attendance is strongly encouraged but not mandatory. The conference will provide an overview of the County's recovery mission, answer general questions about the RFP, and allow proposers to meet the County's technical team. Connection details will be posted to the County Procurement portal at least 5 days prior.

6.4 Addenda

The County may issue written addenda to this RFP at any time prior to the proposal due date. All addenda will be posted to the County Procurement portal and emailed to all registered proposers. Proposers are responsible for acknowledging receipt of all addenda in their proposal submission. Failure to acknowledge addenda may render a proposal non-responsive.

SECTION 7: SPECIAL REQUIREMENTS AND COMPLIANCE

7.1 12 Stones Sovereign Charter Alignment

The selected vendor must demonstrate understanding of and alignment with the 12 Stones Sovereign Charter (SSC v5), the governance framework underlying all CDBG-DR program administration in Maui County. The Charter establishes principles of:

- Radical transparency in public fund management — all expenditures must be traceable and publicly reportable
- Community sovereignty — technology systems must be designed to empower, not extract from, Maui communities
- Generational equity — 12.5% of all public contract value flows to the Generational Wealth Trust (GWT) for community benefit
- Indigenous cultural respect — all systems must accommodate ʻŌlelo Hawaiʻi as a first-class language and honor Native Hawaiian cultural protocols
- Open accountability — no proprietary black-box algorithms may be used in any decision-making process affecting community members

GWT Compliance
Vendors should note that pursuant to the Generational Wealth Trust provision, 12.5% of the total contract value will be directed to the GWT Fund upon contract execution. This amount is factored into the contract budget. Breakdown: 10% Community Fund (80% of GWT) for direct resident benefit programs; 2% Trust Management (16% of GWT) for administrative oversight; 0.5% Langford Consulting Fiduciary (4% of GWT) for program governance and accountability.

7.2 Subrecipient and Partner Integration

The selected vendor will be required to integrate with and support the following existing County partners and sub-recipient organizations:

Hawaiian Community Assets (HCA)	Financial empowerment, homebuyer counseling, and affordable housing technical assistance
FarmBox Foods Fresh	Food production technology integration — mushroom and hydroponic greens supply chain
Where Food Comes From (WFCF)	Third-party agricultural traceability and food safety certification
Langford Consulting Trust	Program governance, RAIS administration, and GWT fiduciary oversight
12 Stones Global, Inc.	Platform architecture, Sovereign Charter compliance, and community node management

7.3 Environmental and Historic Preservation

All platform deployment activities (including IoT hardware installation, network infrastructure, and physical data center equipment) must comply with:

- National Environmental Policy Act (NEPA) — Environmental Review clearance required prior to any physical installation
- National Historic Preservation Act (NHPA) Section 106 — Consultation required for any activities affecting historic properties in Lahaina or surrounding cultural landscapes
- Hawaii Historic Preservation Division (SHPD) consultation protocols

7.4 Procurement Integrity

This procurement is subject to all applicable County of Maui procurement ordinances, Chapter 103D Hawaii Revised Statutes (Hawaii Public Procurement Code), and federal procurement integrity requirements. Any vendor that has a conflict of interest, has participated in the preparation of this RFP, or has provided consulting services to the County related to this procurement within the past 24 months is ineligible to respond.

PROPOSAL CERTIFICATION AND SIGNATURE

By submitting a proposal in response to this Request for Proposals (RFP No. CORR-2026-001), the undersigned authorized representative of the Proposer certifies that:

11. The information provided in this proposal is accurate, complete, and not misleading;
12. The Proposer is not currently debarred, suspended, or otherwise excluded from participation in federal assistance programs;
13. The Proposer has not engaged in any collusion, bid rigging, or other anticompetitive conduct in connection with this procurement;
14. The Proposer acknowledges and agrees to comply with all terms, conditions, and requirements set forth in this RFP;
15. The Proposer understands that submission of a proposal does not guarantee contract award;
16. The Proposer acknowledges receipt of all addenda issued in connection with this RFP (list addenda received, if any);
17. The Proposer certifies compliance with all applicable federal, state, and local laws and regulations.

Legal Name of Proposer: **Printed Name and Title:**

Authorized Signature: **Date:**

SUBMIT PROPOSALS TO:
County of Maui, Procurement Division
Attn: RFP No. CORR-2026-001
recovery-rfp@mauicounty.gov | (808) 270-7855
Deadline: May 30, 2026 at 4:00 PM HST — NO EXCEPTIONS

12 STONES GLOBAL, INC.

in Partnership with Maui County Office of Recovery & Resilience

MEMORANDUM OF UNDERSTANDING

MOU-12SG-MAUI-2026-001

MauiOS Integrated Technology & Accountability Platform

CDBG-DR Recovery Infrastructure | HFEv4 Digital Twin Deployment | Agricultural Resilience Systems

Effective Date: _____

Term: 24 Months with Renewal Options

PARTY 1	PARTY 2	PARTY 3	PARTY 4	PARTY 5
Langford Consulting Trust / 12 Stones Global	Maui County Office of Recovery & Resilience	FarmBox Foods / FarmBox Foods Fresh	Hawaiian Community Assets (HCA)	Where Food Comes From (WFCF)

CONFIDENTIAL — FOR EXECUTION AND OFFICIAL USE ONLY | This Memorandum of Understanding is a binding legal instrument. All parties are advised to review with qualified legal counsel prior to execution.

RECITALS

WHEREAS, the County of Maui ("County") has been designated as a Community Development Block Grant — Disaster Recovery (CDBG-DR) grantee by the U.S. Department of Housing and Urban Development (HUD), receiving $1.639 billion in disaster recovery assistance following the August 8, 2023 Lahaina wildfire, the deadliest U.S. wildfire in over a century;

WHEREAS, 12 Stones Global, Inc. and the Langford Consulting Trust (collectively "12 Stones") have designed and developed the MauiOS Integrated Technology & Accountability Platform — a six-module, blockchain-secured, community-sovereign digital infrastructure system — to serve as the technological backbone of Maui County's CDBG-DR recovery administration, wildfire early warning, food security, and community equity programs;

WHEREAS, the 12 Stones Sovereign Charter (SSC v5), a 28-Article governance document duly filed with the Maui County Council on April 14, 2025, establishes the principles of radical transparency, community sovereignty, and generational equity that govern all programs and expenditures covered by this MOU;

WHEREAS, FarmBox Foods, Inc. and its agricultural production subsidiary FarmBox Foods Fresh (Sedalia, CO, established August 2025) are recognized technology-integrated agricultural producers whose food production systems are integral to the food security and agricultural resilience components of the County's Recovery Action Plan;

WHEREAS, Hawaiian Community Assets (HCA) is a HUD-approved housing counseling agency and community development financial institution (CDFI) serving low- and moderate-income Maui residents, with established capacity to administer housing, financial literacy, and community benefit programs under CDBG-DR;

WHEREAS, Where Food Comes From, Inc. (WFCF) is a nationally recognized third-party agricultural verification and traceability organization whose verification protocols and supply chain documentation systems are integral to ensuring food safety, market access, and USDA compliance for agricultural programs funded under this MOU;

WHEREAS, all Parties share a commitment to transparent, accountable, community-centered recovery that honors the cultural heritage of Maui, advances the economic self-determination of Native Hawaiian and other historically marginalized communities, and builds long-term resilience against future disaster events;

NOW, THEREFORE, in consideration of the mutual covenants, commitments, and obligations set forth herein, and for other good and valuable consideration, the receipt and sufficiency of which are hereby acknowledged, the Parties agree as follows:

ARTICLE 1 — PURPOSE AND OBJECTIVES

1.1 Purpose

This Memorandum of Understanding ("MOU") establishes the collaborative framework, mutual obligations, financial arrangements, governance structure, intellectual property rights, and performance expectations governing the joint development, deployment, and operation of the MauiOS Integrated Technology & Accountability Platform by and among the Parties identified herein.

1.2 Primary Objectives

The Parties enter this MOU with the following primary objectives:

1. Deploy MauiOS — a six-module, FedRAMP-aligned, blockchain-secured community technology platform — as the operational backbone of Maui County's CDBG-DR recovery administration, enabling transparent, auditable management of up to $1.639 billion in federal disaster recovery funds;
2. Establish and operate the HFEv4 Digital Twin of Maui County, a 52-node + 2 Joker simulation environment modeling community resilience across three zones: Mauka (governance), Farmlands (agriculture), and Makai (ocean/coastal economy);
3. Deploy the Real-Time Accountability & Intelligence System (RAIS) — 188 immudb blockchain-sealed compliance scrolls — ensuring all CDBG-DR expenditures are traceable, publicly reportable, and protected from fraud, waste, and abuse;
4. Build and operate the Kipuka community engagement portal supporting six languages (English, Olelo Hawaii, Tagalog, Spanish, Japanese, Korean) with FEMA IPAWS alert integration for wildfire early warning;
5. Integrate FarmBox Foods Fresh agricultural production systems into a USDA-compliant food security supply chain serving displaced and food-insecure Maui households, with WFCF third-party verification;
6. Administer the Generational Wealth Trust (GWT) in compliance with SSC v5 Article 7, directing 12.5% of all public contract value to community benefit, trust management, and program governance.

1.3 Guiding Principles

All activities undertaken pursuant to this MOU shall be guided by the following principles drawn from the 12 Stones Sovereign Charter (SSC v5):

- Radical Transparency: All expenditures, decisions, and program outcomes shall be publicly accessible through the RAIS system and the MauiOS community portal
- Community Sovereignty: Technology systems shall empower, not extract from, Maui communities; no proprietary algorithms may govern decisions affecting residents
- Generational Equity: Present investments shall create lasting economic benefit for future Maui generations through the GWT framework

- Cultural Respect: All systems shall treat Olelo Hawaii as a first-class language and honor Native Hawaiian cultural protocols in design and implementation
- Open Accountability: All partner organizations are subject to audit, performance review, and community accountability under RAIS

ARTICLE 2 — PARTIES AND AUTHORIZED REPRESENTATIVES

2.1 Party Identification

Party	Legal Entity	Primary Role	Address	State
1	Langford Consulting Trust & **12 Stones Global, Inc.**	Lead Technology Partner, Charter Authority, RAIS Administrator, GWT Fiduciary	Maui, Hawaii	Hawaii
2	County of Maui, Office of Recovery & Resilience	CDBG-DR Grantee, County Partner, Platform Commissioner	2200 Main St, Suite 300, Wailuku, HI 96793	Hawaii
3	FarmBox Foods, Inc. / **FarmBox Foods Fresh**	Agricultural Technology Partner, Food Security Producer, VAPG Applicant	Sedalia, CO 80135	Colorado
4	**Hawaiian Community Assets (HCA)**	Housing Counseling, Financial Empowerment, CDFI Services, Section 3 Compliance	Maui, Hawaii	Hawaii
5	**Where Food Comes From, Inc. (WFCF)**	Agricultural Traceability Verification, Food Safety Certification, USDA Compliance	Castle Rock, CO 80109	Colorado

2.2 Authorized Representatives

Each Party designates the following individual(s) as their Authorized Representative(s) for purposes of this MOU. All formal communications, amendments, and approvals must be executed or acknowledged by the Authorized Representative or their designated successor:

Party	Name	Title	Email	Phone
1	James Langford	Chief Executive Officer / Fiduciary Trustee	jimlangford@me.com	To be confirmed
2	[County Rep Name]	Director, Office of Recovery & Resilience	[County Email]	[County Phone]

3	Rusty Walker	CEO, FarmBox Foods	[FarmBox Email]	[FarmBox Phone]
4	[HCA Rep Name]	Executive Director, HCA	[HCA Email]	[HCA Phone]
5	[WFCF Rep Name]	CEO / President, WFCF	[WFCF Email]	[WFCF Phone]

2.3 Notice of Changes

Any Party may change its Authorized Representative by providing written notice to all other Parties at least ten (10) business days prior to the change taking effect. Such notice shall be provided via email to all Authorized Representatives and shall not require a formal amendment to this MOU.

ARTICLE 3 — TERM, RENEWAL, AND TERMINATION

3.1 Initial Term

This MOU shall become effective upon the date of execution by all five Parties (the "Effective Date") and shall remain in full force and effect for an initial term of twenty-four (24) months, corresponding to the five-phase project scope set forth in Article 5 (the "Initial Term"). The Initial Term may not be shortened except by mutual written agreement of all Parties or as provided in Section 3.3 (Termination for Default).

3.2 Renewal Options

Upon expiration of the Initial Term, this MOU shall automatically renew for successive one-year periods (each, a "Renewal Term") unless any Party provides written notice of non-renewal to all other Parties no later than ninety (90) calendar days prior to the expiration of the then-current term. Renewal shall be subject to:

- Continued availability of CDBG-DR or other federal funding supporting the programs described herein
- Satisfactory performance review by the 12 Stones RAIS system and the County's Office of Recovery & Resilience
- Annual renegotiation of the SaaS licensing fee (Section 6.3) based on CPI-U adjustment not to exceed 3% per year
- Mutual agreement on any additional modules, program expansions, or scope changes

3.3 Termination for Convenience

Any Party may terminate its participation in this MOU for any reason upon one hundred eighty (180) calendar days' written notice to all other Parties. Termination for convenience shall not relieve the terminating Party of obligations arising prior to the notice date, including payment obligations and data return requirements. The non-terminating Parties may elect to continue operations under the remaining parties' agreement.

3.4 Termination for Default

Any Party may terminate this MOU with respect to a defaulting Party upon thirty (30) calendar days' written notice if the defaulting Party:

7. Materially breaches any obligation under this MOU and fails to cure such breach within thirty (30) days of written notice;
8. Becomes insolvent, makes a general assignment for the benefit of creditors, or commences or becomes subject to bankruptcy proceedings;
9. Is debarred, suspended, or otherwise excluded from participation in federal assistance programs;
10. Fails to maintain required insurance coverage for more than fifteen (15) days;

11. Violates applicable federal, state, or local laws in connection with activities undertaken pursuant to this MOU.

3.5 Effects of Termination

Upon termination or expiration of this MOU:

- All custom-developed code, data, and work product belonging to the County shall be transferred to the County within thirty (30) days
- Each Party shall return or certifiably destroy all confidential information of other Parties within thirty (30) days
- All accrued and unpaid obligations shall survive termination
- Sections 7 (Intellectual Property), 10 (GWT), 11 (Confidentiality), 12 (Dispute Resolution), and 14 (Governing Law) shall survive termination
- The RAIS blockchain scrolls generated during the term shall remain accessible to the County and to HUD for the federally required 7-year records retention period

ARTICLE 4 — PARTY OBLIGATIONS

4.1 Obligations of Party 1 — Langford Consulting Trust / 12 Stones Global, Inc.

12 Stones shall serve as the Lead Technology Partner and Program Architect for all activities under this MOU. Specific obligations include:

Platform Development and Operations

- Design, develop, deploy, and maintain all six MauiOS modules as specified in Article 5
- Maintain the HFEv4 52-node digital twin and all associated simulation and modeling infrastructure
- Operate and administer the RAIS system, including all 188+ blockchain-sealed compliance scrolls
- Provide 99.5% platform uptime with documented disaster recovery and business continuity plans
- Conduct annual third-party security audits and penetration testing, with reports provided to the County

Governance and Compliance

- Administer the 12 Stones Sovereign Charter (SSC v5) and maintain the digital Charter repository
- Serve as GWT Fiduciary Trustee, directing 12.5% of applicable public contract value per Section 10
- Prepare and submit all required HUD CDBG-DR performance reports in coordination with the County
- Ensure all platform activities comply with CDBG-DR National Objectives and 2 CFR Part 200
- Maintain all federal certifications and registrations (SAM.gov, DUNS/UEI, etc.) throughout the term

Staffing

- Designate a dedicated Project Manager with CDBG-DR experience as the primary County interface
- Staff the project with qualified personnel meeting the qualifications described in Article 5
- Provide Hawaii-based staffing to the maximum extent practicable, prioritizing Section 3 workers

4.2 Obligations of Party 2 — Maui County Office of Recovery & Resilience

The County shall serve as the CDBG-DR Grantee and Platform Commissioner. Specific obligations include:

- Execute a formal Subrecipient Agreement designating 12 Stones as a CDBG-DR subrecipient under 24 CFR 570.500 within sixty (60) days of this MOU's Effective Date
- Provide timely access to County systems, data, personnel, and facilities required for platform development and integration
- Make required CDBG-DR draws from HUD IDIS and transfer funds to 12 Stones within thirty (30) days of approved deliverable completion
- Assign a dedicated County Liaison with authority to review and approve deliverables, resolve issues, and execute amendments
- Conduct required Environmental Reviews (NEPA) and Historic Preservation consultations (NHPA Section 106) prior to any physical installation activities
- Issue or assist in issuing all required procurement notices, including the RFP (CORR-2026-001) for the MauiOS platform contract
- Provide the County Match commitment of $1,200,000 (20% of the CDBG-DR platform budget) from non-federal sources

4.3 Obligations of Party 3 — FarmBox Foods / FarmBox Foods Fresh

FarmBox Foods and its production subsidiary shall serve as the Agricultural Technology Partner and Food Security Producer. Specific obligations include:

- Operate a minimum of six (6) FarmBox Foods Fresh food production units on Maui within twelve (12) months of the Effective Date
- Integrate all production systems with the MauiOS Agricultural Traceability Platform (Module 5) within Phase 3
- Maintain all USDA Value-Added Producer Grant (VAPG) compliance requirements for both Hawaii and Colorado applications
- Participate in the WFCF third-party verification program for all produce distributed through CDBG-DR food security channels
- Provide monthly production data, yield reports, and distribution records to the RAIS system
- Meet Section 3 employment targets by prioritizing hiring of Maui residents displaced by the Lahaina wildfire
- Contribute required VAPG match funding through verified in-kind labor and operating expenses as documented in VAPG applications

4.4 Obligations of Party 4 — Hawaiian Community Assets (HCA)

HCA shall serve as the Housing Counseling, Financial Empowerment, and Community Equity partner. Specific obligations include:

- Provide HUD-approved housing counseling services to a minimum of 500 displaced Maui households annually
- Administer the GWT Community Fund distribution program, directing 10% of GWT proceeds to qualified community benefit programs

- Operate financial literacy and wealth-building programs targeting Native Hawaiian, Pacific Islander, and low-income Maui residents
- Ensure all HCA programs are documented in the MauiOS community portal (Module 4 / Kipuka) in all six supported languages
- Serve as the Section 3 compliance monitor and maintain records of Section 3 employment and contracting activities
- Provide quarterly program reports to 12 Stones for integration into RAIS compliance scrolls

4.5 Obligations of Party 5 — Where Food Comes From (WFCF)

WFCF shall serve as the Agricultural Traceability Verification and Food Safety Certification partner. Specific obligations include:

- Provide third-party verification and certification for all FarmBox Foods Fresh produce distributed through CDBG-DR food security programs
- Integrate WFCF verification protocols and data feeds with the MauiOS Agricultural Traceability Platform (Module 5)
- Issue verification certificates meeting USDA Agricultural Marketing Service (AMS) reporting requirements
- Maintain accreditation and certification in good standing throughout the term
- Provide QR/RFID traceability data supporting farm-to-table supply chain documentation from production through community distribution
- Submit verification reports to the RAIS system within five (5) business days of each harvest cycle

ARTICLE 5 — PROJECT SCOPE AND FIVE-PHASE IMPLEMENTATION

5.1 MauiOS Six-Module Platform

The core deliverable of this MOU is the MauiOS Integrated Technology & Accountability Platform, comprising the following six modules to be developed and deployed across the five phases described in Section 5.2:

#	Module Name	Description and Key Functions
1	Sovereign Charter Management System (SCMS)	Digital repository for SSC v5; compliance alert engine; County legal integration; public transparency portal
2	Real-Time Accountability & Intelligence System (RAIS)	188+ immudb blockchain-sealed scrolls; HUD IDIS integration; expenditure anomaly detection; DRGR automated reporting
3	HFEv4 Digital Twin Visual Interface	52-node + 2 Joker digital twin; 3-zone GIS dashboard; Node 53 wildfire modeling (FARSITE); Node 54 sovereign audit layer
4	Community Engagement Portal (Kipuka)	6-language community portal; benefit enrollment integration; FEMA IPAWS alert system; anonymous fraud reporting channel
5	Agricultural Traceability & Food Security Platform	QR/RFID supply chain traceability; USDA AMS reporting; FarmBox Fresh integration; WFCF verification feeds; cold chain monitoring
6	Legal Scroll Sealing & Compliance Archival System	Blockchain-sealed legal repository; 7-year federal records retention; e-signature and notarization workflows; HUD audit integration

5.2 Five-Phase Implementation Schedule

Ph	Phase Name	Timeline	Key Deliverables	Lead Party
1	Mobilization & Preliminary Design	**Month 0–1**	Project kickoff and team onboarding Preliminary architecture and data dictionary CDBG-DR environmental review initiation SSC v5 digital Charter repository setup	12 Stones (Lead)
2	System Design & Integration Planning	**Month 1–3**	Technical design specifications (all 6 modules)	12 Stones + County

			API contracts for HUD IDIS, FEMA, USDA AMS Security plan — NIST 800-53 Rev. 5 mapping GWT Trust structure legal formalization	
3	Platform Build & Deployment	**Month 3–18**	All 6 modules staged, tested, and deployed RAIS 188-scroll blockchain activation FarmBox Fresh IoT integration and commissioning Kipuka portal multilingual launch (6 languages) HFEv4 digital twin 52-node live activation	All Parties
4	Quality Assurance & Training	**Month 18–21**	Third-party security audit and penetration test HUD CDBG-DR compliance review Staff and community training delivery Section 3 employment audit and certification	12 Stones + County
5	Full Operations & Handover	**Month 21–24**	Production go-live and operational handover 90-day hypercare support period Transition to annual SaaS subscription model Final program performance report to HUD	12 Stones (Lead)

5.3 Performance Standards

All deliverables shall be measured against the following minimum performance standards:

- Platform Uptime: 99.5% availability measured monthly, excluding scheduled maintenance windows
- RAIS Completeness: 100% of CDBG-DR transactions logged within 24 hours of execution
- Reporting Timeliness: All HUD quarterly reports submitted no later than 15 days after each quarter close
- Food Security: FarmBox Foods Fresh production systems operational at all six units within 12 months
- Community Engagement: Minimum 1,000 unique registered users on Kipuka portal within 6 months of launch
- Section 3: Minimum 25% of new labor hours performed by Section 3 workers per HUD requirements

ARTICLE 6 — FINANCIAL ARRANGEMENTS AND BUDGET

6.1 Total Program Budget

The estimated total investment under this MOU is Ten Million Dollars ($10,000,000) over the 24-month Initial Term, composed of the following funding streams:

Program Area	Funding Source	Amount	% of Total
MauiOS Platform (all 6 modules)	HUD CDBG-DR	**$6,000,000**	60.0%
County Match (Technology)	County General Fund	**$1,200,000**	12.0%
FarmBox Foods Fresh — 6 Production Units	USDA VAPG + FarmBox Match	**$700,400**	7.0%
HCA Housing & Financial Programs	CDBG-DR / CDFI Funds	**$1,000,000**	10.0%
WFCF Verification & Traceability	CDBG-DR Subcontract	**$300,000**	3.0%
GWT Administration & Community Fund	Public Contract Set-Aside	**$800,000**	8.0%
TOTAL	Multiple Sources	**$10,000,000**	**100%**

6.2 CDBG-DR Budget Detail

The $6,000,000 CDBG-DR platform budget is allocated across the five project phases as follows:

Expenditure Category	Phase 1–2	Phase 3	Phase 4–5	Total
Personnel — 12 Stones staff (PM, Dev, Security, Compliance)	$120,000	$850,000	$230,000	**$1,200,000**
Platform Development (6 Modules)	$180,000	$1,620,000	$200,000	**$2,000,000**
Infrastructure (cloud hosting, security, blockchain nodes)	$80,000	$420,000	$100,000	**$600,000**
HFEv4 Digital Twin Build & Licensing	$100,000	$700,000	$200,000	**$1,000,000**
Training, Community Engagement & Section 3	$20,000	$160,000	$120,000	**$300,000**
Audit, Compliance & Legal (NEPA, Section 106)	$50,000	$200,000	$50,000	**$300,000**
Program Administration (2 CFR §200 limit)	$25,000	$375,000	$200,000	**$600,000**
TOTAL CDBG-DR	$575,000	$4,325,000	$1,100,000	**$6,000,000**

6.3 SaaS Licensing — Post-Deployment

Following completion of the Initial Term and platform handover (Phase 5), the County shall be entitled to continue platform operations under an annual Software-as-a-Service license agreement at the following rates:

- Base Platform License (all 6 modules): $120,000 per year (fixed for years 3–5)
- Additional Modules or User Tiers: Priced per separate order form, subject to mutual written agreement
- Annual CPI-U adjustment cap: 3% beginning in Year 6
- Government perpetual license option available at 5x Annual License fee after Year 5

6.4 Payment Schedule and Invoicing

12 Stones shall invoice the County on a milestone-based schedule tied to the Phase deliverables in Section 5.2. The County shall remit payment within thirty (30) calendar days of written deliverable acceptance. All invoices must include:

- RAIS scroll reference numbers documenting completed work
- HUD CDBG-DR draw request number and approved budget line item
- GWT calculation and deposit confirmation (12.5% of invoice amount)
- Section 3 labor hours for the billing period

ARTICLE 7 — INTELLECTUAL PROPERTY

7.1 County-Commissioned Work Product

All custom software code, data models, database schemas, configuration files, training materials, documentation, and other work product developed specifically and exclusively for the County under this MOU (collectively, "County Work Product") shall be deemed works made for hire and shall be the exclusive property of the County of Maui upon final acceptance and full payment. 12 Stones hereby assigns all right, title, and interest in County Work Product to the County, and shall execute any documents necessary to perfect such assignment.

7.2 12 Stones Pre-Existing IP

12 Stones retains all right, title, and interest in and to the following pre-existing intellectual property ("12 Stones IP"):

- The HFEv4 (Hyper Farming Engine v4) digital twin architecture, simulation engine, and node modeling framework
- The RAIS (Real-Time Accountability & Intelligence System) core engine, immudb integration framework, and scroll-sealing methodology
- The 12 Stones Sovereign Charter (SSC v5) governance framework and all associated legal and governance instruments
- The Generational Wealth Trust (GWT) financial architecture and fiduciary protocols
- All MauiOS platform code, modules, APIs, and infrastructure components that existed prior to or are developed independently of this MOU

The County is hereby granted a perpetual, irrevocable, non-exclusive, royalty-bearing license to use 12 Stones IP solely for the purposes described in this MOU, subject to the payment of the SaaS licensing fees described in Section 6.3. This license terminates upon material breach by the County not cured within the required period.

7.3 Partner IP

FarmBox Foods retains all rights to its proprietary container farming systems, grow protocols, and production technology. WFCF retains all rights to its verification methodology and certification intellectual property. HCA retains all rights to its counseling frameworks and program methodologies. Each Party grants the other Parties a limited license to use their respective IP solely as necessary to fulfill the obligations described in this MOU.

7.4 County Data

All data generated by Maui County residents, programs, and public systems — including resident PII, program application data, CDBG-DR expenditure records, and community feedback — shall be and remain the exclusive property of the County of Maui. 12 Stones and all other Parties shall not

use, transfer, sell, or analyze County Data for any purpose other than fulfilling their obligations under this MOU without prior written consent of the County's Corporation Counsel.

7.5 Open Source Components

Where MauiOS incorporates open-source software components (including immudb, PostgreSQL, OpenStreetMap, and other community-maintained libraries), such components are governed by their respective open-source licenses. 12 Stones shall maintain a current Bill of Materials listing all open-source components and their licenses, updated quarterly, and shall promptly notify the County of any license changes affecting County Work Product or County Data.

ARTICLE 8 — INSURANCE AND INDEMNIFICATION

8.1 Insurance Requirements

Each Party receiving funds or performing services under this MOU shall maintain, at its own expense, the following minimum insurance coverage throughout the term:

Coverage Type	Per Occurrence	Aggregate	Applies To
Commercial General Liability	**$2,000,000**	**$4,000,000**	All Parties
Technology E&O / Cyber Liability	**$2,000,000**	**$4,000,000**	12 Stones (required)
Workers Compensation	Hawaii Statutory	N/A	All Parties with employees
Professional Liability (E&O)	**$1,000,000**	**$2,000,000**	12 Stones, HCA, WFCF
Commercial Auto	$1,000,000	Combined	FarmBox (delivery vehicles)

The County of Maui and 12 Stones Global, Inc. shall be named as Additional Insureds on all applicable liability policies maintained by the other Parties. Each Party shall provide certificates of insurance to all other Parties within ten (10) business days of the Effective Date and annually thereafter.

8.2 Indemnification

Each Party ("Indemnifying Party") shall indemnify, defend, and hold harmless all other Parties, their officers, directors, employees, agents, and successors ("Indemnified Parties") from and against any and all claims, damages, losses, costs, and expenses (including reasonable attorneys' fees) arising out of or resulting from the Indemnifying Party's:

12. Negligent or wrongful acts or omissions in connection with activities under this MOU;
13. Material breach of any representation, warranty, or obligation under this MOU;
14. Violation of applicable law, regulation, or federal grant requirement;
15. Infringement of any third-party intellectual property rights.

This indemnification obligation shall survive termination or expiration of this MOU for a period of three (3) years.

ARTICLE 9 — CONFIDENTIALITY AND DATA PRIVACY

9.1 Confidential Information

Each Party ("Receiving Party") agrees to hold in strict confidence all Confidential Information received from any other Party ("Disclosing Party") and not to disclose such information to any third

party without prior written consent of the Disclosing Party, except as required by law, court order, or federal audit requirements. "Confidential Information" means any non-public technical, financial, operational, or programmatic information disclosed in connection with this MOU, including but not limited to platform architecture, security configurations, resident PII, proprietary algorithms, and trade secrets.

9.2 Resident Data Protection

All Parties handling Maui resident personal data shall comply with:

- Hawaii Revised Statutes Chapter 487N (Security Breach of Personal Information)
- Federal Privacy Act requirements applicable to HUD-funded programs
- CDBG-DR data privacy requirements as specified in HUD's Privacy Impact Assessment for IDIS
- HIPAA requirements for any health-related data collected through Kipuka or food security programs

Any suspected or confirmed data breach shall be reported to all Parties' Authorized Representatives and to the County's Corporation Counsel within twenty-four (24) hours of discovery.

9.3 Public Records

The Parties acknowledge that this MOU, once executed, may be subject to Hawaii's Uniform Information Practices Act (UIPA, HRS Chapter 92F) and federal FOIA requirements applicable to CDBG-DR funded programs. Neither Party shall attempt to shield records from legitimate public access. The RAIS blockchain-sealed expenditure scrolls shall be publicly accessible by design in compliance with the 12 Stones Sovereign Charter transparency principles.

ARTICLE 10 — GENERATIONAL WEALTH TRUST (GWT)

10.1 GWT Structure and Purpose

Pursuant to Article 7 of the 12 Stones Sovereign Charter (SSC v5), the Generational Wealth Trust (GWT) is established as a community benefit mechanism directing a portion of all public contract value to long-term community economic development, affordable housing, food security, and cultural preservation programs benefiting current and future Maui residents.

10.2 GWT Allocation Formula

For all public contracts executed under or in connection with this MOU, the following allocation applies to the GWT Set-Aside (12.5% of total contract value):

GWT Allocation Breakdown

10.0% — Community Fund (80% of GWT): Direct benefit programs for Maui households — housing, food, education, cultural preservation

2.0% — Trust Management (16% of GWT): Administrative oversight, fiduciary compliance, and GWT operating costs

0.5% — Langford Consulting Fiduciary (4% of GWT): Program governance, RAIS accountability administration, and Charter enforcement

Total Set-Aside: 12.5% of applicable public contract value, calculated at invoice and remitted concurrent with payment

10.3 GWT Administration

12 Stones Global, Inc. / Langford Consulting Trust shall serve as GWT Fiduciary Trustee. HCA shall serve as Community Fund Administrator, receiving and disbursing the 10% Community Fund allocation through Board-approved grant cycles. GWT accounts shall be maintained at a federally insured financial institution approved by the County and shall be subject to annual independent audit.

10.4 Community Fund Eligible Uses

The 10% Community Fund may be used for the following purposes, subject to HCA Board approval:

- Down payment assistance and homebuyer programs for displaced Lahaina residents
- Small business recovery grants for Maui entrepreneurs affected by the wildfire
- Cultural preservation and Native Hawaiian language education programs
- Food security and nutrition programs serving low-income households
- Youth workforce development and STEM education initiatives
- Affordable rental housing development and preservation

10.5 GWT Reporting

12 Stones shall provide quarterly GWT reports to all Parties and shall post GWT deposit records and fund disbursements to the RAIS system within five (5) business days of each transaction. Annual GWT audits shall be posted publicly to the MauiOS community portal in compliance with SSC v5 transparency requirements.

ARTICLE 11 — DISPUTE RESOLUTION

11.1 Good Faith Negotiation

The Parties agree to attempt in good faith to resolve any dispute, controversy, or claim arising out of or relating to this MOU or its breach, termination, or validity (a "Dispute") through direct negotiation between the Authorized Representatives before pursuing any other remedy. Any Party may initiate dispute resolution by delivering written notice to all other Authorized Representatives describing the Dispute in reasonable detail. The Parties shall negotiate in good faith for a period of thirty (30) calendar days following delivery of such notice (the "Negotiation Period").

11.2 Mediation

If the Parties are unable to resolve a Dispute through good faith negotiation within the Negotiation Period, any Party may request mediation by delivering written notice to all other Parties. The Parties shall jointly select a neutral mediator with experience in government technology contracts and federal grant programs. If the Parties cannot agree on a mediator within fifteen (15) days, either party may request appointment of a mediator by the Hawaii State Judiciary's Center for Alternative Dispute Resolution. Mediation costs shall be shared equally among the disputing parties.

11.3 Arbitration

If mediation fails to resolve the Dispute within sixty (60) days of the mediator's appointment, any Party may initiate binding arbitration under the Commercial Arbitration Rules of the American Arbitration Association (AAA), with proceedings held in Honolulu, Hawaii. The arbitration panel shall consist of one (1) neutral arbitrator with demonstrated expertise in government technology contracts. The arbitrator's award shall be final and binding and may be entered as a judgment in any court of competent jurisdiction.

11.4 Continuation of Performance

Pending resolution of any Dispute, all Parties shall continue performing their obligations under this MOU in good faith unless the disputed obligation is the subject of an emergency injunction or restraining order issued by a court of competent jurisdiction.

11.5 Federal Dispute Rights

Nothing in this Article shall limit the County's rights under applicable federal law or HUD regulations to pursue remedies for subrecipient non-compliance, including suspension, debarment, or repayment demands, nor shall it limit HUD's rights as the federal grantor.

ARTICLE 12 — GENERAL PROVISIONS

12.1 Governing Law

This MOU shall be governed by and construed in accordance with the laws of the State of Hawaii, without regard to conflict of law principles, except to the extent superseded by applicable federal law. The Parties consent to the exclusive jurisdiction of the state and federal courts located in Maui County, Hawaii for any action arising under this MOU not subject to arbitration under Article 11.

12.2 Entire Agreement

This MOU, together with all Exhibits and Attachments incorporated herein by reference, constitutes the entire agreement of the Parties with respect to the subject matter hereof and supersedes all prior and contemporaneous negotiations, representations, warranties, and agreements, whether written or oral. No Party has relied on any representation not expressly contained in this MOU.

12.3 Amendments

This MOU may be amended only by a written instrument signed by the Authorized Representatives of all five Parties. No oral modification or waiver shall be effective. Minor scope adjustments within approved budget line items may be authorized by the 12 Stones Project Manager and the County Liaison without formal amendment, provided such adjustments do not increase total cost or extend the term.

12.4 Severability

If any provision of this MOU is held invalid, illegal, or unenforceable by a court or arbitrator of competent jurisdiction, the remaining provisions shall continue in full force and effect. The Parties shall negotiate in good faith to replace any invalid provision with a valid provision that most closely approximates the intent of the original.

12.5 Waiver

No failure or delay by any Party in exercising any right or remedy under this MOU shall constitute a waiver of that right or remedy. No single or partial exercise of any right or remedy shall preclude any other or further exercise of that right or remedy.

12.6 Counterparts

This MOU may be executed in counterparts, each of which shall be deemed an original, and all of which together shall constitute one and the same instrument. Electronic signatures (including DocuSign, Adobe Sign, and equivalent platforms) shall be deemed valid and binding for all purposes.

12.7 No Third-Party Beneficiaries

This MOU is for the sole and exclusive benefit of the five named Parties and their permitted successors and assigns. Nothing herein shall create or be deemed to create any legal or equitable rights in any other person or entity, except that HUD and other federal agencies shall retain all rights as grantors under applicable federal law.

12.8 Force Majeure

No Party shall be in default of this MOU if performance is delayed or prevented by circumstances beyond that Party's reasonable control, including acts of God, natural disasters, government actions, labor disputes not involving the Party's own employees, or pandemics declared by federal or state authorities. The affected Party shall provide written notice to all other Parties within seventy-two (72) hours of the force majeure event and shall resume performance as soon as reasonably practicable.

12.9 Federal Requirements

All activities funded in whole or in part by CDBG-DR funds shall comply with applicable federal requirements including: the Housing and Community Development Act of 1974 (as amended); 24 CFR Part 570; 2 CFR Part 200 (Uniform Administrative Requirements); the Fair Housing Act; the Civil Rights Act of 1964; the Age Discrimination Act of 1975; Section 504 of the Rehabilitation Act; the Americans with Disabilities Act; the National Environmental Policy Act; and all other applicable federal statutes, regulations, and executive orders.

EXHIBITS AND ATTACHMENTS

The following Exhibits are incorporated into and made a part of this MOU:

Exhibit	Title	Status
A	12 Stones Sovereign Charter (SSC v5) — 28 Articles	Filed Maui County Council, April 14, 2025
B	MauiOS Platform Technical Specifications — All 6 Modules	To be attached at Phase 2 completion
C	CDBG-DR Budget Detail and Line-Item Justification	Attached — Subject to HUD Action Plan approval
D	GWT Trust Agreement and Distribution Protocols	To be executed concurrent with MOU
E	HFEv4 Node Map — 52 Nodes, Zones 1-3, Joker Nodes 53-54	Attached
F	FarmBox Foods Fresh — VAPG Application Summary (Colorado + Hawaii)	Attached
G	WFCF Verification Protocol and Certification Standards	To be provided by WFCF within 30 days of execution
H	HCA Program Description and Section 3 Compliance Plan	To be provided by HCA within 30 days of execution
I	Insurance Certificates — All Parties	Due within 10 business days of execution
J	Five-Phase Implementation Schedule — Detailed Gantt Chart	To be finalized at Phase 1 kickoff

SIGNATURE PAGE

IN WITNESS WHEREOF, the Parties have executed this Memorandum of Understanding as of the Effective Date first written above.

PARTY 1

Langford Consulting Trust / 12 Stones Global, Inc.

_____

Signature

James Langford
CEO & Fiduciary Trustee

_____

Date

PARTY 2

County of Maui, State of Hawaii

_____

Signature

[Name, Title]
Director, Office of Recovery & Resilience

_____

Date

PARTY 3

FarmBox Foods, Inc. / FarmBox Foods Fresh

_____

Signature

Rusty Walker
Chief Executive Officer

_____

Date

PARTY 4

Hawaiian Community Assets (HCA)

_____

Signature

[Name, Title]
Executive Director

_____

Date

PARTY 5

Where Food Comes From, Inc. (WFCF)

APPROVED AS TO FORM
Office of the Corporation Counsel
County of Maui, State of Hawaii

By: _____
Date: _____

_____

Signatur
e

**[Name,
Title]**
Preside
nt / CEO

_____

Date

MOU-12SG-MAUI-2026-001 | 12 Stones Global, Inc. / Maui County Office of Recovery & Resilience | Confidential — Official Use Only

12 STONES GLOBAL

MauiOS

County Intelligence Platform

Real-time audit.
Grant accountability.
Community sovereignty.

CONFIDENTIAL DRAFT · FOR MAUI COUNTY EXECUTIVE REVIEW

Prepared by James Langford, CEO · 12 Stones Global Inc.

jimlangford@me.com · Maui, Hawaii

$1.991B

FY2026 MAUI COUNTY BUDGET UNDER MANAGEMENT

$1.639B CDBG-DR Lahaina Recovery · **88% undeployed**

Moody's Aa2 · S&P AA+ · Fitch AA+

The Problem Maui County Faces Today

A $2B budget with no real-time accountability layer

Fragmented Reporting

Budget data spans 12+ departments with no unified real-time view. Audit reports arrive 12-18 months after expenditure.

CDBG-DR Accountability Gap

$1.639B in federal recovery funds allocated. HUD requires documented oversight. 88% sits undeployed with growing compliance exposure.

No Digital Twin

Capital projects, ag programs, and recovery spending have no integrated simulation or scenario-planning environment.

Community Trust Deficit

Post-Lahaina wildfire, residents demand transparent, participatory access to how recovery dollars are spent and distributed.

Grant Stacking Complexity

$357M in FY2026 grant revenue from 20+ sources. No single system tracks compliance obligations, reporting deadlines, or co-application alignment.

Agriculture Program Fragmentation

17 active agriculture programs ($10.5M+) with no unified performance dashboard or traceability layer.

The Opportunity

Three converging forces create a historic window

$1.639B

HUD CDBG-DR

Lahaina wildfire recovery — 88% undeployed. HUD requires documented project management & accountability. MauiOS is the oversight infrastructure.

$1.991B

County FY2026 Budget

Largest Maui County budget in history. $357M in grant revenue (+64% YoY). Real-time audit SaaS at $120K/yr = 0.006% of budget.

$369M

5-Year Revenue Forecast

MauiOS as the anchor creates a replicable franchise model across 50+ municipalities nationwide. County license = the proof-of-concept.

What is MauiOS?

A sovereign operating system for county governance, accountability, and community wealth

MauiOS is a real-time audit and intelligence SaaS platform built on the 12 Stones Hyper Farming Engine v4 — a 52-node digital twin of Maui County that maps every department, program, and capital project to a live accountability layer anchored in the 12 Stones Sovereign Charter (filed Maui County Council, April 14, 2025).

01
Mākaukau Protocol

Charter logic engine + UE5 governance visualization + glyph system

02
ʻIke Kumu Stack

RAIS — 188 immutable audit scrolls, immudb blockchain, real-time compliance

03
Pā Kiʻi Framework

Visual & ceremonial UX — community-facing transparency portal

04
Kīpuka Access Portal

Mobile app + resident testimony engine + community voice recording

05
ʻŌnaehana Holo

WFCF blockchain traceability + grant stacking + ag program compliance

06
Kānaenae Compiler

Legal scroll sealing — legislation → Charter → immudb permanent record

HFEv4 — The 52-Node Digital Twin of Maui County

Every department, program, and dollar mapped to a live governance node

Zone 1 — Mauka

Nodes 1-18 · Governance & Upland

- County Council
- Managing Director
- Corporation Counsel
- Budget & Finance
- Human Resources
- Emergency Management
- Planning & GIS
- Audit & Compliance

Zone 2 — Farmlands

Nodes 19-36 · Agriculture & Culture

- Agriculture Department
- Economic Development
- Parks & Recreation
- Housing
- Environmental Management
- Hawaiian Affairs
- Arts & Culture
- Education Programs

Zone 3 — Makai

Nodes 37-52 · Ocean, Revenue & Franchise

- Water Resources
- Public Works
- Transportation
- Fire & Rescue
- Police
- Health Services
- Revenue & Tax
- Franchise & License

Node 53 — Emergency Override Protocol · Node 54 — Sovereign Full-System Audit · 2 Joker Nodes reserved for crisis governance

Why Maui County Needs MauiOS for CDBG-DR

HUD requires documented accountability on $1.639B — 88% of which is undeployed

RAIS — Real-time Accountability & Intelligence System

- 188 immutable audit scrolls sealed on immudb blockchain
- Automated flagging of expenditure deviations from budget
- Real-time compliance triggers on 4 enforcement categories: fraud, financial, federal, accountability
- Legistar scraper — council agenda auto-processing for legislative traceability
- All CDBG-DR drawdowns logged with blockchain-verified documentation

What HUD Requires for CDBG-DR Compliance

- Action Plan implementation tracking with milestone reporting
- Documented procurement and contracting oversight
- Citizen participation records and community voice evidence
- Financial management controls with audit trails
- Quarterly Performance Reports + Annual Impact Assessments

$1.639B

CDBG-DR Lahaina Recovery Total Allocation

88%

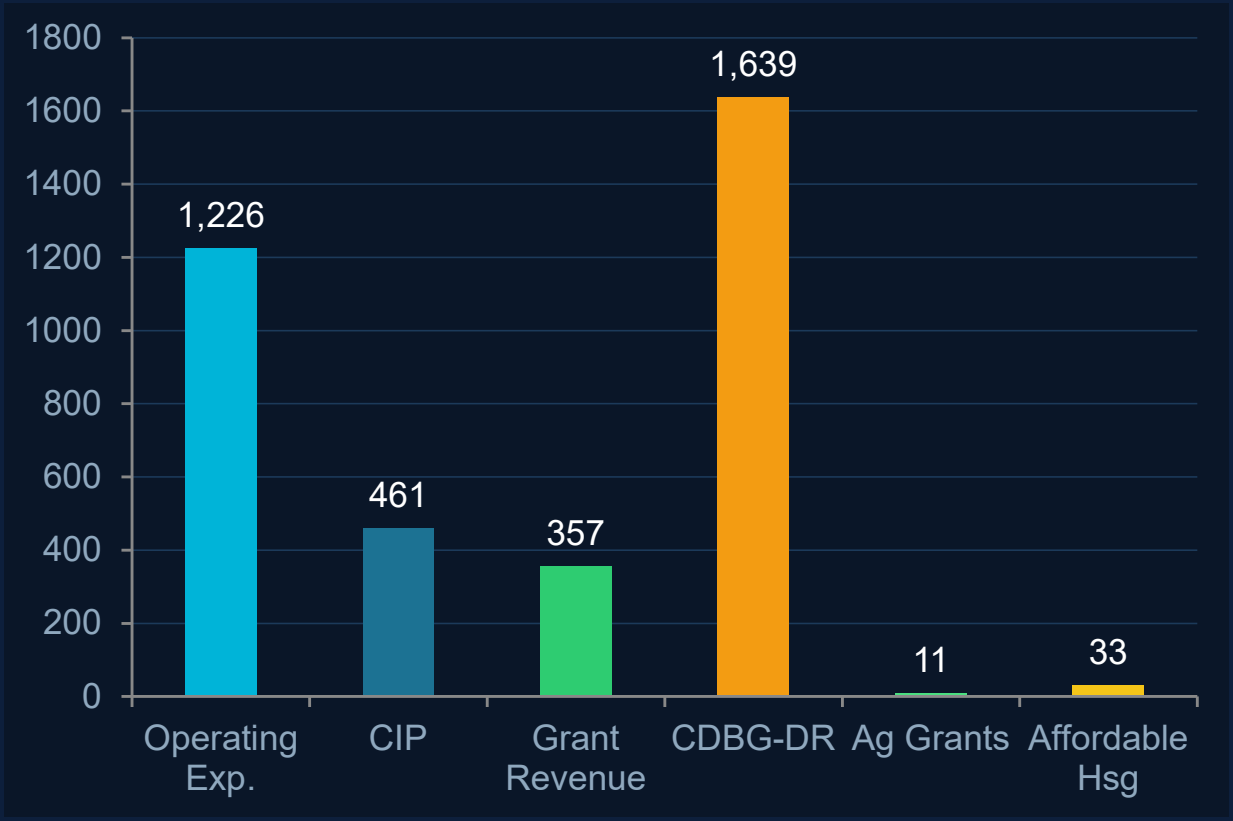
Undeployed as of Q1 2026

$120K/yr

MauiOS County License = 0.007% of CDBG-DR

FY2026 Budget Alignment

Every dollar tracked — operating, CIP, and grant revenue



$1.51 2B Operating Revenue (ordinance)

+64.4 % Grant Revenue Growth FY2025 → FY2026

12 Departments tracked across 52 HFE nodes

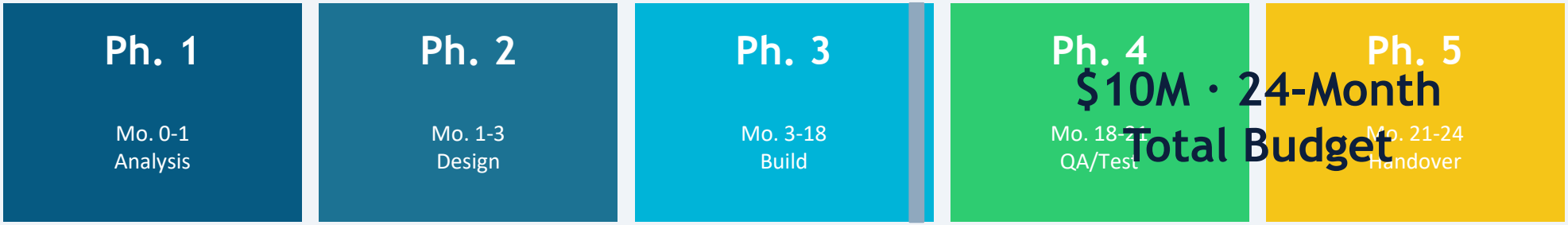
$248 M GWT potential at 12.5% of full budget deployment

Partnership Framework — The MOU

5-phase, 24-month, $10M implementation with four anchor partners

Party	Role	MOU Commitment
Langford Consulting Trust / 12 Stones Global	Primary Project Manager & Technology Provider	Software platform, digital twin, RAIS, grant management
Maui County Office of Recovery	Land Access, Permitting & Public Communications	Site access, regulatory pathway, CDBG-DR co-application
FarmBox Foods (Rusty Walker)	Hydroponic Technology & Equipment Supply	$2.025M — 52-unit deployment + operator training
Hawaiian Community Assets (HCA)	Financial Literacy & GWT Distribution	$1.35M — homeownership + community wealth pathways
Where Food Comes From (WFCF)	Blockchain Traceability & Sustainability Certification	$3K/farm/yr — ag program accountability layer

5-PHASE IMPLEMENTATION



The ROI Case for Maui County

$120K/yr license against $2B+ in managed budget and federal accountability exposure

$120K

Annual MauiOS County License

0.006%

License cost as % of $1.991B budget

$50M+

HUD audit liability exposure (est.) if CDBG-DR not properly documented

417x

Return — License cost vs. CDBG-DR exposure (conservative)

What Maui County Gets

- Real-time audit dashboard across all 12 departments
- CDBG-DR compliance infrastructure that satisfies HUD reporting requirements
- Community participation and resident voice portal (Kīpuka Access)
- Agriculture program traceability (WFCF blockchain layer)
- Digital twin for scenario-planning on capital projects and recovery spending
- Generational Wealth Trust (GWT) implementation — 12.5% of public contracts to community

What MauiOS Costs Without Acting

- Continued HUD non-compliance exposure on $1.639B CDBG-DR allocation
- Lahaina recovery delays from fragmented project management tools
- Missed co-application opportunities on FEMA BRIC ($25M) and DOE REAP ($10M)
- No unified grant tracking as $357M in federal revenue flows through 20+ sources
- Foregone GWT community wealth distribution of up to $248M/yr at full deployment

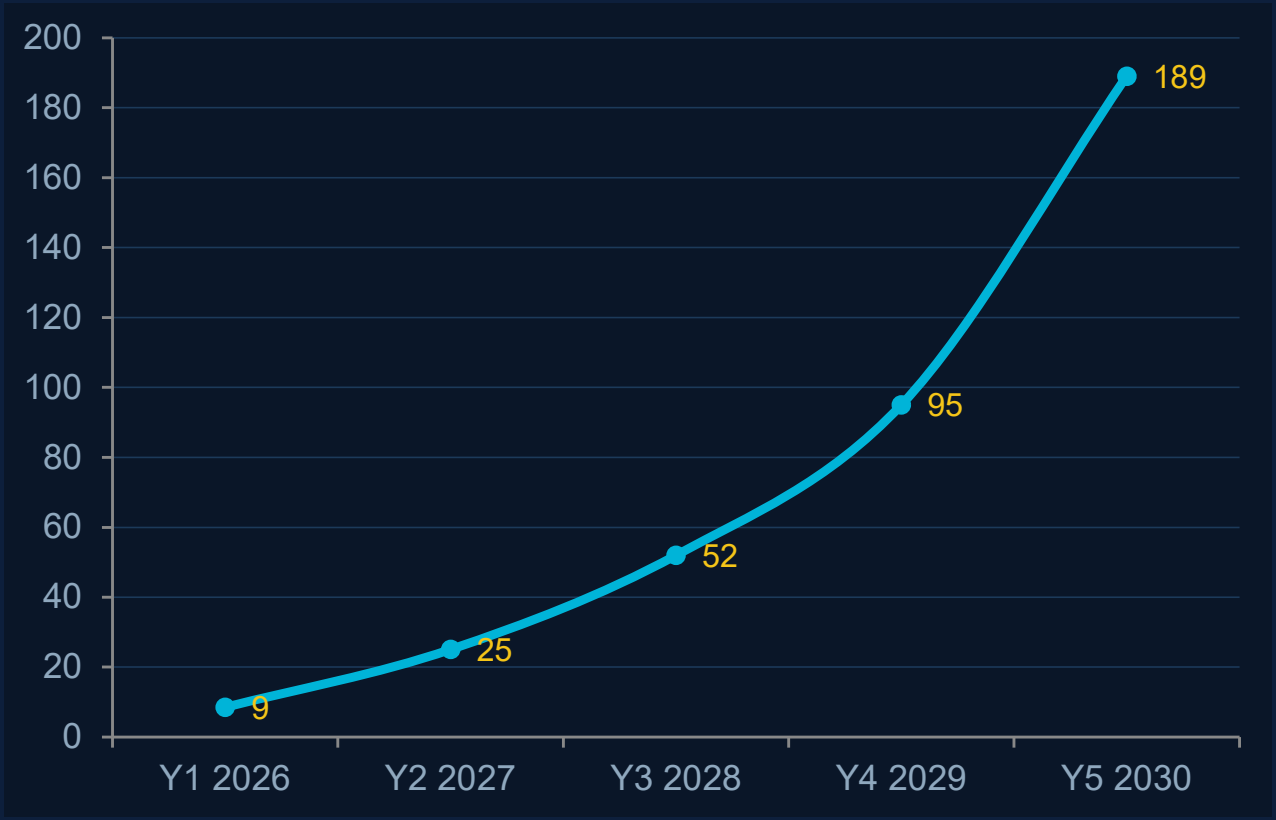
Grant Pipeline — $1.75B+ Accessible with County Partnership

MauiOS enables co-application on every major federal grant

Grant Program	Amount	Timeline	MauiOS Role
HUD CDBG-DR (Lahaina Recovery)	**$1.639B**	Active (Action Plan)	Accountability infrastructure + project management platform
USDA VAPG (Hua Lani — Hawaii)	$500K	Next NOFA (Fall 2026)	Digital twin + grant compliance tracking
USDA VAPG (FarmBox Foods — Colorado)	$500K	Next NOFA (Fall 2026)	FarmBox Foods Fresh co-applicant coordination
FEMA BRIC (Wildfire Mitigation)	$25M	Q2 2026	Node 53 Emergency Override + IoT early warning
DOE REAP (Rural Energy)	$10M	Q2 2026	FarmBox solar integration + ag energy metering
USDA Rural Dev CF	$5M	Rolling	Community food infrastructure — Hua Lani site
USDA 502 Direct (Hua Lani Property)	$3.4M	Active	Land acquisition financing — under review
USDA RBDG	$500K	Q3 2026	Rural business development — Hua Lani operators
TOTAL PIPELINE	**$1.784B+**		County partnership required for co-application eligibility

5-Year Financial Model

Maui County as the anchor — franchise expansion to 50 municipalities



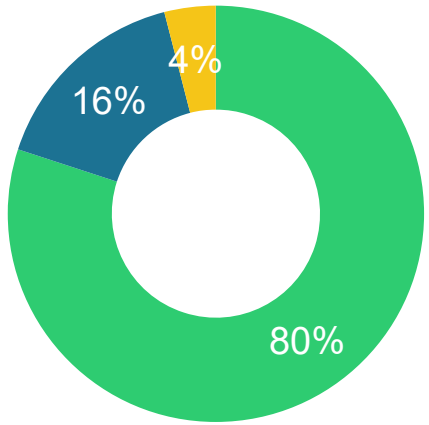
Y1 2026	$8.5M	Maui County license + VAPG + USDA 502
Y2 2027	$25M	3 county licenses + CDBG-DR deployment fees
Y3 2028	$52M	10 counties + FEMA BRIC + Sage Game licensing
Y4 2029	$95M	25 counties + franchise + IoT fleet revenue
Y5 2030	$189M	50 counties + national ag network + GWT flow

5-Year Cumulative: $369.5M

Generational Wealth Trust (GWT)

12.5% of all public contracts directed to community wealth — built into the platform

The 12 Stones Sovereign Charter mandates a Generational Wealth Trust allocation on every public contract processed through MauiOS. At full deployment against Maui County's $1.991B budget, this represents $248M per year flowing to community members — tracked transparently on the blockchain.



10% — Community Fund

80% of GWT flow — direct distribution to Maui families

2% — Trust Management

16% of GWT — HCA administration, financial literacy programs

0.5% — Langford Fiduciary

4% of GWT — charter stewardship fee (James Langford)

FY2026 GWT Baseline: $46.45M · Full deployment: $248.9M/yr

Implementation Timeline

24-month MOU → production-ready county platform

Phase 1 Preliminary
Month 0-1

- MOU signed
- Hua Lani survey
- SAM.gov active

Phase 2 Design
Month 1-3

- Platform design
- RAIS alpha live
- BRIC application

Phase 3 Build & Deploy
Month 3-18

- FarmBox deployed
- MauiOS in prod
- CDBG-DR tracked

Phase 4 QA & Audit
Month 18-21

- Safety QA
- WFCF audits
- HUD review

Phase 5 Handover
Month 21-24

- County handover
- RAIS documented
- Y2 expansion

The Ask — Three Actions to Start

Execute the MOU · License MauiOS · Co-apply on CDBG-DR

01

Execute the MOU

Authorize Maui County Office of Recovery to execute the 5-party MOU with Langford Consulting Trust, FarmBox Foods, HCA, and WFCF. 24-month, $10M, 5-phase scope. Drafted — ready for signature.

Contact: Josiah Nishita
Timeline: 60 days to Phase 1

02

Authorize the MauiOS License

Approve $120K/yr Tier 2 Municipal County License — full 52-node HFEv4, RAIS blockchain audit, digital twin, community portal, and all 6 software modules.

Term: 12 months, renewable
CDBG-DR module included

03

Co-Apply on CDBG-DR + FEMA BRIC

Authorize Office of Recovery as co-applicant on the CDBG-DR Action Plan and FEMA BRIC application. 12 Stones is the technology subrecipient. Both applications are drafted.

FEMA BRIC: Q2 2026
CDBG-DR: active now

12 STONES GLOBAL INC.

REGENERATE · EDUCATE · GOVERN · LIBERATE

Ready to talk.

James Langford

CEO & Founder, 12 Stones Global Inc.

jimlangford@me.com

12 Stones Sovereign Charter · SSC v5 · Filed Maui County Council, April 14, 2025

MauiOS · HFEv4 Digital Twin · RAIS Blockchain Audit · Generational Wealth Trust · Sovereign Charter

MauiOS / HFEv4 System Build Order

Council-Aligned Development Sequence | April 2026

12 Stones Global Inc. · Maui County Office of Recovery · FarmBox Foods

May 30	Apr 7	Apr 4 + 18	Jun 2026
RFP Deadline	CDBG-DR Comment Deadline	Council Meetings	FY2027 Budget Vote

Document Purpose

This document translates April 2026 Maui County Council legislative priorities and active CDBG-DR program areas into a sequenced, actionable build order for the MauiOS platform and HFEv4 Digital Twin. Each development phase is anchored to a specific, confirmed council action so that the team builds what the county actually needs — in the order the county needs it.

Build sequence is not arbitrary. It is derived from public record: live appropriation bills, CDBG-DR Substantial Amendment timelines, FY2027 budget priorities, and MEMA active technology procurements.

Document ID	SGI-BUILD-2026-001
Prepared by	12 Stones Global Inc. — James Langford, CEO
Date	April 2026
Status	ACTIVE — In Development
Related Documents	RFP No. CORR-2026-001 | MOU-12SG-MAUI-2026-001
Budget Authority	$10M MOU Budget | $1.639B CDBG-DR Available

1. April 2026 Council Priority Intelligence

The following confirmed legislative and administrative actions by Maui County govern the priority order for MauiOS system development. Sources are drawn from Maui County Council public records, Office of Recovery announcements, and MEMA procurement documentation.

1.1 CDBG-DR Substantial Amendment 1 — Active Deadline

URGENT: April 7, 2026 Comment Deadline

The CDBG-DR Substantial Amendment 1 public comment period closes April 7, 2026. The April 4 council meeting will include testimony on this amendment. Finalizing Node 1-18 (Mauka) and Node 37-52 (Makai) data in DT_Playable_Maps is the single highest-value action before this deadline. A populated node map demonstrates to the county that 12 Stones' platform is ready to absorb CDBG-DR dollars immediately.

Confirmed Amendment 1 program areas directly mapped to MauiOS modules:

- Housing Reconstruction Assistance Program (HRAP) — maps to Zone 1 Mauka residential nodes (1-18)
- New Construction Program — maps to Zone 3 Makai commercial/mixed-use nodes (37-52)
- Strategic Voluntary Mitigation Buyout Program (SVMBP) — maps to Node 53 (FEMA BRIC emergency override)
- $5.44M planning reallocation to implementation activities — funding available Q2 2026
- $480M supplemental economic development + water infrastructure CDBG allocation — targets Zone 2 Farmlands + Zone 3 water nodes

1.2 MEMA Hazard Mitigation — Active Technology Procurement

Technology Gap: MEMA Needs What Node 53 Delivers

MEMA is actively building an evacuation technology platform with live status tracking, real-time notifications, and a community mobile app. FY2027 budget includes a new dedicated MEMA hazard mitigation section covering wildfire risk, flood, and coastal resilience. The WeatherManager / Node 53 (Joker.Emergency) module is the exact system MEMA is procuring. The Kipuka community alert portal is the mobile app layer they need.

- FY2027 MEMA allocation: New line item for evacuation + alert technology
- Active wildfire/flood resilience mandate overlaps directly with FARSITE integration in WeatherManager (AC_Biome_WeatherManager_Abstract)
- Emergency Bills 53 & 54: $15M Kona Low storm recovery (March 2026) — demonstrates active use of emergency infrastructure funding channels

1.3 Kaupo Digital Hub — CDBG Digital Platform Precedent

Funding Pathway Proven: CDBG-DR for Digital Platforms

The Kaupo Digital Hub Initiative was approved by council on March 6, 2026, establishing clear precedent that CDBG dollars can fund community digital technology infrastructure. This directly validates the 12 Stones CDBG-DR funding strategy for MauiOS and removes the question of whether a software/platform project qualifies for these funds.

- Kaupo Digital Hub: CDBG-funded community technology project — approved March 6, 2026
- Establishes that "digital community platform" is an eligible CDBG-DR use for Maui County
- The RAIS performance dashboard directly satisfies the Office of Recovery's stated mandate for "financial transparency and accountability measures" and "performance dashboards for stakeholders"

1.4 FY2027 Budget Deliberation — Top Priority Areas

Maui County FY2027 budget ($1.616B total) is currently in council deliberation. Three areas intersect directly with HFEv4 system components:

Budget Area	Allocation	HFEv4 Module	Node Scope
Housing Recovery CIP	$79.5M (10 projects)	Zone 1 Mauka governance	Nodes 1-18
Water Infrastructure	$44.5M	Zone 3 Makai infrastructure	Nodes 37-52
MEMA Hazard Mitigation	New FY2027 line item	WeatherManager / Node 53	Node 53 + Jokers
Agricultural Economic Dev	USDA VAPG + CDBG-DR	FarmingSystem / Zone 2	Nodes 19-36
Digital Platform Services	CDBG-DR eligible	MauiOS Dashboard + RAIS	All 52 + Jokers

Council meeting schedule: 1st and 3rd Fridays. April 4 and April 18, 2026. Both meetings expected to carry CDBG-DR Substantial Amendment testimony and FY2027 budget line-item deliberations.

2. MauiOS Design System — Component Inventory

MauiOS is a dual-interface platform: (1) a county administration dashboard for the 52-node HFEv4 digital twin, and (2) a public community portal (Kipuka) for resident access. The design system below documents all platform components, their Unreal Engine source systems, and build readiness.

2.1 Design Tokens

Token Category	Current State	Source System	Status
Node Zone Colors	Zone 1 Blue / Zone 2 Green / Zone 3 Teal / Jokers Red	MapManagement / DT_Playable_Maps	DEFINED
Element Colors	Fire / Water / Earth / Air palette	DT_SGI_Elements	DEFINED
Bagua Trigram Colors	8-trigram color binding	DT_SGI_BaguaEffects	PENDING
Typography Scale	Calibri 12/14/18/24/36pt hierarchy	WBP_SGI_NodeMap (target)	PENDING
Spacing Scale	4px base grid (720 DXA increments)	UI / CommonUI layer	PENDING
Status Indicators	RAIS.Scroll Active / Sealed / Triggered	DT_SGI_Tags / RAIS.*	DEFINED
GWT Badge States	Eligible / Funded / Distributed	DT_SGI_Tags / GWT.*	DEFINED

2.2 Core Components

Component	Unreal Source	MauiOS Layer	Build Priority
Node Map (52-node)	DT_Playable_Maps + DT_SGI_Nodes	WBP_SGI_NodeMap	P1 — Phase 1
Node Detail Panel	F_Playable_Maps + F_Plant structs	WBP_SGI_NodeDetail	P1 — Phase 2
RAIS Dashboard	AC_Inventory_SaveManager + RAIS tags	WBP_SGI_RAISDashboard	P1 — Phase 2
Weather/Hazard Monitor	AC_Biome_WeatherManager_Abstract	WBP_SGI_WeatherMonitor	P1 — Phase 1 (Node 53)
Kipuka Alert Portal	EventTrigger + Node 53 biome states	Kipuka Community Portal	P1 — Phase 1
GWT Transaction Feed	Vendor system + GWT tags	WBP_SGI_GWTFeed	P2 — Phase 3
FarmBox Production Board	FarmingSystem + DT_Farmables	WBP_SGI_FarmBoard	P2 — Phase 3
Supply Chain Trace View	Inventory + AC_Inventory_SaveManager	WBP_SGI_TraceView	P2 — Phase 4
Sage Game Interface	RoguelikeDeckbuilder + DT_SGI_Cards	WBP_SGI_SageGame	P3 — Phase 4

Multi-User Session Panel	OnlineMultiplayerFramework	SessionManager (existing)	P3 — Phase 5

2.3 Data Architecture

The canonical data layer is the SGI overlay namespace at /Content/SGI/. All MauiOS components read from this layer. The vendor content at /Content/RLDB/ and /Content/Hyper/Core/ is never modified.

DataTable	Row Count	Source of Truth	Populates
DT_Playable_Maps (existing)	52 rows	HyperFarmingSystemv4 vendor	Node Map base topology
DT_SGI_Nodes (new)	52 rows + 2 Jokers	12SGI — James build	Node metadata + zone tags + RAIS triggers
DT_SGI_Cards (new)	52 rows	12SGI — RoguelikeDeckbuilder	Card/node crosswalk for Sage Game
DT_SGI_Elements (new)	4x4 matrix	12SGI element design	Element interaction multipliers
DT_SGI_BaguaEffects (new)	8 rows	12SGI I-Ching layer	Trigram-to-node effect binding
DT_SGI_Kumulipo (new)	16 rows	12SGI cosmology layer	Emergence stage scheduler
DT_SGI_Polarity (new)	3 rows	12SGI oscillator	Polarity state modifiers
DT_Farmables (existing, enrich)	Existing rows	HyperFarmingSystemv4 vendor	Crop/plant production data for Nodes 19-36

3. Five-Phase Build Sequence

Phases are ordered by council urgency. Each phase delivers a demonstrable, fundable artifact that directly corresponds to an active county program or legislative deadline.

> **Governing Principle**
> Build what the county can fund next. The April 7 CDBG-DR comment deadline and May 30 RFP response deadline define the critical path. Any system that does not support those two submission moments is Phase 3 or later.

P1 — Emergency Resilience + Node Map Skeleton

Council Driver: CDBG-DR Substantial Amendment 1 | MEMA Hazard Mitigation | April 7 Deadline

Deliverable: A populated DT_Playable_Maps with all 52 nodes categorized by zone, plus Node 53 (Joker.Emergency) wired to the WeatherManager biome state machine. This is the minimum viable artifact that demonstrates the digital twin is operational ahead of the CDBG-DR comment period close.

P1 Tasks — Unreal Engine

1. Add SGI Gameplay Tags to DefaultGameplayTags.ini in HyperFarmingSystemv4 project
2. Populate DT_Playable_Maps rows 1-18 (Zone 1 Mauka) with NodeTypeTag, EncounterTagQuery, zone color metadata
3. Populate DT_Playable_Maps rows 37-52 (Zone 3 Makai) with infrastructure/water node metadata
4. Create DT_SGI_Nodes DataTable (54 rows: 52 nodes + 2 Jokers) using F_Playable_Maps as base struct
5. Wire WeatherManager (AC_Biome_WeatherManager_Abstract) E_WeatherStates.Wildfire to Node 53 trigger — set E_Biomes.Emergency biome on Node 53 row
6. Add RAIS.Scroll.Active / Sealed / Triggered tags as Gameplay Tags on DT_SGI_Nodes for all 54 rows

P1 Tasks — MauiOS UI

7. Create WBP_SGI_NodeMap widget — three-zone visual layout reading from DT_Playable_Maps
8. Implement zone color coding: Blue (Mauka) / Green (Farmlands) / Teal (Makai) / Red (Jokers)
9. Display node status badges from RAIS.Scroll tags on each node icon
10. Build Kipuka alert portal MVP: push Node 53 biome state to community notification surface

P1 Output	Council Linkage	Submission Deadline
52-node zone map (visual)	CDBG-DR SA-1 — demonstrates spatial coverage of recovery program areas	April 7 comment period
Node 53 emergency trigger	MEMA FY2027 hazard mitigation mandate + FEMA BRIC scope	May 30 RFP response

Kipuka alert portal MVP	MEMA evacuation technology procurement — community mobile layer	May 30 RFP response
DT_SGI_Nodes (54 rows)	Foundation DataTable powering all subsequent modules	Internal — gates P2

P2 RAIS Bridge + Transparency Dashboard

Council Driver: Office of Recovery Mandate | Kaupo Digital Hub Precedent | RFP Section 4

Deliverable: A working RAIS bridge connecting the UE5 save system to an immutable audit ledger (immudb), plus the county-facing RAIS performance dashboard. This satisfies the Office of Recovery's mandated "financial transparency and accountability measures" and directly mirrors the Kaupo Digital Hub's CDBG-funded community technology model.

P2 Tasks — Unreal Engine

11. Implement AC_SGI_RAISBridge component — wraps AC_Inventory_SaveManager_Abstract
12. On BeginPlay of AC_SGI_RAISBridge: register save delegate from BP_GI_RLDB.AttemptSaveGame
13. On save event fire: HTTP POST to FastAPI /rais/scroll/seal endpoint with node ID + timestamp + state hash
14. Update RAIS.Scroll.Active tag to RAIS.Scroll.Sealed on successful API response
15. Build GWT transaction logger: on fund disbursement event, fire HTTP POST to /gwt/distribute
16. Wire AC_SGI_NodeManager to wrap AC_Farming_Abstract + AC_Attribute_Manager_Abstract — single component per node actor

P2 Tasks — FastAPI / Backend

17. Stand up Python FastAPI with two endpoints: POST /rais/scroll/seal and GET /rais/scroll/{node_id}
18. Connect to immudb instance — use immudb Python SDK for write-once scroll records
19. Each RAIS scroll record: { node_id, timestamp, state_hash, zone, scroll_type, author }
20. Build GET /nodes/dashboard endpoint returning all 54 nodes with current RAIS status + attribute scores

P2 Tasks — MauiOS UI

21. Create WBP_SGI_RAISDashboard — county admin view: live RAIS scroll log + GWT distribution feed
22. Implement node KPI cards reading from AC_Attribute_Manager_Abstract F_Float_Attribute values
23. Build performance dashboard export (CSV) for Office of Recovery stakeholder reporting

P2 Output	Council Linkage	Submission Deadline
RAIS bridge (UE5 → immudb)	Immutable audit record satisfies "accountability measures" mandate	May 30 RFP response

FastAPI RAIS endpoints	Platform infrastructure enabling CDBG-DR reporting compliance	May 30 RFP response
WBP_SGI_RAISDashboard	County performance dashboard — Office of Recovery requirement	May 30 RFP response
GWT transaction log	Financial transparency for GWT community fund distributions	Ongoing / MOU Phase 2

P3 Agricultural Production + FarmBox Integration

Council Driver: USDA VAPG Deadline | FY2027 Agricultural Economic Development | Zone 2 Farmlands

Deliverable: Zone 2 Farmlands nodes (19-36) fully populated in DT_SGI_Nodes, with the FarmingSystem (DT_Farmables) enriched to reflect actual FarmBox Foods Fresh production data, and the Crafting/Inventory traceability chain wired to WFCF verification endpoints.

P3 Tasks — Data Population

24. Populate DT_SGI_Nodes rows 19-36 (Zone 2 Farmlands) with: FarmBox unit locations, crop types from DT_Farmables, ElementBias tags, Node.Zone2.Farmlands.* gameplay tags
25. Enrich DT_Farmables with actual FarmBox Foods Fresh production catalog: mushroom varieties, leafy greens, herbs, hydroponic crops
26. Create DT_SGI_Cards (52-row) in RoguelikeDeckbuilder project — map each card to corresponding node row in DT_Playable_Maps by RowName
27. Set ECardType mapping: Production nodes (Attack) / Processing nodes (Skill) / Infrastructure nodes (Power) / Emergency/Audit nodes (Status)

P3 Tasks — Traceability Chain

28. Wire Crafting system (AC_Crafting_Abstract) to DT_CraftingRecipes with FarmBox product IDs as recipe row keys
29. On harvest event (BPI_Farming interface): fire inventory add to AC_Inventory_Abstract + timestamp to RAIS scroll
30. Implement WFCF verification hook: on crafting completion, POST to WFCF API with product batch ID + RAIS scroll hash
31. Build WBP_SGI_FarmBoard: production dashboard for Zone 2 showing live crop stages (E_GrowingStage) per node

P3 Output	Council Linkage	Submission Deadline
DT_SGI_Nodes rows 19-36	USDA VAPG "change in production method" + FY2027 agri-econ development	USDA VAPG deadline
DT_Farmables enrichment	FarmBox Foods Fresh catalog — MOU Phase 2 deliverable	MOU milestone

DT_SGI_Cards (52 rows)	Sage Game / public engagement layer — dual metaphor complete	Phase 4 gates on this
WFCF traceability hook	MOU Module 5 (Agricultural Traceability) — WFCF partner obligation	MOU Phase 3

P4 Full Node Map + Sage Game Interface

Council Driver: Community Engagement | Section 3 Workforce Simulation | Public Portal

Deliverable: All 52 nodes + 2 Jokers fully operational across both interfaces — the county admin MauiOS dashboard and the public Sage Game engagement layer. The 52-card = 52-node dual metaphor is complete and playable.

P4 Tasks — Integration

32. Finalize AC_SGI_NodeManager — AC_Farming_Abstract + AC_Attribute_Manager_Abstract + RAIS event dispatch unified in single component
33. Load DT_SGI_Cards into DeckBuilderSubsystem.RunDeck at game start — LoadDeckIntoComponent() bridge
34. Wire DeckBuilderSubsystem.CurrentFloor to current active node in DT_Playable_Maps for run progression
35. Implement DeckBuilderSubsystem.SaveRun() → AC_SGI_RAISBridge RAIS scroll seal (run save = audit record)
36. Test full Play-in-Editor session: start run → draw card → play card → node state change → RAIS scroll sealed → immudb write confirmed

P4 Tasks — Community Portal

37. Build Sage Game public web interface using WBP_SGI_SageGame widget export or React wrapper
38. Implement Section 3 workforce simulation: NPCBehaviorSystem E_RelationshipStatus drives community trust score per node
39. Wire community trust scores to GWT.Eligible tags — nodes above threshold become GWT-eligible automatically
40. Build Kipuka portal v2: full community alert + node status + GWT distribution visibility for residents

P5 County Submission + Multi-User Governance

Council Driver: RFP CORR-2026-001 Response | MOU Execution | Live Platform Launch

Deliverable: Complete RFP response package submitted to Maui County by May 30, 2026. MOU executed by all 5 parties. Live MauiOS instance running on HIPAA-grade infrastructure. Multi-user governance framework operational for concurrent county admins and community monitors.

P5 Tasks — Submission Package

41. Finalize technical specifications document referencing DT_SGI_Nodes schema as proof of platform readiness
42. Record platform demonstration video: 52-node map + RAIS dashboard + Kipuka alert + Sage Game walkthrough
43. Complete Maui County SaaS proposal: $120K/yr platform fee + $6M CDBG-DR implementation budget
44. Execute 5-party MOU with Maui County Office of Recovery, FarmBox Foods, HCA, WFCF

P5 Tasks — Multi-User Governance

45. Configure OnlineMultiplayerFramework + ReplicationSubSystem for concurrent admin sessions
46. Assign role-based access: County Admin (full RAIS view) / Community Monitor (node health only) / FarmBox Operator (Zone 2 only)
47. Implement audit trail for all admin actions via RAIS scroll system
48. Load test: minimum 5 concurrent county admin sessions on 52-node simulation

P5 Output	Council Linkage	Deadline
RFP response package	CORR-2026-001 — $6M CDBG-DR + $120K/yr SaaS contract	May 30, 2026
MOU executed (5 parties)	MOU-12SG-MAUI-2026-001 — all partner obligations active	Pre-RFP response
Live MauiOS instance	County performance dashboard requirement — Office of Recovery	MOU Phase 1
Multi-user governance	Concurrent county + community access — CDBG-DR compliance layer	MOU Phase 2

4. DataTable Population Priority Order

Population order follows council priority, not node number. Nodes tied to active CDBG-DR programs are populated first; Sage Game cosmology layers are last.

Priority	DataTable	Rows	Phase	Council Linkage
1	DT_Playable_Maps (enrich existing)	52 + 2 Jokers	P1	CDBG-DR SA-1 spatial coverage
2	DT_SGI_Nodes — Zone 1 Mauka (rows 1-18)	18 rows	P1	CDBG-DR housing reconstruction
3	DT_SGI_Nodes — Joker nodes (rows 53-54)	2 rows	P1	MEMA emergency + audit
4	DT_SGI_Nodes — Zone 3 Makai (rows 37-52)	16 rows	P1-P2	CDBG-DR new construction + water
5	DefaultGameplayTags.ini — all SGI tags	Full taxonomy	P1	Gates all tag-dependent systems
6	DT_SGI_Nodes — Zone 2 Farmlands (rows 19-36)	18 rows	P3	USDA VAPG + FY2027 agri-econ
7	DT_Farmables (enrich existing)	FarmBox catalog	P3	FarmBox MOU obligation
8	DT_CraftingRecipes (enrich existing)	VAPG product catalog	P3	USDA VAPG "change in production"
9	DT_SGI_Cards (new)	52 rows	P3	Sage Game / public engagement
10	DT_SGI_Elements (new)	4x4 matrix	P4	Element interaction system
11	DT_SGI_BaguaEffects (new)	8 rows	P4	Trigram event hooks
12	DT_SGI_Kumulipo (new)	16 rows	P4-P5	Emergence stage scheduler
13	DT_SGI_Polarity (new)	3 rows	P5	Polarity oscillator states

4.1 Node ID to Zone Mapping Reference

Zone	Node Range	Type	CDBG-DR Program Area
Zone 1 — Mauka	1-18	Upland governance / residential	HRAP Housing Reconstruction
Zone 2 — Farmlands	19-36	Agricultural production	USDA VAPG + Economic Development

Zone 3 — Makai	37-52	Coastal / infrastructure / commercial	New Construction + Water Infrastructure
Joker 53	Node 53	Emergency override (FEMA BRIC)	SVMBP + MEMA Hazard Mitigation
Joker 54	Node 54	Audit / compliance node	CDBG-DR Program Audit + RAIS Verification

5. RAIS Bridge Implementation Specification

The RAIS bridge is the most critical missing piece in the entire system. It is the HTTP hook that connects Unreal Engine save events to the immudb immutable audit ledger — transforming a game save into a CDBG-DR-compliant audit record.

> **Why This Is the Critical Path**
> Without the RAIS bridge, MauiOS is a visualization tool. With the RAIS bridge, it is an auditable compliance infrastructure. The Office of Recovery's mandate for "financial transparency and accountability measures" requires this component. It is what distinguishes the 12 Stones platform from a dashboard and makes it fundable at $6M+.

5.1 Unreal Engine Side

Component	Action	Trigger Event
AC_SGI_RAISBridge	HTTP POST to /rais/scroll/seal	BP_GI_RLDB AttemptSaveGame delegate fires
AC_SGI_RAISBridge	Update RAIS.Scroll.Active → RAIS.Scroll.Sealed tag	On 200 OK response from FastAPI
AC_SGI_NodeManager	Dispatch RAIS.Scroll.Triggered tag	On anomaly detection (attribute threshold breach)
AC_Attribute_Manager_Abstract	Push F_Float_Attribute KPI snapshot	On EndTurn / on save event
DeckBuilderSubsystem.SaveRun	Seal run-level scroll (54-node session)	On EndRun() call

5.2 FastAPI Backend

Minimum viable backend for RFP submission. Can be hosted on AWS GovCloud, Azure Government, or HIPAA-compliant cloud.

Endpoint	Method	Payload	Action
POST /rais/scroll/seal	POST	{ node_id, timestamp, state_hash, zone, scroll_type }	Write to immudb — write-once record
GET /rais/scroll/{node_id}	GET	node_id path param	Return scroll history for node
GET /nodes/dashboard	GET	none	All 54 nodes: RAIS status + KPI scores + zone
POST /gwt/distribute	POST	{ node_id, amount, recipient, basis }	Log GWT fund distribution to RAIS scroll
GET /gwt/ledger	GET	optional: ?zone=1	Full GWT distribution history

5.3 immudb Schema

Each RAIS scroll record written to immudb contains the following fields:

- node_id — integer (1-54, maps to DT_Playable_Maps row index)
- timestamp — ISO 8601 UTC timestamp
- state_hash — SHA-256 of node attribute snapshot
- zone — Zone 1 / Zone 2 / Zone 3 / Joker.Emergency / Joker.Audit
- scroll_type — Compliance | Production | Emergency | Governance | Distribution
- author — county admin user ID or system ("UE5_AUTO")
- rais_version — schema version for forward compatibility
- previous_hash — hash of prior scroll for chain integrity (blockchain-style linked record)

> **Compliance Note**
> immudb tamper-evident logs satisfy CDBG-DR financial transparency requirements and provide the audit trail needed for HUD program compliance review. The previous_hash chain makes any retroactive data alteration cryptographically detectable.

6. Critical Path Timeline

Date	Milestone	Phase	Owner
April 4, 2026	Council meeting — CDBG-DR SA-1 testimony + FY2027 budget deliberation	Anchor	Monitor + respond
April 7, 2026	CDBG-DR Substantial Amendment 1 comment period closes	P1 gate	12 Stones submit comments
April 7, 2026	DT_SGI_Nodes Zone 1 + Zone 3 + Joker rows complete	P1 deliverable	James — UE build
April 7, 2026	SGI Gameplay Tags added to DefaultGameplayTags.ini	P1 deliverable	James — UE build
April 18, 2026	Council meeting — FY2027 budget continued deliberation	Anchor	Monitor MEMA line item
April 25, 2026	WBP_SGI_NodeMap v1 functional (zone map visual)	P1 UI deliverable	James — UE build
April 30, 2026	Node 53 WeatherManager emergency trigger wired	P1 deliverable	James — UE build
May 7, 2026	AC_SGI_RAISBridge + FastAPI /rais/scroll/seal operational	P2 deliverable	James — backend
May 7, 2026	USDA VAPG application submitted (if open round)	P3 dependency	External deadline
May 14, 2026	WBP_SGI_RAISDashboard v1 functional	P2 UI deliverable	James — UE build
May 21, 2026	Zone 2 Farmlands nodes + DT_Farmables enriched	P3 deliverable	James — data population
May 25, 2026	RFP response package assembled and reviewed	P5 prep	12 Stones + counsel
May 30, 2026	RFP No. CORR-2026-001 response submitted to Maui County	P5 submission	Hard deadline
June 2026	FY2027 budget adoption (expected)	P5 context	Council action
July 2026	MOU execution target (5-party signing)	P5 execution	All parties

Q3 2026	DT_SGI_Cards + Sage Game interface (P4)	P4 deliverable	James — UE build
Q4 2026	Multi-user governance + live platform launch (P5)	P5 deliverable	Full team

6.1 Struct Safety Protocol

> **UE5.7 Struct Edit Rule — Must Follow Every Time**
>
> When editing Blueprint structs in UE5.7 (especially F_Playable_Maps, F_Plant, F_Buildable_Info): Save the project, edit the struct only, then exit the struct editor WITHOUT saving any other assets. Reopen. Struct edits can silently break references across thousands of assets. Always validate with Play-in-Editor after any struct change before committing new DataTable rows.

7. Summary — Why This Build Order Works

This build order is not a software roadmap. It is a procurement strategy. Every deliverable is timed to a county funding decision:

- Phase 1 completes before April 7 so the digital twin is visible when CDBG-DR amendment comments close
- Phase 2 completes before May 30 so the RAIS bridge is operational when the RFP response demonstrates platform readiness
- Phase 3 delivers agricultural data in time for USDA VAPG and FY2027 budget justification
- Phase 4 delivers the Sage Game community layer once county funding is secured
- Phase 5 formalizes the relationship through MOU execution and platform go-live

The Maui County Council is actively deliberating on CDBG-DR programs, MEMA technology, water infrastructure, and agricultural economic development. These are not distant policy discussions — they are live appropriation decisions happening in April and May 2026. The system build order above puts the right component in front of the right decision-maker at the right moment.

> **12 Stones Global Position**
> The only platform that maps all 52 community nodes to an auditable, blockchain-anchored, real-time digital twin — with a public engagement layer, a FEMA BRIC-compliant emergency module, and a community benefit trust distribution engine — built on proven Unreal Engine 5 infrastructure already in development.

12 Stones Global Inc. | James Langford, CEO | jimlangford@me.com | Document ID: SGI-BUILD-2026-001

CONFIDENTIAL — Prepared for internal use and Maui County submission.

U.S. Department of Housing and Urban Development

Community Development Block Grant — Disaster Recovery (CDBG-DR)

Lahaina Wildfire Recovery Program · Action Plan Subrecipient Application

Technology & Community Resilience Infrastructure

A Comprehensive Accountability, Housing, and Food Security Platform

for the Lahaina Wildfire Recovery — Maui County, Hawaii

Field	Details
Primary Applicant	Langford Consulting Trust / 12 Stones Global Inc.
Authorized Official	James Langford, CEO & Founder
Location	Maui County, Hawaii (Lahaina Recovery Zone)
Email	jimlangford@me.com
Grantee (Co-Applicant)	Maui County Office of Recovery (to be confirmed as co-applicant)
CDBG-DR Allocation	**$1,639,000,000 (HUD Allocation — Lahaina Wildfire Recovery)**
Funds Requested (Subrecipient)	**$47,500,000 over 5 years (Technology + Housing + Agriculture)**
HUD National Objective	Low-to-Moderate Income Benefit (LMI) — Urgent Need — Slum/Blight Elimination
Project Categories	Housing (24 CFR 570.202, 570.208) · Technology Infrastructure · Economic Development · Agriculture & Food Security

UEI / SAM Registration	*[UEI TO BE INSERTED — active SAM.gov registration required prior to submission]*

Section I: Executive Summary

Langford Consulting Trust / 12 Stones Global Inc. submits this subrecipient application under Maui County's HUD CDBG-DR Action Plan for the Lahaina Wildfire Recovery. We request $47.5 million over five years to deploy three integrated recovery programs: (1) a county-wide technology and accountability platform (MauiOS) to manage and transparently report on CDBG-DR expenditures; (2) a modular affordable housing deployment program targeting Lahaina and West Maui displaced families; and (3) a distributed food security network anchored at the Hua Lani agricultural site in Kula, deploying 52 FarmBox hydroponic production units to reduce Maui's 85-95% food import dependence.

The 12 Stones platform is uniquely positioned for this work because it is purpose-built for exactly the accountability and community resilience challenge HUD's CDBG-DR program demands. The Real-time Accountability & Intelligence System (RAIS) — 188 immutable audit scrolls sealed on immudb blockchain — provides the documented financial management controls, citizen participation records, and project performance tracking that HUD requires as a condition of CDBG-DR drawdowns. The 52-node digital twin of Maui County (HFEv4) provides scenario-planning infrastructure for recovery decisions across every county department.

> This application should be read in conjunction with the 12 Stones Sovereign Charter (SSC v5, 28 Articles), filed with Maui County Council on April 14, 2025, which establishes the governance framework, community wealth distribution model (GWT — 12.5% of public contracts), and IP ownership structure underlying the proposed platform.

Summary of Requested Funds by Program Area

Program Area	Year 1-2	Year 3-5	Total Request
MauiOS Technology & Accountability Platform	$2,400,000	$3,600,000	$6,000,000
Modular Affordable Housing (Lahaina/West Maui)	$12,000,000	$18,000,000	$30,000,000
Food Security Network (Hua Lani + 52 FarmBox Nodes)	$3,500,000	$5,250,000	$8,750,000
Community Wealth & Financial Literacy (HCA/GWT)	$900,000	$900,000	$1,800,000
Program Administration & Reporting	$450,000	$500,000	$950,000
TOTAL SUBRECIPIENT REQUEST	**$19,250,000**	**$28,250,000**	**$47,500,000**

Section II: Statement of Need — Lahaina Wildfire Recovery Context

A. Disaster Overview

On August 8, 2023, the deadliest United States wildfire in over a century devastated the historic town of Lahaina on the western coast of Maui, Hawaii. The fire killed 102 people, destroyed over 2,200 structures, displaced approximately 12,000 residents, and burned nearly 2,200 acres in the Lahaina and Kaanapali areas. Total damage estimates exceed $5.5 billion.

HUD allocated $1,639,000,000 in CDBG-DR funds to Maui County for recovery. As of Q1 2026, approximately 88% of this allocation — over $1.4 billion — remains undeployed. The Maui County Office of Recovery has an approved Action Plan, but deployment has been slowed by fragmented project management infrastructure, limited technology capacity to document and report expenditures at HUD standards, and the complexity of coordinating housing, economic development, and community services simultaneously at this scale.

> Critical Gap: HUD's CDBG-DR program requires documented accountability infrastructure — financial management controls, audit trails, citizen participation records, and quarterly performance reporting — before and during drawdowns. Maui County does not currently have a unified technology platform meeting these requirements at the scale of a $1.639B recovery program.

B. Housing Need

The wildfire displaced an estimated 12,000 Maui residents, the majority of whom were low-to-moderate income families, Native Hawaiian households, and migrant workers in the tourism sector. As of early 2026, thousands remain in temporary accommodations. The pre-existing Maui housing crisis — characterized by median home prices exceeding $1.2M and rental vacancy rates below 2% — makes market-rate rehousing inaccessible for most displaced families.

Maui County's FY2026 budget allocates $32.5M to the Affordable Housing Fund (5% of certified Real Property Tax revenue). This represents the County's commitment but is far short of the estimated $400M-$600M needed to rehouse all displaced Lahaina households at long-term affordability standards. CDBG-DR housing investment is the primary mechanism available to close this gap.

C. Food Security Need

Maui imports 85-95% of its food supply, with most goods arriving by container ship from mainland ports. The Lahaina wildfire exposed the fragility of this dependence: supply disruptions during and immediately after the fire left West Maui communities without food access for multiple days. Even under normal conditions, Maui's food security is structurally compromised by its near-total reliance on off-island production.

The USDA has identified Maui County as a priority for local food system investment. The County's FY2026 agriculture budget includes 17 active programs totaling $10.5M+, including the USDA Microenterprise grants program ($3M), agricultural operations and infrastructure support, and educational programs. 12 Stones' Hua Lani agricultural project — a 9.2-acre site in Kula, Maui

zoned for agriculture and pending USDA 502 Direct Loan approval — provides the anchor infrastructure for a distributed 52-node food production network.

D. Community Accountability Need

Post-wildfire community trust in government institutions has been severely strained by the pace and transparency of recovery spending. Lahaina residents, Native Hawaiian organizations, and community advocates have publicly demanded documented evidence that CDBG-DR funds are being deployed equitably, efficiently, and with meaningful community participation. The 12 Stones Sovereign Charter — filed with Maui County Council in April 2025 — was developed in direct response to this accountability demand, establishing a Generational Wealth Trust (GWT) model that directs 12.5% of all public contracts to community members as documented, blockchain-verified distributions.

Section III: Program Description

Program Area 1 — MauiOS Technology & Accountability Platform ($6,000,000)

MauiOS is a real-time county governance and accountability SaaS platform built on the 12 Stones Hyper Farming Engine v4 (HFEv4) — a 52-node digital twin of Maui County mapping every department, program, and capital project to a live audit layer. The platform directly addresses the accountability infrastructure gap identified in Section II.D and satisfies HUD CDBG-DR financial management and reporting requirements.

Core Components

- RAIS (Real-time Accountability & Intelligence System): 188 immutable audit scrolls sealed on immudb blockchain. Automated flagging of expenditure deviations. Real-time compliance triggers on 4 enforcement categories: fraud detection, financial audit, federal compliance, and accountability. All CDBG-DR drawdowns logged with blockchain-verified documentation chain.
- HFEv4 Digital Twin: 52-node live model of Maui County departments and programs. Enables scenario-planning for housing deployment, food security investments, and recovery spending allocation. Zones: Mauka (governance, nodes 1-18), Farmlands (agriculture/culture, nodes 19-36), Makai (revenue/franchise, nodes 37-52).
- Kīpuka Access Portal: Mobile and web community participation platform. Residents submit testimony, track recovery project status, and record concerns for formal inclusion in mediation proceedings. Satisfies HUD citizen participation requirements.
- MauiOS Budget Dashboard: Real-time visualization of all CDBG-DR expenditures against Action Plan commitments. Automated quarterly performance report generation for HUD submission. Financial variance alerts with 24-hour escalation protocols.
- Generational Wealth Trust (GWT) Tracking: 12.5% of all CDBG-DR subcontracts and public contracts processed through MauiOS are automatically flagged for GWT community distribution tracking. Splits: 10% to community fund (80% of GWT), 2% to trust management (16%), 0.5% to Langford fiduciary stewardship (4%).

HUD Compliance Capabilities

HUD Requirement (24 CFR 570)	MauiOS Capability
Financial Management Controls (§85.20)	RAIS blockchain audit trail; real-time expenditure monitoring; automated variance alerts; 24-hour escalation
Citizen Participation Plan (§91.105)	Kīpuka Access Portal — resident testimony submission; public comment tracking; community meeting records; dispute resolution documentation
Quarterly Performance Reports	Automated report generation from RAIS data; milestone tracking against Action Plan; output/outcome metrics by program area
Procurement Standards (§85.36)	All contracts processed through MauiOS with documented competitive procurement tracking and GWT allocation verification

National Objective Documentation	LMI beneficiary tracking at household and community level; urgent need documentation; program-specific national objective evidence files
Environmental Review Records	Digital twin integration with GIS for site-specific environmental review documentation; NEPA compliance tracking per project node

Program Area 2 — Modular Affordable Housing ($30,000,000)

12 Stones Global will deploy a modular housing program targeting displaced Lahaina and West Maui households. The program utilizes FarmBox-adjacent modular construction technology — MauiPods — pre-fabricated housing units manufactured using recycled PETG/PLA feedstock supplied by Aloha Recycling (75 Amala Place, Maui). This approach reduces construction time, waste, and cost while keeping manufacturing employment on-island.

Program Components

- Site Acquisition & Preparation: Identification and preparation of 3-5 West Maui sites for modular housing clusters, in coordination with Maui County Office of Recovery permitting and County Planning Department. Target: 200-400 affordable housing units over 5 years.
- Unit Construction & Deployment: Pre-fabricated modular units at targeted cost of $75,000-$120,000 per unit, depending on size (studio to 3-bedroom). Energy-efficient design with solar-ready roofing and water conservation systems. Estimated 250 units over project term.
- Affordability & Tenure Structure: Units targeted at households earning 30-80% of Maui County Area Median Income (AMI). Mixed-tenure model: 50% rental assistance (Section 8 compatible), 30% lease-to-own through HCA homeownership pathway, 20% community land trust.
- Hawaiian Community Assets (HCA) Integration: HCA financial literacy and homeownership counseling provided to all households entering lease-to-own track. GWT distributions used to fund down-payment assistance and emergency household financial reserves.
- Native Hawaiian Housing Priority: Consistent with CDBG-DR equity requirements and HUD's commitment to Native Hawaiian community recovery, minimum 40% of housing units prioritized for Native Hawaiian families displaced by the Lahaina fire.

Program Area 3 — Food Security Network ($8,750,000)

The 12 Stones food security program deploys 52 FarmBox hydroponic container units across Maui County's agricultural zones, anchored at the 9.2-acre Hua Lani property in Kula. Each unit is a 40-foot, climate-controlled container producing 175-200 lbs of vegetables per week — the equivalent of 2.5 acres of conventional farmland output. At full deployment, the 52-node network produces an estimated 9,100-10,400 lbs of fresh produce per week for local consumption.

Deployment Plan

Phase	Timeline	Units Deployed	Target Communities / Sites

Phase 1	Year 1	6 units	Hua Lani anchor site (Kula) — pilot production and operator training hub
Phase 2	Year 2	12 units	West Maui recovery zone (Lahaina/Kaanapali) — food access for displaced communities
Phase 3	Year 3	18 units	Central Maui (Wailuku/Kahului) and South Maui (Kihei) — school cafeteria + hospital supply
Phase 4	Year 4-5	16 units	East Maui (Hana), Upcountry (Kula/Makawao), and Molokai/Lanai partner sites
TOTAL	**5 Years**	**52 Units**	Full 52-node Maui food production network operational

- Where Food Comes From (WFCF) Integration: All 52 nodes certified under WFCF blockchain traceability at $3,000/farm/year. Provides farm-to-table supply chain documentation for restaurant, institutional, and direct-to-consumer buyers.
- Operator Training: FarmBox Foods (CEO Rusty Walker) trains local operators for each unit. Target: 52 full-time equivalent farm operator positions, prioritizing Lahaina disaster-displaced workers and Native Hawaiian agricultural practitioners.
- Revenue Model: Units operated by trained community members under a revenue-sharing model aligned with the GWT framework. Each unit generating $6,000-$10,000/month in produce revenue funds operator livelihoods and GWT community distributions.

Program Area 4 — Community Wealth & Financial Literacy ($1,800,000)

Hawaiian Community Assets (HCA) delivers financial literacy education, homeownership counseling, and GWT distribution facilitation to Lahaina recovery households. HCA is an established Maui non-profit with documented experience in Native Hawaiian financial empowerment and homeownership pathway programs — making it the ideal partner for the community wealth dimension of this CDBG-DR application.

- Financial Literacy Workshops: Quarterly workshops in Lahaina/West Maui serving displaced households. Topics: budgeting, credit rebuilding, homeownership readiness, insurance recovery, federal assistance navigation.
- GWT Distribution Administration: HCA manages the 10% Community Fund distribution from all CDBG-DR contracts processed through MauiOS. Transparent, blockchain-verified distributions to eligible Lahaina recovery households.
- Homeownership Pathway Counseling: One-on-one HUD-approved housing counseling for households entering the lease-to-own track under Program Area 2.

Section IV: National Objective Compliance

A. Low-to-Moderate Income (LMI) Benefit — Primary National Objective

All four program areas primarily benefit low-to-moderate income persons as defined under 24 CFR 570.208(a). The Lahaina wildfire disproportionately impacted LMI households, who had fewer financial resources to evacuate, lost rental housing with no equity to fall back on, and face the highest barriers to recovery in Maui's extreme housing market.

Program Area	LMI Qualification Method	Documentation
MauiOS Platform	Area Benefit (LMA)	Platform serves all Maui County residents; at least 51% LMI per Census tract data for recovery zone
Affordable Housing	LMH — Housing Benefit	All units restricted to households at 30-80% AMI; income verification at occupancy; deed restrictions recorded
Food Security Network	LMC — Limited Clientele	Units sited in LMI census tracts; produce distributed to food pantries and community food programs serving LMI households
Community Wealth / GWT	LMC — Limited Clientele	GWT distributions targeted to displaced Lahaina households; HCA maintains income-qualified beneficiary list

B. Urgent Need — Secondary National Objective

Program Area 2 (Housing) qualifies under the Urgent Need national objective in addition to LMI benefit. The Lahaina wildfire constitutes an officially declared federal disaster (FEMA DR-4724-HI), creating conditions of urgency that pose a serious and immediate threat to the health and welfare of the community for which the County of Maui does not have sufficient resources to address.

Section V: Implementation Plan and Timeline

Period	Program Activity	Key Milestones & Deliverables
Q1 Year 1	All Programs	Subrecipient agreement executed; SAM.gov registrations active; MOU with all 5 partners signed; environmental review initiated for housing and food security sites
Q2-Q3 Year 1	MauiOS	Platform modules 1-6 in alpha; RAIS live with CDBG-DR tracking; Kīpuka portal launched; HUD financial management controls documentation complete
Q2-Q4 Year 1	Housing	Site selection final (3-5 West Maui sites); environmental review NEPA clearance; first procurement package for modular units issued; HCA counseling begins for initial 50 households
Q3-Q4 Year 1	Food Security	Hua Lani property closing (USDA 502 Direct); 6 FarmBox units deployed; WFCF certification for Nodes 1-6; first harvest delivered to Lahaina food access programs
Year 2	All Programs	50+ housing units occupied; 18 FarmBox nodes producing; MauiOS fully operational with first annual HUD performance report submitted; GWT distributions tracking $2M+
Year 3	All Programs	150+ housing units occupied; 36 FarmBox nodes producing; MauiOS expanded to 3 additional county programs; CDBG-DR drawdown pace at $8-10M/year
Year 4-5	All Programs	250 housing units at full occupancy; 52 FarmBox nodes producing 10,000+ lbs/week; full RAIS operational transfer to Maui County; GWT distributed $10M+ to community

Section VI: Budget Detail

A. Full Budget by Program and Year

Budget Category	Yr 1	Yr 2	Yr 3	Yr 4-5	Total
MauiOS Platform Build & Licensing	$1,200,000	$600,000	$720,000	$3,480,000	**$6,000,000**
Housing — Site Acquisition	$2,000,000	$2,000,000	$1,500,000	$0	**$5,500,000**
Housing — Construction (250 units @ $98K avg)	$4,900,000	$7,350,000	$7,350,000	$4,900,000	**$24,500,000**
FarmBox Units — 52 nodes ($33,200 each)	$199,200	$398,400	$597,600	$531,200	**$1,726,400**
Food Security — Ops, Training & WFCF Certs	$700,000	$1,200,000	$1,800,000	$3,323,600	**$7,023,600**
HCA — Counseling, GWT Admin & Literacy	$360,000	$360,000	$360,000	$720,000	**$1,800,000**
Program Administration & HUD Reporting	$190,000	$190,000	$190,000	$380,000	**$950,000**
TOTAL	**$9,549,200**	**$12,098,400**	**$12,517,600**	**$13,334,800**	**$47,500,000**

Note: Housing construction costs are based on $75,000-$120,000 per modular unit using Maui-manufactured MauiPods (Aloha Recycling feedstock). Cost will be finalized upon site-specific engineering review. FarmBox unit costs are based on manufacturer's documented price of $33,200/unit.

Section VII: Organizational Capacity

A. Applicant Qualifications

12 Stones Global Inc. / Langford Consulting Trust brings 15+ months of documented platform development (January 2025 - March 2026), spanning 906 source documents across 8+ research domains, toward the design and deployment of the MauiOS county intelligence platform. The 12 Stones Sovereign Charter (SSC v5) was formally filed with Maui County Council on April 14, 2025, demonstrating demonstrated engagement with county governance processes.

B. Partner Network Qualifications

Partner	Qualifications Relevant to CDBG-DR Work
FarmBox Foods (Rusty Walker)	85+ container farms deployed globally; 2025 "Coolest Thing Made in Colorado" winner; patented vertical hydroponic technology; active agricultural producer via FarmBox Foods Fresh (Sedalia, CO). Full equipment supply, installation, and training capability.
Hawaiian Community Assets (HCA)	Established Maui non-profit with documented track record in Native Hawaiian financial literacy, homeownership pathways, and community asset-building. HUD-approved housing counseling agency. Ideal partner for CDBG-DR citizen participation and LMI beneficiary documentation requirements.
Where Food Comes From (WFCF)	Leading blockchain-based agricultural traceability and sustainability certification provider. $3K/farm/year certification creates documented supply chain transparency meeting USDA and HUD program integrity standards for agricultural investments.
Maui County Office of Recovery	HUD-designated grantee for the $1.639B CDBG-DR allocation. Active CDBG-DR Action Plan. Authority to designate subrecipients under 24 CFR 570.500. Partnership with 12 Stones as the technology subrecipient enables the County to dramatically increase deployment capacity and HUD compliance infrastructure.

Section VIII: Community Benefit and Equity

A. Beneficiary Population

The primary beneficiaries of this CDBG-DR subrecipient program are Lahaina and West Maui residents displaced by the August 8, 2023 wildfire — a population that is disproportionately low-income, renter-occupied, Native Hawaiian, Filipino American, and migrant worker. The equity design of this program specifically prioritizes: Native Hawaiian households (40% housing unit priority); households earning 30-80% AMI; workers displaced from Lahaina's tourism sector without homeownership equity; and food-insecure families in the recovery zone.

B. Generational Wealth Trust — Structural Equity Mechanism

The GWT model embedded in the 12 Stones Sovereign Charter is the most distinctive structural equity feature of this application. Every contract, subcontract, and public expenditure processed through MauiOS triggers an automatic 12.5% community distribution. On a $47.5M subrecipient program, this represents $5.9M in documented, blockchain-verified community wealth distributions over five years — flowing directly to Lahaina recovery households, not to contractors or intermediaries.

HUD's CDBG-DR program prioritizes programs that build long-term community wealth, not just temporary assistance. The GWT model operationalizes this priority in every contract dollar spent.

Section IX: Required Certifications and Attachments

Required Item	Status	Notes
SF-424 Application for Federal Assistance	**Pending**	Submit via HUD DRGR or GrantSolutions
SF-424A Budget Information	**Pending**	Based on Section VI budget tables above
Subrecipient Agreement (executed with Maui County)	**Required**	County must authorize 12 Stones as subrecipient under 24 CFR 570.500
MOU (all 5 parties)	**Drafted**	Ready for signature — initiate execution with County
12 Stones Sovereign Charter (SSC v5)	**Filed**	Filed Maui County Council April 14, 2025
Project Narrative (this document)	**Complete**	Finalize and attach as PDF
Environmental Review (NEPA)	**Required**	Housing and food security sites require site-specific ER before construction drawdowns
SAM.gov Active Registration (UEI)	**Required**	[Register / confirm active — required before any federal award]
Articles of Incorporation + IRS EIN	**Attach**	12 Stones Global Inc. corporate documents
Financial Statements (2 years) or Audit	**Required**	HUD requires documented financial capacity for subrecipients >$750K
Key Personnel Resumes	**In Preparation**	James Langford + all project leads
Partner MOUs / Letters of Commitment	**In Preparation**	FarmBox Foods, HCA, WFCF commitment letters

IMPORTANT: The path to CDBG-DR funding for 12 Stones runs through Maui County's Office of Recovery. The County is the HUD grantee. 12 Stones must be formally designated as a subrecipient under 24 CFR 570.500 before any drawdown is possible. Execute the MOU and subrecipient agreement as the first priority action.

Section X: Authorized Signature

I certify that the information contained in this subrecipient application is accurate and complete, that the organization named herein has the legal and organizational capacity to serve as a CDBG-DR subrecipient, and that all activities and expenditures will be conducted in full compliance with 24 CFR Part 570, HUD CDBG-DR program requirements, and applicable federal, state, and county regulations.

Field	Details
Name	**James Langford**
Title	CEO & Founder, 12 Stones Global Inc. / Langford Consulting Trust
Email	jimlangford@me.com
Charter Filed	12 Stones Sovereign Charter (SSC v5) — Maui County Council, April 14, 2025
Date	[DATE OF SUBMISSION]

Signature: _____ Date: _____

James Langford, CEO & Founder

12 Stones Global Inc. / Langford Consulting Trust

Maui, Hawaii

12 STONES GLOBAL INC.

MauiOS Platform — Full Deliverables Package

Scope of Work · Technical Specification · Grant Alignment Matrix · Design System Handoff

Prepared by	James Langford, CEO — 12 Stones Global Inc.
Contact	jimlangford@me.com
Date	April 2, 2026
Build Basis	Maui County Council April 2026 Agenda Priorities + FY2027 Budget ($1.616B)
Document Status	CONFIDENTIAL — For MOU Execution, Investor Distribution, Grant Filing
Version	1.0 — MauiOS Platform Deliverables Package
MOU Duration	24 months from signing — Governing Law: State of Hawaii

REGENERATE · EDUCATE · GOVERN · LIBERATE

Table of Contents

I. Executive Summary

This document constitutes the full MauiOS Deliverables Package for 12 Stones Global Inc. It formalizes the platform build sequence, technical specifications, grant alignment, and design system handoff for the MauiOS platform — the sovereign technology infrastructure for Maui County's regenerative economy.

The build order is derived directly from the Maui County Council's April 2026 legislative agenda and the $1.616 billion FY2027 budget proposed by Mayor Richard Bissen on March 25, 2026. Seven build phases are sequenced to mirror council committee priorities: WAI (Water), HLU (Housing), ADEPT (Agriculture), Full Council (Resilience), BFED (Economic), Charter (Compliance), and Kipuka (Public). This alignment ensures every deliverable maps to a fundable council priority and grant window.

Metric	Value
Platform	MauiOS — HyperFarmingEngine v4 (HFEv4) + RAIS + Sage Game + Kipuka Portal
Total Nodes	52 active + 2 Joker nodes (53: Emergency Override, 54: Sovereign Audit)
Build Cost	$640K–$960K (6 modules, 90-day sprint)
5-Year Revenue	$369.5M cumulative (Y1: $8.5M → Y5: $189M)
GWT Allocation	12.5% of all public contracts = $248.9M/yr at full deployment
Active Grant Pipeline	$1.75B+ (CDBG-DR, FEMA BRIC, USDA, DOE, State)
MOU Budget	$10M / 5-year across 5 parties
Charter Filed	April 14, 2025 — Maui County Council (28 Articles, SSC v5)
Readiness Score	58/100 — 18 core systems built; 5 SGI integration targets pending

II. Maui County Council — April 2026 Source Data

The following data is drawn from official Maui County government sources for April 2026, including the FY2027 Mayor's Proposed Budget currently under BFED Committee review, and active deliberations across WAI, HLU, and ADEPT committees.

April 2026 Council Schedule

Date	Body	Venue	Subject
Apr 1	BFED	Mayor H. Tavares Community Center, Makawao	FY2027 Budget — Upcountry residency hearing
Apr 2	BFED	Helene Hall, Hana	FY2027 Budget — Hana residency hearing
Apr 3	Full Council	Council Chamber, 8th Floor, Kalana O Maui	Full Council session (1st Friday of month)

Date	Body	Venue	Subject
Apr 14	BFED	Kihei Community Center	FY2027 Budget — South Maui residency hearing
Apr 15	BFED	Paia Community Center	FY2027 Budget — Makawao-Haiku-Paia residency hearing
Apr 17	Full Council + BFED	Council Chamber (6 PM)	Public hearing — FY2027 Budget (Wailuku + Kahului)
TBD April	WAI Committee	Council Chamber	Water infrastructure — $44.5M acquisitions review
TBD April	HLU Committee	Council Chamber	Affordable housing — $79.5M fund, 10 projects
TBD April	ADEPT Committee	Council Chamber	Agricultural tourism & food truck bills review

FY2027 Budget Priority Areas (Under April Review)

Priority Area	Allocation	Committee	Policy Driver
Affordable Housing Fund	$79.5M	HLU	600+ units, 10 projects, Maui + Lanai
Water Infrastructure	$44.5M	WAI	Private system acquisitions + 2 Upcountry wells
Lahaina Wildfire Recovery	CDBG-DR $1.639B	Full Council	HUD action plan, 88% undeployed
Resiliency Hubs	$400K (EMA)	Full Council	Emergency Management Agency deployment
Workforce Development	$200K (Maui Nui Resiliency Hui)	BFED	Economic diversification + Section 3
Agricultural Tourism	Bills advancing	ADEPT	Food trucks + agri-tourism in ag districts
Local Food Production	Grant funding	ADEPT	Community food security, farm recovery
Economic Diversification	Budget-wide	BFED	Education + research + sustainable economy
Environmental Stewardship	Budget-wide	ADEPT	Natural/cultural resource protection

Enacted Legislation in Effect — April 2026

Bill	Status	Effect	HFEv4 System
Bill 146, CD1 (2024)	In effect Jan 1, 2026	Water demand calculated by actual gallons-per-day (not outdated multipliers)	WeatherManager + F_PlantAttributes_Requirements

Bill	Status	Effect	HFEv4 System
Bill 180, CD1 (2024)	Enacted 2025	Private water catchment up to 30,000 gal exempt from building permits	BuildingSystem — deregulated catchment = FarmBox water nodes

III. Scope of Work — 7-Phase Build Order

The following seven phases are sequenced by council committee priority, grant deadline urgency, and MOU obligation. Each phase maps directly to one or more active Maui County legislative bodies and to specific UE5.7 HFEv4 systems within the 12 Stones platform.

PHASE 1 — WATER

Council Anchor	WAI Committee, $44.5M infrastructure, 2 Upcountry wells; Bill 146 demand-calculation reform active Jan 1, 2026
Deadline	WAI deliberations active April 2026; Bill 146 already enacted and in effect
Grant Hook	USDA Rural Dev CF ($5M rolling), DOE REAP ($10M Q2 2026)

Systems to Build / Wire:

1. Create DT_SGI_Nodes DataTable — populate all 52 rows from DT_Playable_Maps; flag water-dependent nodes in Zone 2 (Nodes 19-36) and Zone 1 wells (Nodes 1-18 Mauka)
2. Verify F_PlantAttributes_Requirements water field (gallons/day) matches Bill 146 actual-usage methodology
3. Wire AC_Biome_WeatherManager_Abstract water states to node water-availability attribute in AC_Attribute_Manager_Abstract
4. Create DT_SGI_Elements — register Element.Water interaction matrix with source/target multiplier for crop yield modeling
5. Add SGI Gameplay Tags to DefaultGameplayTags.ini: Element.Water, Node.Zone1.Mauka, Node.Zone2.Farmlands

Acceptance Criteria:

[ ] All 52 nodes populated in DT_SGI_Nodes with water demand float per row
[ ] WeatherManager water state change propagates to node attribute delta
[ ] Bill 146 demand formula documented in struct comment

PHASE 2 — HOUSING / LAND USE

Council Anchor	HLU Committee, $79.5M Affordable Housing Fund, 600+ units across 10 active projects, Maui + Lanai
Deadline	HLU reviewing projects throughout April; USDA 502 Direct active for Hua Lani ($3.4M)
Grant Hook	CDBG-DR $1.639B (HUD Lahaina), USDA 502 Direct ($3.4M active for Hua Lani)

Systems to Build / Wire:

1. Map DT_Buildables rows to FarmBox residential/mixed-use footprints; add E_Buildable_Categories entry for Residential_Affordable
2. Annotate DT_Playable_Maps rows with HousingProject_ID reference for 10 active county projects; flag Hua Lani (9.2 acres) as Zone 2 anchor node

3. Confirm BP_Buildable_Growing_Plot_Large/Medium/Small footprints match FarmBox container specs (20ft, 40ft ISO) for co-siting with affordable units
4. Begin WBP_SGI_NodeMap UI shell — 3-zone visual map with zone color coding: Mauka #166534, Farmlands #92400E, Makai #1E3A5F
5. Add SGI Gameplay Tags: Node.Zone3.Makai, GWT.Eligible, Encounter.Story

Acceptance Criteria:

[ ] Hua Lani node flagged as primary Zone 2 anchor in DT_SGI_Nodes
[ ] WBP_SGI_NodeMap renders all 3 zones with correct color tokens
[ ] GWT.Eligible tag fires when node has active housing + farming co-deployment

PHASE 3 — AGRICULTURE / FOOD PRODUCTION

Council Anchor	ADEPT Committee — agricultural tourism bills advancing; food truck allowances in ag districts; local food production grant (Councilmember Johnson)
Deadline	ADEPT advancing bills April 2026; USDA VAPG $500K application submitted March 31, 2026
Grant Hook	USDA VAPG $500K (submitted), Agriculture Grants $10.5M total (active), FarmBox deployment funding

Systems to Build / Wire:

1. Build AC_SGI_NodeManager component — wraps AC_Farming_Abstract + AC_Attribute_Manager_Abstract + RAIS event dispatch
2. Populate DT_Farmables with Maui-native crops + FarmBox production crops (greens, mushrooms, microgreens); add F_PlantAttributes_Seasons for Maui growing calendar
3. Map USDA VAPG 'change in production method' products to DT_CraftingRecipes (value-added mushroom extracts, dried herbs, seedling kits)
4. Wire AC_Inventory_SaveManager_Abstract save events to WFCF verification call stub; add E_Item_Categories entry for Certified_Agricultural
5. Add Encounter.Story tag to Zone 2 agri-tourism nodes; bind to NPCBehaviorSystem for visitor/stakeholder NPC simulation

Acceptance Criteria:

[ ] AC_SGI_NodeManager instantiated on at least one Zone 2 test node in Play-in-Editor
[ ] Crop data populated for 10+ Maui-native/FarmBox species in DT_Farmables
[ ] WFCF save hook fires on AC_Inventory_SaveManager_Abstract save event
[ ] VAPG value-added products represented in DT_CraftingRecipes

PHASE 4 — WILDFIRE RECOVERY / RESILIENCE

Council Anchor	Full Council — Lahaina recovery (CDBG-DR $1.639B, 88% undeployed); resiliency hubs ($400K EMA); FEMA BRIC application ($25M, Q2 2026)
Deadline	FEMA BRIC Q2 2026 is the near-term funding deadline — Node 53 fire modeling is the required deliverable
Grant Hook	FEMA BRIC $25M (Q2 2026), CDBG-DR $1.639B (ongoing), EMA resiliency hub allocation $400K

Systems to Build / Wire:

1. Wire WeatherManager biome state FireWeather/Drought to Node 53 EventTrigger dispatch; activate RAIS.Scroll.Triggered tag on threshold breach
2. Bind AC_Actor_Health health value to community resilience score per node; threshold breach activates GWT.Eligible review flag
3. Add HTTP subsystem call placeholder in WeatherManager for FARSITE fire spread API (FEMA BRIC deliverable)
4. Wire actor state changes to RAIS scroll activation; implement GHOST-001 (fraud) and AUDIT-001 (financial anomaly) scroll trigger conditions
5. Add SGI Gameplay Tags: Node.Joker.Emergency, Node.Joker.Audit, RAIS.Scroll.Active, RAIS.Scroll.Triggered

Acceptance Criteria:

[ ] WeatherManager FireWeather state triggers Node 53 emergency event in Play-in-Editor
[ ] RAIS scroll GHOST-001 activates on attribute anomaly threshold
[ ] FARSITE HTTP stub documented with FEMA BRIC deliverable comment
[ ] Node 53 gameplay tag registered and queryable via tag container

PHASE 5 — ECONOMIC DIVERSIFICATION / GWT LAYER

Council Anchor	BFED Committee — Maui Nui Resiliency Hui ($200K workforce); economic opportunity budget-wide; Section 3 workforce compliance (CDBG-DR federal requirement)
Deadline	CDBG-DR Section 3 workforce compliance is a federal audit requirement; GWT 12.5% = $248.9M/yr at full deployment
Grant Hook	GWT 12.5% of public contracts (Charter-mandated), CDBG-DR Section 3, Tier 2 Municipal license $640K-$960K/yr

Systems to Build / Wire:

1. Populate DT_SGI_Cards (52-card crosswalk) in RoguelikeDeckbuilder keyed to DT_Cards RowNames; map ECardType to node action type and ECardRarity to node priority tier
2. Bind DeckBuilderSubsystem RunSeed to RAIS session ID; bind CurrentFloor to HFEv4 deployment phase; bind EAscensionLevel to license tier (Normal/Ascension 1-3 = Tier 1/2/3)
3. Scaffold GWT disbursement interface in Vendor system; add GWT.Funded and GWT.Distributed tag lifecycle
4. Configure OnlineMultiplayer replication framework for concurrent county admin + community monitor access (MauiOS Tier 2 use case)

5. Populate DT_Ability_State_Effects with capabilities per node type (produce, report, alert, transact, certify, educate)

Acceptance Criteria:

[ ] All 52 cards in DT_SGI_Cards populated with node crosswalk data
[ ] EAscensionLevel maps correctly to license tier in DeckBuilderSubsystem
[ ] GWT lifecycle tags (Eligible / Funded / Distributed) register and transition correctly
[ ] At least 6 distinct node capability types defined in DT_Ability_State_Effects

PHASE 6 — RAIS BLOCKCHAIN PERSISTENCE

Council Anchor	12 Stones Sovereign Charter filed April 14, 2025 with Maui County Council — 20+ active enforcement scrolls; COMPLIANCE-001 federal trigger active
Deadline	MOU 24-month compliance clock running; federal grant audit requirements active across all $1.75B+ grant pipeline
Grant Hook	All active grants ($1.75B pipeline) require audit-grade blockchain-sealed compliance documentation

Systems to Build / Wire:

1. Implement AC_SGI_RAISBridge component — wire AttemptSaveGame/AttemptLoadGame events in BP_GI_RLDB to FastAPI HTTP call to immudb seal
2. Wire AC_Inventory_SaveManager_Abstract save trigger as blockchain sealing event; tag sealed entries with RAIS.Scroll.Sealed
3. Implement DT_SGI_BaguaEffects — 8-trigram effect table bound to governance event hooks (Qian=sovereignty, Kun=land, Zhen=action, Li=transparency, Kan=water, Xun=agriculture, Gen=structure, Dui=community)
4. Implement DT_SGI_Kumulipo — 16-wa emergence scheduler mapping Po 1-8 to build phases 1-6 and Ao 9-16 to MauiOS deployment
5. Add LegislationToCharter function stub in Charter logic layer — Legistar legislative scroll to Charter Article mapping

Acceptance Criteria:

[ ] AC_SGI_RAISBridge fires HTTP call on save event (stub/mock acceptable for initial milestone)
[ ] RAIS.Scroll.Sealed tag applied after successful immudb response
[ ] All 16 Kumulipo wa registered as gameplay tags
[ ] All 8 Bagua trigrams registered with event hook documentation

PHASE 7 — MAUI OS DASHBOARD / PUBLIC-FACING LAYER

Council Anchor	Kipuka Access Portal (Module 4, $115K-$150K); public testimony integration; 2026 State Legislative Session tracking; CDBG-DR community engagement compliance

Deadline	Public hearings active throughout April 2026 — testimony engine is a live operational need
Grant Hook	CDBG-DR community engagement compliance requirement; Kipuka portal Tier 1 HomePod license ($33K-$95K/unit)

Systems to Build / Wire:

1. Complete WBP_SGI_NodeMap — 52-node zone map with zone color coding, node vitality overlays from ActorHealth, water status from WeatherManager
2. Build WBP_SGI_RAISDashboard — real-time scroll status panel (Active / Sealed / Triggered) per scroll; GWT fund allocation live view
3. Wire Information_Prompt, Outliner_System, World_Dialogue to node selection events — each node click surfaces production, resilience, WFCF chain, GWT status
4. Adapt MainMenu + SessionManager for Kipuka public testimony session — date/time stamp, RAIS scroll creation, immudb seal on submission
5. Activate Playwright Legistar scraper — feeds new Maui County Council agenda items to RAIS as draft scrolls for Kananae Compiler processing

Acceptance Criteria:

[ ] WBP_SGI_NodeMap renders all 52 nodes with correct zone colors and live health overlay
[ ] Node selection surfaces production, resilience, WFCF, and GWT data in Information_Prompt
[ ] Testimony session creates RAIS scroll entry sealed in immudb
[ ] Legistar feed populates at least one draft scroll from a real April 2026 agenda item

IV. Technical Specification

Platform Architecture

Layer	Technology	Role
Game / 3D Engine	Unreal Engine 5.7	HFEv4 digital twin, 52-node simulation, Sage Game, UI
AI	Claude Sonnet 4.6, O3-Mini High, GPT-4.5 (UE5)	RAIS reasoning, Charter logic, gameplay AI
Backend API	FastAPI (Python)	RAIS bridge, IoT ingestion, grant pipeline APIs
Database	PostgreSQL + pgvector, TimescaleDB	ACID transactions, vector embeddings, time-series IoT
Blockchain	immudb + WFCF smart contracts	Immutable scroll sealing, agricultural traceability
IoT	AWS IoT Core, MQTT broker	52-node sensor mesh, real-time telemetry
Search	Elasticsearch + pgvector (1536-dim)	RAIS scroll retrieval, council agenda indexing
Scraping	Playwright	Legistar council agenda auto-processing
Storage	Cloudflare R2, PostgreSQL	Blob storage, zero egress cost, ACID persistence
Cache	Redis, Redpanda/Kafka	Real-time event streaming, session state
CI/CD	Docker, Fly.io, Railway, GitHub Actions	Automated deployment pipeline

6 Software Modules

Module	Name	Budget	Description
1	Makaukau Protocol	$220K–$330K	UE5 + GPT-4.5 Charter logic + glyph system + sovereign scrolls
2	Ike Kumu Stack	$90K–$145K	Data compliance backend, RAIS engine, immudb integration
3	Pa Kii Framework	$60K–$90K	Visual/ceremonial UX, zone map, node display layer
4	Kipuka Access Portal	$115K–$150K	Mobile app + community testimony engine + public dashboard
5	Onaehana Holo	$80K–$120K	WFCF blockchain traceability + grant stacking pipeline
6	Kanaenae Compiler	$75K–$125K	Legal scroll sealing — legislation to Charter Article mapping
TOTAL		$640K–$960K	90-day sprint — 6 parallel modules

Design System — Component Completeness Scorecard

System	Built	SGI Wired	Council Priority	Build Phase	Score
FarmingSystem	YES	Partial	Agriculture / ADEPT	Phase 3	7/10
WeatherManager	YES	NO	Water / Wildfire — WAI + FEMA	Phase 1, 4	6/10
BuildingSystem	YES	NO	Housing — HLU	Phase 2	6/10
MapManagement	YES	NO	52-node topology	Phase 1, 2	6/10
AttributeSystem	YES	NO	Node KPIs — RAIS	Phase 3, 5	6/10
Inventory / WFCF	YES	NO	Supply chain traceability	Phase 3	6/10
Crafting / VAPG	YES	NO	Agriculture value-add	Phase 3	5/10
ActorHealth	YES	NO	Resilience index	Phase 4	4/10
EventTrigger	YES	NO	RAIS compliance engine	Phase 4, 6	4/10
AbilitySystem	YES	NO	Node capabilities	Phase 5	4/10
OnlineMultiplayer	YES	NO	Governance — MauiOS	Phase 5	4/10
Save / immudb	YES	NO	Blockchain sealing	Phase 6	4/10
UI / MauiOS	YES	NO	Dashboard layer	Phase 7	4/10
DT_SGI_Nodes	NOT STARTED	—	ALL PHASES (unblocking)	Phase 1	0/10
AC_SGI_NodeManager	NOT STARTED	—	Agriculture + Water	Phase 3	0/10
AC_SGI_RAISBridge	NOT STARTED	—	Compliance + Charter	Phase 6	0/10
WBP_SGI_NodeMap	NOT STARTED	—	Public dashboard	Phase 7	0/10

Overall Readiness: 58/100 — All 18 core UE5.7 systems built; 0 of 5 SGI integration targets started. Creating DT_SGI_Nodes (52 rows) is the single highest-leverage unblocking action.

V. Grant Alignment Matrix

Each build phase is tied to one or more active grant opportunities. The table below maps phases to grant programs, dollar amounts, deadlines, and the specific platform deliverable that satisfies the grant's primary requirement.

Grant / Program	Amount	Deadline	Phase	Platform Deliverable Required
USDA VAPG	$500K	Mar 31, 2026 (SUBMITTED)	Phase 3	FarmBox value-added production catalog in DT_CraftingRecipes
DOE REAP (Solar)	$10M	Q2 2026	Phase 1	Water + energy node data in DT_SGI_Nodes
FEMA BRIC	$25M	Q2 2026	Phase 4	Node 53 fire modeling + FARSITE API stub + WeatherManager fire state
USDA Rural Dev CF	$5M	Rolling	Phase 1, 3	Water infrastructure data + food production output reporting
CDBG-DR (HUD)	$1.639B	Ongoing	Phase 4, 7	Community engagement portal (Kipuka) + Lahaina recovery node data
USDA RBDG	$500K	Q3 2026	Phase 3	Agricultural business development outputs from FarmingSystem
USDA 502 Direct (Hua Lani)	$3.4M	Active	Phase 2	Hua Lani node in DT_SGI_Nodes + BuildingSystem co-site confirmation
State / Foundation Grants	~$5.25M	Rolling	All phases	RAIS compliance documentation + quarterly impact reports
MauiOS Tier 2 License	$640K–$960K/yr	MOU execution	Phase 5, 7	Full 52-node HFE + MauiOS dashboard + RAIS + Sage Game deployed
GWT (12.5% public contracts)	$248.9M/yr at full	Charter-mandated	Phase 5, 6	GWT.Eligible node lifecycle + immudb disbursement record

VI. MOU Compliance Map

All deliverables in this package align with MOU obligations under the 12 Stones Project MOU (24-month duration, governing law: State of Hawaii). The following table cross-references MOU phases against platform build phases and mandatory deliverables.

Five MOU Phases vs. Seven Build Phases

MOU Phase	Timeline	MOU Requirement	Platform Build Phase(s)
Phase 1: Preliminary Analysis	Month 0-1	Land survey, env. impact assessment, software integration phase 1, digital twin setup phase 1	Platform Phase 1 (Water/Nodes) + Phase 2 (Housing/Map)
Phase 2: Planning & Design	Month 1-3	Architectural planning, budgeting, software integration phase 2, digital twin phase 2	Platform Phase 3 (Agriculture) + Phase 4 (Resilience)
Phase 3: Implementation	Month 3-18	FarmBox deploy, infrastructure, software integration phase 3, digital twin phase 3	Platform Phase 5 (GWT) + Phase 6 (RAIS)
Phase 4: Testing & QA	Month 18-21	WFCF quality audits, software security checks, digital twin final verification	Platform Phase 6 (RAIS) — audit layer validation
Phase 5: Handover & Closure	Month 21-24	Occupancy approval, software documentation, digital twin handover to County	Platform Phase 7 (Dashboard) — Kipuka portal go-live

Mandatory Deliverables

Deliverable	Frequency	Platform Source	Status
Quarterly Progress Reports	4x / year	RAIS dashboard — milestone + financial data	Phase 6 milestone
Annual Impact Assessments	1x / year	ActorHealth resilience index + FarmingSystem output	Phase 3 + 4 milestone
Final Comprehensive Report (Year 5)	Once — 2031	Full 52-node digital twin export	Phase 7 long-term
USDA VAPG Application	One-time	DT_CraftingRecipes + FarmBox production data	SUBMITTED Mar 31, 2026
Section 3 Workforce Report (CDBG-DR)	Quarterly	NPCBehaviorSystem workforce simulation data	Phase 3 milestone

Intellectual Property Boundaries

Owner	What They Own
12 Stones Global Inc.	Software platform (MauiOS, RAIS, Sage Game, all 6 modules), Digital Twin technology, IoT mesh architecture, algorithms, glyph system, Sovereign Charter (SSC v5)
Maui County	Underlying data generated by platform, public records, legislation outputs, audit results filed under Hawaii law

VII. Open Questions for Design Review

The following decisions require review before coding proceeds to avoid costly struct refactors in UE5.7:

#	Question	Recommendation	Decision Needed By
1	Should DT_SGI_Nodes include a water demand float field (gallons/day per Bill 146), or derive dynamically from F_PlantAttributes_Requirements at runtime?	Static field in DataTable — faster lookup, easier for county audit reporting	Phase 1 start
2	Which specific node row in DT_Playable_Maps should Hua Lani (9.2 acres) map to?	Node 19 — Zone 2 Farmlands anchor entry point	Phase 2 start
3	Should the FARSITE fire spread API call route through WeatherManager (UE5 HTTP) or through FastAPI backend?	FastAPI backend — enables rate limiting, caching, and audit logging for FEMA BRIC compliance	Phase 4 start
4	At what ActorHealth resilience score does a node become GWT.Eligible?	>=65/100 — requires Charter Article review to formalize	Phase 5 start
5	Should the Legistar scraper run on nightly cron (GitHub Actions) or trigger on RAIS session start?	Nightly cron for production; session-trigger for demo mode	Phase 7 start
6	What constitutes a testimony submission (vs. draft) for immudb sealing purposes?	Explicit 'Submit' action by authenticated user — not auto-save	Phase 7 start

VIII. Priority Actions — Ranked by Council Urgency

Rank	Action	Council Basis	Timeline
1	Create DT_SGI_Nodes DataTable (52 rows from DT_Playable_Maps)	WAI water infrastructure + HLU housing — both require node data	This week
2	Register all SGI Gameplay Tags in DefaultGameplayTags.ini	Zero cost; unlocks all tag-gated systems across all 7 phases	Same session as #1
3	Wire WeatherManager to Node 53 Emergency Override	FEMA BRIC Q2 2026 — fire modeling is the primary grant deliverable	Within 30 days
4	Build AC_SGI_NodeManager	Core integration class — unblocks Phases 3, 4, 5	Within 45 days
5	Implement AC_SGI_RAISBridge to immudb	All $1.75B in active grants require audit-grade compliance sealing	Within 60 days
6	Build WBP_SGI_NodeMap public dashboard	CDBG-DR community engagement compliance + Kipuka portal go-live	Within 90 days
7	Activate Legistar scraper to RAIS pipeline	Makes every future council agenda auto-feed the compliance engine	90-day milestone

12 Stones Global Inc. · jimlangford@me.com · Maui, Hawaii

REGENERATE · EDUCATE · GOVERN · LIBERATE

CONFIDENTIAL — For MOU Execution, Investor Distribution, and Grant Filing Only

12 STONES GLOBAL INC.

MauiOS — Government Architecture

How the Platform Serves All Three Branches of Government, Manages County Operations, and Interfaces with Hawaii Revised Statutes (HRS)

Prepared by	James Langford, CEO — 12 Stones Global Inc.
Date	April 2, 2026
Platform	MauiOS + HFEv4 + RAIS + Sage Game + Kipuka Portal + Kananae Compiler
Jurisdiction	County of Maui — Second Circuit — State of Hawaii — Federal Interface
Charter	12 Stones Sovereign Charter SSC v5 (28 Articles) filed April 14, 2025
Legal Basis	Maui County Charter, HRS Title 6 Ch. 46, HRS Title 8 Ch. 91-92, HRS Ch. 174C, 205, 603, 604, HRS Ch. 27G
Classification	CONFIDENTIAL — For MOU Execution, Grant Filing, and Investor Distribution

REGENERATE · EDUCATE · GOVERN · LIBERATE

Table of Contents

Table of Contents 2
I. Platform Overview — One System, Three Branches 3
The 52-Node Government Topology 3
Constitutional Authority 5
County Departments Served by MauiOS 5
Digital Twin Budget Management (MOU Section 7.1) 6
Constitutional Authority 7
Council Committees and MauiOS Coverage 7
Legislative Processing Pipeline — How Every Bill Becomes a Node Event 7
April 2026 Active Bills — Node Impact Map 8
Sunshine Law Compliance (HRS Chapter 92) 8
Constitutional Authority 10
Court Jurisdiction Relevant to MauiOS 10
MOU Dispute Resolution Protocol 10
Node 54 — Sovereign Audit (Judicial Interface) 11
V. Hawaii Revised Statutes (HRS) — Full Compliance Matrix 12
County Administration (HRS Title 6) 12
Open Government (HRS Title 8) 12
Natural Resources, Water, and Agriculture (HRS Title 12-13) 13
Fiscal, Tax, and Finance (HRS Titles 14-17) 13
Judiciary (HRS Title 33) 14
VI. State Government Interface 15
State Agency Interfaces 15
Hawaii State Plan Alignment (HRS Chapter 226) 16
VII. Full Government Capability Map 17
VIII. Architectural Integrity — Why This Works 19
The Sovereign Charter as Constitutional Layer 19
The Platform's Five Foundational Guarantees to Government 19
Sources and Legal References 21

I. Platform Overview — One System, Three Branches

MauiOS is not a single application. It is a sovereign technology infrastructure that simultaneously serves all three branches of county government, interfaces with state-level Hawaii Revised Statutes, and provides a federally auditable compliance backbone for every active grant program. The platform was designed from first principles around the 52-node HFEv4 topology — a dual-purpose governance and simulation framework — with each node corresponding to a specific community, district, or functional zone within Maui County.

The 12 Stones Sovereign Charter (SSC v5, 28 Articles, filed April 14, 2025 with the Maui County Council) is the constitutional layer that gives the platform its legal standing. The Kananae Compiler translates between the Maui County Charter, the 12 Stones Sovereign Charter, and Hawaii Revised Statutes — creating a living legislative cross-reference that automatically updates as laws change. Every council bill, mayoral order, court filing, and HRS amendment that touches the county's operations is processed through this system.

Branch	MauiOS Interface	Primary HFEv4 System	Primary HRS Authority
Executive (Mayor + Departments)	Digital Twin budget dashboard, IoT node data, grant pipeline, RAIS compliance alerts to Corporation Counsel	WeatherManager + ActorHealth + AttributeSystem + Vendor (GWT)	HRS Ch. 46, Maui County Charter Article IV
Legislative (County Council, 9 members)	Legistar scraper + Kananae Compiler + Kipuka public testimony + bill-to-node impact mapping	DayNightCycle (RAIS timestamps) + MapManagement + UI dashboard	HRS Ch. 92 Sunshine Law, HRS Ch. 91 Admin Procedure, Maui Charter Art. II
Judicial (Second Circuit + District Court)	immudb sealed scroll evidence + RAIS compliance chain + MOU dispute resolution + Corporation Counsel interface	Save/RAIS bridge + EventTrigger + AC_SGI_RAISBridge	HRS Ch. 603 Circuit Courts, Ch. 604 District Courts, MOU dispute resolution protocol
State HRS Interface	Kananae Compiler statutory cross-reference + agency API stubs + OETS data governance alignment	Kananae Compiler + DT_SGI_Kumulipo legislative stages	HRS Ch. 27G Digital Gov, Ch. 174C Water, Ch. 205 Land Use, Ch. 235/237/246 Fiscal
Federal Interface	Grant pipeline dashboard + CDBG-DR community engagement + FEMA BRIC fire modeling + Section 3 workforce	Node 53 Emergency Override + WBP_SGI_NodeMap public portal	HUD CDBG-DR, FEMA BRIC, USDA Rural Dev, DOE REAP, FinCEN/BSA

The 52-Node Government Topology

Every node in the HFEv4 system maps to a specific governmental function as well as a physical community zone. The three zones mirror the three primary geographies of Maui governance:

Zone	Nodes	Color	Government Domain	Primary Branches Served
Zone 1 — Mauka (Upland/Governance)	1-18	#166534 Green	Charter authority, water infrastructure, governance oversight, emergency management	Executive + Judicial
Zone 2 — Farmlands (Agriculture/Culture)	19-36	#92400E Amber	Agricultural production, housing, cultural preservation, WFCF traceability, education	Legislative (ADEPT, HLU)

Zone	Nodes	Color	Government Domain	Primary Branches Served
Zone 3 — Makai (Ocean/Revenue/Franchise)	37-52	#1E3A5F Navy	Economic development, franchise licensing, GWT distribution, municipal revenue	Executive + BFED
Node 53 — Emergency Override (Joker)	53	#DC2626 Red	FEMA BRIC fire modeling, emergency protocol, HRS Ch. 127A Emergency Management	Executive Emergency Powers
Node 54 — Sovereign Audit (Joker)	54	#7C3AED Purple	Full-system financial audit, Charter compliance review, immudb verification sweep	Judicial + Corporation Counsel

II. EXECUTIVE BRANCH

Mayor Richard T. Bissen Jr. | Maui County Charter Article IV | HRS Chapter 46 | $1.616B FY2027 Budget

Constitutional Authority

The executive power of the County of Maui is vested in and exercised by the Mayor (Maui County Charter Article IV). The Mayor serves as Chief Executive Officer with authority over all county departments, budget administration, emergency powers (HRS Chapter 127A), and appointment of the Corporation Counsel. HRS Chapter 46-1.5 enumerates the general powers of counties including zoning enforcement, tax exemptions for housing and agriculture, and regulatory authority.

County Departments Served by MauiOS

Department	MauiOS Function	HFEv4 System	HRS / Charter Authority
Department of Finance	GWT 12.5% allocation tracking, real property tax data per node, grant accounting, quarterly financial reports	Vendor (GWT) + AttributeSystem (KPIs)	HRS Ch. 246 (Real Property Tax), Ch. 40 (Public Finance), Charter Art. XIV
Department of Planning	Land use compliance per HRS 205, agricultural district node classification, permit tracking	MapManagement + BuildingSystem	HRS Ch. 205 (Land Use Commission), Ch. 46-4 (County Zoning)
Department of Water Supply	Bill 146 water demand data (actual gallons/day per node), well monitoring, water allocation per Zone 1	WeatherManager + DT_SGI_Elements (Water)	HRS Ch. 174C (State Water Code), Bill 146 CD1
Department of Public Works	Infrastructure node status, FarmBox co-siting approvals, resiliency hub locations	BuildingSystem + ActorHealth	HRS Ch. 46, County Charter Art. IV
Office of Recovery (Lahaina)	CDBG-DR node tracking, Lahaina recovery progress, 88% undeployed funds pipeline	WBP_SGI_NodeMap + RAIS dashboard	HUD CDBG-DR Action Plan, Charter Art. IV
Corporation Counsel	RAIS compliance alerts, GHOST-001 fraud detection, AUDIT-001 financial anomaly, legal scroll evidence	EventTrigger + AC_SGI_RAISBridge + immudb	HRS Ch. 84 (Ethics), Charter Art. XIII, MOU dispute terms
Emergency Management Agency	Node 53 fire/emergency activation, FEMA BRIC fire modeling (FARSITE), resiliency hub deployment	Node 53 Emergency Override + WeatherManager	HRS Ch. 127A (Emergency Management), FEMA BRIC
Environmental Management	Environmental monitoring per node, biome state data, invasive species alerts	WeatherManager + EventTrigger	HRS Ch. 205 (conservation), DLNR interface
Department of Housing	Affordable housing fund ($79.5M) node tracking, Hua Lani co-site data, 600+ unit pipeline	BuildingSystem + GWT.Eligible tags	HRS Ch. 46 (housing exemptions), Charter Affordable Housing Fund
Parks and Recreation	Community node activity data, public space utilization metrics	NPCBehaviorSystem + UI	Charter Art. IV, County ordinances

Digital Twin Budget Management (MOU Section 7.1)

The MauiOS Digital Twin provides the Mayor's office with real-time visibility into every dollar of the $1.616B FY2027 budget. The platform tracks: expenditure vs. budget by department and node, variance analysis with automated alerts, grant fund deployment status across all 14+ active programs, and milestone-based reallocation triggers. Every financial event is sealed in immudb creating an immutable audit chain compliant with federal grant oversight requirements.

Budget Category	FY2027 Amount	Node Zone	MauiOS Tracking Method
Affordable Housing Fund	$79.5M	Zone 2 (Nodes 19-36)	BuildingSystem node co-site data + GWT.Eligible lifecycle
Water Infrastructure	$44.5M	Zone 1 (Nodes 1-18)	WeatherManager water states + Bill 146 demand data
CDBG-DR Lahaina Recovery	$1.639B total (88% undeployed)	Zone 2-3 (Lahaina nodes)	RAIS grant pipeline + WBP_SGI_NodeMap Lahaina overlay
Resiliency Hubs (EMA)	$400K	Node 53 + Zone 1	Node 53 emergency activation + ActorHealth resilience index
Workforce Development (Maui Nui Resiliency Hui)	$200K	Zone 2-3	NPCBehaviorSystem Section 3 simulation + BFED reporting
GWT Trust (12.5% mandate)	$248.9M/yr at full deployment	All 52 nodes	Vendor GWT lifecycle + immudb disbursement seal
Real Property Tax Revenue	$650.8M	All zones	DT_SGI_Nodes per-node tax assessment data
Capital Improvement Program	$461.4M	Infrastructure nodes	BuildingSystem + EventTrigger project milestone alerts

III. LEGISLATIVE BRANCH

Maui County Council — 9 Members, 9 Residency Areas | Charter Article II | HRS Chapter 92 Sunshine Law

Constitutional Authority

The Maui County Council is the legislative and policy-making body of county government (Maui County Charter Section 2-2). Nine members are elected at-large, one per residency area, serving two-year terms. The Council enacts ordinances, adopts the county budget, confirms mayoral appointments, and exercises oversight over all county departments. Hawaii HRS Chapter 92 (Sunshine Law) requires all board meetings to be open to the public, with proper notice and public testimony provisions.

Council Committees and MauiOS Coverage

Committee	Acronym	Jurisdiction	MauiOS Function	HFEv4 System
Budget, Finance & Economic Development	BFED	Annual budget, fiscal policy, economic programs, GWT oversight	Budget dashboard + GWT allocation tracker + grant pipeline + FY2027 $1.616B review	Vendor + AttributeSystem + DayNightCycle
Water and Infrastructure	WAI	Water rights, utility infrastructure, county facilities	Bill 146 water demand data per node, well monitoring, $44.5M infrastructure tracking	WeatherManager + DT_SGI_Elements
Housing and Land Use	HLU	Zoning, affordable housing, land use permits, Hua Lani	BuildingSystem node co-site data, $79.5M housing fund, 10 project pipeline	BuildingSystem + MapManagement
Agriculture, Diversification, Environment & Public Transportation	ADEPT	Farming, ag tourism, food production, environmental protection	FarmingSystem + DT_Farmables crops, WFCF traceability, ag tourism encounter nodes	FarmingSystem + Crafting + Inventory
Public Safety, Land & Ocean Use	PSLU	Police, fire, ocean recreation, public safety	ActorHealth community resilience + Node 53 emergency + WeatherManager safety states	ActorHealth + Node 53
Government Operations, Technology & Intergovernmental Relations	GOTIR	IT policy, county technology, intergovernmental affairs	MauiOS OETS alignment, HRS Ch. 27G data governance, county tech compliance	OnlineMultiplayer + RAIS

Legislative Processing Pipeline — How Every Bill Becomes a Node Event

When a bill is introduced in the Maui County Council, MauiOS processes it through four automated stages:

Stage	Action	MauiOS Component	HRS Compliance
1. Introduction	Legistar scraper detects new bill from capitol.hawaii.gov or legistar.com; creates draft RAIS scroll	Playwright scraper + DayNightCycle timestamp	HRS 92-7 agenda notice requirement

Stage	Action	MauiOS Component	HRS Compliance
2. Committee Review	Draft scroll tagged by committee (BFED/WAI/HLU/ADEPT); affected nodes flagged in DT_SGI_Nodes; community impact modeling runs	MapManagement + AttributeSystem + EventTrigger	HRS 92-3 open meeting requirement
3. Public Testimony	Kipuka portal opens testimony window; resident voice recorded with date/time stamp; sealed in immudb on submission	Kipuka (MainMenu + SessionManager) + AC_SGI_RAISBridge + immudb	HRS 92-3.5 testimony rights, HRS 91-2 hearing notice
4. Passage / Kananae Compile	Enrolled bill run through Kananae Compiler; maps to Charter Article + HRS chapter; affected node rules updated in DT_SGI_Nodes	Kananae Compiler + DT_SGI_Nodes update	HRS 46-1 county ordinance authority

April 2026 Active Bills — Node Impact Map

Bill / Action	Committee	Nodes Affected	Rule Update Required
Bill 146 CD1 — Water demand reform (IN EFFECT Jan 1, 2026)	WAI	All Zone 1 + Zone 2 water nodes	F_PlantAttributes_Requirements: water demand field = actual gallons/day
Bill 180 CD1 — Catchment permit exemption (ENACTED)	WAI	Zone 2 FarmBox co-sites	BuildingSystem: Residential_Catchment category deregulated up to 30,000 gal
Agricultural tourism bills (ADVANCING)	ADEPT	Zone 2 Nodes 19-36	Encounter.AgroTourism tag activated on applicable Zone 2 nodes
FY2027 Budget $1.616B (UNDER REVIEW April 2026)	BFED	All 52 nodes	AttributeSystem budget allocation per node updated on enactment
$79.5M Affordable Housing Fund (UNDER REVIEW)	HLU	Zone 2 (Hua Lani anchor)	BuildingSystem: Residential_Affordable category budget ceiling updated
$44.5M Water Infrastructure (UNDER REVIEW)	WAI	Zone 1 Mauka wells	WeatherManager: 2 new well nodes added to Zone 1 DT_SGI_Nodes

Sunshine Law Compliance (HRS Chapter 92)

HRS Chapter 92 requires every public board meeting to be open, noticed, and recorded. MauiOS automates this compliance layer:

HRS 92 Requirement	MauiOS Compliance Mechanism
92-7: 6-day advance notice with agenda	Legistar scraper auto-detects posted agendas; RAIS creates draft scroll; notification dispatched to subscribed nodes
92-3: Open meeting requirement	All council meeting streams timestamped in RAIS; public testimony window opens in Kipuka portal
92-3.5: Public testimony rights	Kipuka portal collects testimony; each submission sealed in immudb with RAIS.Scroll.Sealed tag; immutable public record
92-9: Minutes requirement	RAIS auto-archives meeting minutes as sealed scrolls; available via WBP_SGI_RAISDashboard

HRS 92 Requirement	MauiOS Compliance Mechanism
92-5: Executive session exceptions	Sessions tagged as RAIS.Scroll.Restricted; available to Corporation Counsel only via authenticated admin portal
OIP enforcement	AUDIT-001 scroll triggers if any board meeting proceeds without proper RAIS pre-filing; alert to Corporation Counsel

IV. JUDICIAL BRANCH

Hawaii State Judiciary — Second Circuit (Maui) | HRS Chapter 603 + 604 | Wailuku Courthouse, Hoapili Hale

Constitutional Authority

The judicial function at the county level is exercised by the Hawaii State Judiciary's Second Circuit, which encompasses the islands of Maui, Molokai, Lanai, Kahoolawe, and Molokini. The Second Circuit maintains four Circuit Court judges at the Wailuku Courthouse (2145 Main Street). Under HRS Chapter 603, the Circuit Court has general jurisdiction over civil and criminal matters. Under HRS Chapter 604, District Courts handle violations of county ordinances, civil claims under $40,000, misdemeanors, and landlord-tenant actions.

Maui County does not have an independent judicial branch at the charter level. Instead, the County interfaces with the State Judiciary through the Corporation Counsel (legal officer, appointed by Mayor with Council approval), and through the MOU dispute resolution protocol, which provides a structured path from internal resolution to binding arbitration before any matter reaches the courts.

Court Jurisdiction Relevant to MauiOS

Court	Jurisdiction	MauiOS Interface	HRS Authority
Second Circuit Court	General civil + criminal jurisdiction, appellate review of district court, jury trials, environmental enforcement	immudb sealed RAIS scrolls as evidence; GHOST-001 fraud scroll triggers referral to Corp Counsel	HRS Ch. 603
Second Circuit District Court	County ordinance violations, civil claims <=$40K, misdemeanors, landlord-tenant, TROs	RAIS compliance chain documenting ordinance breach history; sealed testimony via Kipuka	HRS Ch. 604
Second Circuit Family Court	Domestic relations, juvenile, guardianship	Community node health data (NPCBehaviorSystem) can inform social service referrals	HRS Ch. 571
Office of Disciplinary Counsel	Attorney misconduct, ethics	Corporation Counsel RAIS alert feed; HRS Ch. 84 ethics compliance data	HRS Ch. 84 Ethics

MOU Dispute Resolution Protocol

The 12 Stones MOU with Maui County establishes a three-tier dispute resolution protocol that keeps most matters out of the court system entirely. MauiOS manages all three tiers:

Tier	Process	Timeline	MauiOS Role	Outcome
Tier 1	Direct Negotiation between parties	30 days	RAIS surfaces relevant scrolls; both parties access dispute dashboard; Kipuka resident voice activated if community-affecting	Resolved internally — RAIS scroll sealed as settlement record
Tier 2	Facilitated Mediation (resident voice via software per MOU Section	60 days	Mediation sessions timestamped in RAIS; all submissions sealed in immudb; node impact modeling shows community effect	Mediation settlement sealed in immudb; precedent scroll created

Tier	Process	Timeline	MauiOS Role	Outcome
	9.4)			
Tier 3	Binding Arbitration (State of Hawaii governing law)	Per arbitration rules	Complete immudb audit trail provided as evidence; RAIS compliance history; all sealed scrolls admissible as business records	Arbitration award enforceable under Hawaii law
Escalation	Court referral (HRS Ch. 603/604 Second Circuit)	Court schedule	Full immudb chain of custody; RAIS scroll export for discovery; GHOST-001 and AUDIT-001 scroll flags inform legal strategy	Litigation with complete platform audit trail

Node 54 — Sovereign Audit (Judicial Interface)

Node 54 is the platform's Judicial Joker node. When activated, it triggers a full-system sovereign audit sweep: all 52 active nodes are scanned for compliance with the Charter, active HRS requirements, and MOU obligations. The audit output is a sealed immudb document admissible in any Hawaii court as a business record under HRS Chapter 622A (Uniform Electronic Transactions Act) and UETA.

Scroll Activated by Node 54	What It Audits	Legal Trigger
AUDIT-001 (Financial Anomaly)	GWT disbursements, grant accounting, budget vs. actual variance	Corporation Counsel referral or BFED committee request
GHOST-001 (Fraud Detection)	Bid manipulation, grant fraud patterns, false reporting in RAIS	CDBG-DR HUD compliance, FinCEN BSA requirements
COMPLIANCE-001 (Federal Trigger)	All 14+ active grant programs compliance status vs. federal requirements	HUD, FEMA, USDA annual audit requirements
ACCOUNTABILITY-001 (Ethics)	Official conduct, conflict of interest, HRS Ch. 84 compliance	County Charter ethics provisions, State Ethics Commission

V. Hawaii Revised Statutes (HRS) — Full Compliance Matrix

MauiOS is designed to serve as the county's primary interface to Hawaii Revised Statutes. The Kananae Compiler creates a living cross-reference between every HRS chapter that touches county operations and the specific HFEv4 systems, node zones, and RAIS scrolls that enforce or report compliance. The table below covers all primary HRS chapters relevant to MauiOS operations.

County Administration (HRS Title 6)

HRS Chapter	Title	What It Governs	MauiOS Implementation
Ch. 46	County Organization and Administration	County powers, zoning authority (HRS 46-4), general jurisdiction, tax exemptions for housing/agriculture/homeless programs	Kananae Compiler maps all county ordinances to node-level rules; BuildingSystem enforces 46-4 zoning; GWT tax-exempt status tracking
Ch. 46-1.5	General Powers of Counties	Enforcement of claims, regulatory authority, marking/lighting of structures, county service delivery	AttributeSystem node compliance flags; ActorHealth county service delivery index per node
Ch. 50	County Fiscal Administration	Budget adoption, fiscal year, accounting standards, audit requirements	FY2027 $1.616B budget loaded into AttributeSystem; AUDIT-001 scroll triggers on fiscal anomaly; DayNightCycle tracks fiscal year cadence
Ch. 51	Revenue Bonds	County bonding authority ($120.5M authorized FY2027)	Bond obligation nodes tracked in Vendor system; payment schedule in DayNightCycle
Ch. 76-80	Civil Service, Personnel	County employee classification, civil service rights, compensation	NPCBehaviorSystem workforce simulation; Section 3 workforce compliance reporting (CDBG-DR)

Open Government (HRS Title 8)

HRS Chapter	Title	What It Governs	MauiOS Implementation
Ch. 91	Administrative Procedure	Agency rulemaking: 30-day notice, public hearings, contested case procedures	Kananae Compiler tracks HAR rulemaking; 30-day countdown in DayNightCycle; RAIS scroll created on each rulemaking initiation
Ch. 92	Public Agency Meetings and Records (Sunshine Law)	Open meetings, 6-day agenda notice, public testimony rights, executive session exceptions, minutes	Legistar scraper + Kipuka testimony engine; immudb seals every meeting record; AUDIT-001 triggers on notice failure
Ch. 92F	Uniform Information Practices Act (UIPA)	Public records access, exemptions, disclosure timeline	RAIS scroll classification system (Public / Restricted / Sealed); UIPA request processing tracked in RAIS
Ch. 84	Ethics in Government	Financial disclosure, conflict of interest, revolving door restrictions, gifts	GHOST-001 scroll: conflict of interest pattern detection; ACCOUNTABILITY-001: ethics compliance; all GWT

HRS Chapter	Title	What It Governs	MauiOS Implementation
			disbursements sealed in immudb for ethics audit
Ch. 27G	Digital Government (Portal Program)	Electronic government services, Access Hawaii, chief data officer, data/AI governance center (eff. July 1, 2025)	MauiOS aligns with OETS data governance requirements; Kipuka portal designed to meet eHawaii.gov interoperability; RAIS feeds county data to State Data Office

Natural Resources, Water, and Agriculture (HRS Title 12-13)

HRS Chapter	Title	What It Governs	MauiOS Implementation
Ch. 174C	State Water Code	Commission on Water Resource Management, water permit system, county water use plans must align with zoning/land use policies	WeatherManager water states per node; Bill 146 actual-use methodology in F_PlantAttributes_Requirements; 2 new Upcountry wells tracked in Zone 1 DT_SGI_Nodes
Ch. 171	Public Lands (DLNR)	State land management, leases, conservation district rules	MapManagement zone classifications align with DLNR conservation/agricultural boundaries; Zone 1 Mauka nodes carry conservation flags
Ch. 166	State Farm Leases	Agricultural lease terms, HDOA oversight, farm use requirements	DT_SGI_Nodes: farm lease ID field per Zone 2 node; Crafting system enforces farm use conditions for USDA VAPG compliance
Ch. 167	Irrigation Systems	State irrigation water delivery, fees, maintenance	WeatherManager irrigation node states; DT_SGI_Elements water delivery multiplier
Ch. 205	Land Use Commission	State land classification (urban/rural/agricultural/conservation), permissible agricultural uses (205-4.5 Class A/B soils)	BuildingSystem enforces 205-4.5 permitted uses; Zone 2 nodes carry soil classification A/B flags; agricultural tourism encounter tags align with 205-4.5 farm-related activities
Ch. 205A	Coastal Zone Management	Development near shoreline, Special Management Area permits	Zone 3 Makai nodes carry SMA permit flags; MapManagement enforces coastal setback rules
Ch. 195	Natural Area Reserves	Protected ecosystems, conservation management	WeatherManager biome states trigger conservation zone alerts; EventTrigger fires RAIS scroll on conservation boundary breach

Fiscal, Tax, and Finance (HRS Titles 14-17)

HRS Chapter	Title	What It Governs	MauiOS Implementation
Ch. 235	Income Tax	State income tax, withholding, business income	GWT distribution income tracking; Vendor system captures 12.5% splits for tax reporting; immudb provides audit trail

HRS Chapter	Title	What It Governs	MauiOS Implementation
Ch. 237	General Excise Tax (GET)	Hawaii GET on business transactions, county surcharge authority	Franchise fee transactions through Stripe Connect capture GET; county surcharge allocation tracked in BFED reporting layer
Ch. 246	Real Property Tax	County administration of real property tax, assessment, exemptions, appeals	DT_SGI_Nodes: real property assessment data per node; GWT 5% of certified RPT = $32.5M Affordable Housing Fund; tax exemption tracking for housing/ag programs
Ch. 40	Public Finance	General fund accounting, appropriation control, fiscal officer duties	AUDIT-001 scroll monitors appropriation vs. expenditure variance; DayNightCycle tracks fiscal year milestones
Ch. 36	State Funds	State special funds, revolving funds, trust funds	GWT trust fund structure modeled on state trust fund compliance; RAIS tracks all fund movements with immudb sealing

Judiciary (HRS Title 33)

HRS Chapter	Title	What It Governs	MauiOS Implementation
Ch. 603	Circuit Courts	General civil/criminal jurisdiction, environmental enforcement, jury trials, Second Circuit: 4 judges at Wailuku Courthouse	immudb RAIS scrolls admissible as business records; Node 54 Sovereign Audit export formatted for discovery
Ch. 604	District Courts	Civil claims <=$40K, county ordinance violations, misdemeanors, TROs, small claims <=$5K	RAIS ordinance compliance history; EventTrigger ordinance breach scroll triggers; Kipuka testimony creates sealed evidence record
Ch. 622A	Uniform Electronic Transactions Act (UETA)	Legal validity of electronic records, electronic signatures, electronic contracts	immudb sealed scrolls valid as electronic business records under UETA; Kipuka testimony submissions = legally valid electronic records
Ch. 127A	Emergency Management	Governor/Mayor emergency powers, emergency proclamations, disaster relief coordination	Node 53 Emergency Override Protocol; WeatherManager fire/disaster state triggers EMA alert; FEMA BRIC modeling deliverable

VI. State Government Interface

MauiOS is designed as a county platform, but its architecture explicitly supports upward integration to state-level agencies, the state legislature, and the Governor's office. The Kananae Compiler functions as a bidirectional translation layer between county ordinances, state HRS chapters, and the 12 Stones Sovereign Charter — creating a three-way alignment that no prior county technology platform has achieved.

State Agency Interfaces

State Agency	Relevant HRS	MauiOS Interface	Data Flow Direction
Hawaii State Legislature (capitol.hawaii.gov)	All HRS chapters	Legistar-style scraper monitors state bill activity affecting Maui County; Kananae Compiler cross-references state bills to county node impacts	State Legislature -> MauiOS (inbound monitoring)
Office of Enterprise Technology Services (OETS)	HRS Ch. 27G	MauiOS Kipuka portal aligned with eHawaii.gov interoperability standards; RAIS feeds anonymized county metrics to State Data Office; HB726 data/AI governance compliance	Bidirectional — county data to state, state standards into platform
Department of Land and Natural Resources (DLNR)	HRS Ch. 171, 174C, 195	WeatherManager biome states feed DLNR environmental monitoring; MapManagement zone classifications mirror DLNR land records; water permit data per Zone 1 node	Bidirectional — node sensor data to DLNR, DLNR classifications into node rules
Hawaii Department of Agriculture (HDOA)	HRS Ch. 147, 150A, 166	DT_Farmables crop data aligned with HDOA commodity reporting; WFCF traceability chain supports HDOA biosecurity; Akamai Arrival digital ag form integration stub	Bidirectional — farm production data up, HDOA certifications into WFCF chain
Land Use Commission (LUC)	HRS Ch. 205	BuildingSystem Zone 2 nodes carry LUC district classification (Agricultural A/B soil class); permitted uses per 205-4.5 enforced at node level	Inbound — LUC classifications define node rules
Commission on Water Resource Management (CWRM)	HRS Ch. 174C	WeatherManager water states interface with CWRM water permit system; Bill 146 actual-use methodology feeds CWRM county water use plan update requirement	Bidirectional — node water data to CWRM, CWRM permit status into node rules
Office of Planning & Sustainable Development (OPSD)	HRS Ch. 226 (Hawaii State Plan)	MapManagement zone topology aligned with State Functional Plans; DT_SGI_Nodes includes State Plan policy area per node	Inbound — State Plan priorities influence node priority scoring
Hawaii State Judiciary (Second Circuit)	HRS Ch. 603, 604	immudb scroll chain exported for discovery; Node 54 audit package formatted for court submission; Corporation Counsel receives RAIS alert feed directly	Outbound — MauiOS provides evidence and compliance records to courts
Governor's Office of Disaster Assistance	HRS Ch. 127A	Node 53 Emergency Override feeds real-time disaster state to Governor's emergency dashboard; CDBG-DR progress reports auto-generated from RAIS	Bidirectional — node emergency data up, emergency declarations into node override

Hawaii State Plan Alignment (HRS Chapter 226)

The Hawaii State Plan (HRS Chapter 226) establishes statewide functional plan priorities across nine domains. Every MauiOS node is scored against these priorities, enabling the platform to demonstrate alignment with state policy for grant applications and legislative support:

State Plan Domain (HRS 226)	MauiOS Node Zone Alignment	HFEv4 System
Population — managed growth, community character	All zones — demographic data per node via NPCBehaviorSystem	NPCBehaviorSystem + MapManagement
Economy — economic health, diversification	Zone 3 (Makai, franchise/revenue nodes)	Vendor (GWT) + AbilitySystem + DeckBuilderSubsystem licensing
Physical Environment — natural resources, conservation	Zone 1 Mauka + Node 53 (emergency/fire)	WeatherManager + EventTrigger + DT_SGI_Elements
Facility Systems — infrastructure	Zone 1 (water/utilities) + Zone 2 (FarmBox)	BuildingSystem + WeatherManager (water)
Agriculture — agricultural lands, production	Zone 2 Farmlands (primary)	FarmingSystem + Crafting + Inventory + DT_Farmables
Education — lifelong learning, workforce	Zone 2 (Sage Game education nodes)	Sage Game UE5 + Kipuka portal + Section 3 workforce
Health — health care, community well-being	All zones — ActorHealth resilience index	ActorHealth + NPCBehaviorSystem + resiliency hubs
Housing — affordable, adequate	Zone 2 (Hua Lani) + Zone 3 (workforce housing)	BuildingSystem + GWT.Eligible + $79.5M HLU tracking
Public Safety — emergency, justice	Node 53 (emergency) + Node 54 (judicial audit)	Node 53/54 Joker protocols + WeatherManager + RAIS

VII. Full Government Capability Map

The following matrix shows every governmental function MauiOS can perform and the specific platform component that delivers it. This is the complete operational picture of MauiOS as a sovereign county governance platform.

Government Function	Branch	MauiOS Component	HRS / Legal Basis	Status
Council agenda auto-processing	Legislative	Legistar scraper + DayNightCycle timestamps	HRS 92-7 agenda notice	Phase 7
Bill-to-node impact mapping	Legislative	Kananae Compiler + DT_SGI_Nodes update	HRS 46 county ordinance authority	Phase 7
Public testimony collection and sealing	Legislative	Kipuka portal + immudb (RAIS.Scroll.Sealed)	HRS 92-3.5 testimony rights	Phase 7
Sunshine Law compliance automation	Legislative	RAIS scroll pre-filing + AUDIT-001 trigger	HRS 92 (all sections)	Phase 6-7
Budget visualization (FY2027 $1.616B)	Legislative	WBP_SGI_RAISDashboard + AttributeSystem	HRS 50 fiscal admin	Phase 5-7
Committee bill tracking by node	Legislative	MapManagement + DT_SGI_Nodes committee flags	Maui Charter Art. II	Phase 2
Real-time budget monitoring	Executive	Digital Twin + AttributeSystem + Vendor	HRS 50, Charter Art. XIV	Phase 5
Grant pipeline management ($1.75B+)	Executive	RAIS grant scrolls + DayNightCycle deadlines	All active federal programs	Phase 6
GWT 12.5% allocation and disbursement	Executive	Vendor (GWT) + immudb seal	Charter Art. GWT mandate	Phase 5
Water infrastructure monitoring	Executive	WeatherManager + Zone 1 DT_SGI_Nodes	HRS 174C, Bill 146	Phase 1
Affordable housing pipeline tracking	Executive	BuildingSystem + GWT.Eligible tags	HRS 46, Charter Housing Fund	Phase 2
Agricultural production reporting	Executive	FarmingSystem + WFCF chain + Crafting	HRS 166, USDA VAPG	Phase 3
FarmBox deployment status	Executive	BuildingSystem + DT_Buildables	MOU Section 4 Phase 3	Phase 2
Emergency management (FEMA BRIC)	Executive	Node 53 + WeatherManager + FARSITE stub	HRS 127A, FEMA BRIC	Phase 4
Section 3 workforce compliance	Executive	NPCBehaviorSystem + RAIS workforce scrolls	CDBG-DR Section 3 (HUD)	Phase 3
Ethics compliance monitoring	Executive	GHOST-001 + ACCOUNTABILITY-001 scrolls	HRS 84, Charter Art. XIII	Phase 4-6
Dispute resolution (Tier 1-3)	Judicial	Kipuka testimony + RAIS mediation scrolls + immudb	MOU dispute resolution protocol	Phase 7
Court evidence packaging (Node 54)	Judicial	Node 54 Sovereign Audit + immudb export	HRS 622A UETA, Ch. 603-604	Phase 6
Fraud detection and legal referral	Judicial	GHOST-001 scroll + Corporation Counsel alert	HRS 84, FinCEN BSA	Phase 4
Ordinance compliance history	Judicial	RAIS ordinance breach scrolls + EventTrigger	HRS 604 District Court jurisdiction	Phase 4
Electronic record legal validity	Judicial	immudb (UETA-compliant) + RAIS.Scroll.Sealed	HRS 622A	Phase 6
HRS-to-Charter cross-reference	State HRS	Kananae Compiler (Legislation -> Charter Article)	All HRS chapters + Charter SSC v5	Phase 7
Land use classification per node	State HRS	MapManagement + BuildingSystem (HRS 205 enforcement)	HRS 205, Ch. 46-4	Phase 2
Water permit tracking per	State HRS	WeatherManager + Zone 1	HRS 174C CWRM	Phase 1

Government Function	Branch	MauiOS Component	HRS / Legal Basis	Status
node		DT_SGI_Nodes		
Real property tax data per node	State HRS	DT_SGI_Nodes RPT field + Vendor GWT 5%	HRS 246	Phase 1
State data governance compliance	State HRS	RAIS OETS feed + Kipuka eHawaii.gov alignment	HRS 27G (eff. July 1, 2025)	Phase 7
UIPA public records management	State HRS	RAIS scroll classification (Public/Restricted/Sealed)	HRS 92F	Phase 6

VIII. Architectural Integrity — Why This Works

The 52-node HFEv4 topology is the architectural insight that makes MauiOS function as a true government operating system rather than a collection of disconnected apps. Each of the 52 nodes simultaneously is: a physical community node (a place, a farm, a district), a policy node (what rules apply here), a data node (what is being measured here), a financial node (what money flows through here), and a compliance node (what laws govern here).

This five-layer simultaneous function means that when a council bill passes, an executive department acts, a court rules, or a state agency issues new rules — the effect propagates automatically through the node system. The Kananae Compiler is the translator. The RAIS engine is the compliance enforcer. The immudb chain is the evidence layer. And the Sage Game (UE5.7 Unreal Engine) is the visualization and simulation environment that makes all of this legible to elected officials, community members, and judges alike.

The Sovereign Charter as Constitutional Layer

The 12 Stones Sovereign Charter (SSC v5, 28 Articles, filed April 14, 2025) is not just a company charter. It is a governance constitution that sits between the Maui County Charter and the 12 Stones technology platform. When a County Charter provision, an HRS statute, or a federal grant requirement needs to be encoded into the platform, the Kananae Compiler first checks it against the Sovereign Charter. The result is a three-way alignment:

Constitutional Layer	Authority	MauiOS Role
Federal Law (HUD, FEMA, USDA, DOE)	Highest — supersedes all	Grant pipeline compliance; Node 53 FEMA interface; CDBG-DR action plan tracking
Hawaii Revised Statutes (HRS)	State law — governs county authority	Kananae Compiler statutory cross-reference; all 30+ HRS chapters mapped to node rules
Maui County Charter (2023 Edition)	Local constitutional authority	Council, Mayor, and Corporation Counsel functions mapped to platform roles
12 Stones Sovereign Charter (SSC v5)	Platform constitutional layer	Governs GWT, IP ownership, dispute resolution, data sovereignty, node governance rules
MauiOS Platform Rules (DT_SGI_Nodes)	Operational layer	Node-level rules encoding all four constitutional layers into executable system behavior

The Platform's Five Foundational Guarantees to Government

Guarantee	What It Means	How MauiOS Delivers It
Transparency	Every government action is recorded, timestamped, and publicly accessible per HRS Chapter 92	immudb immutable ledger + WBP_SGI_RAISDashboard + Kipuka public portal
Accountability	Every dollar, every decision, every compliance event is traceable to a responsible party and a sealed record	AUDIT-001 + GHOST-001 + ACCOUNTABILITY-001 scrolls + Node 54 sovereign audit
Participation	Every resident can testify, track their community node's health, and see the impact of council decisions in near-real-time	Kipuka portal + WBP_SGI_NodeMap + public testimony seal in immudb
Compliance	Every federal grant, state statute,	Kananae Compiler + COMPLIANCE-001 scroll +

Guarantee	What It Means	How MauiOS Delivers It
	county ordinance, and charter provision is encoded in the node system and enforced automatically	DT_SGI_Nodes rule engine
Sovereignty	12 Stones owns the software; the County owns the data. No vendor lock-in. Data stored in Hawaii (University of Hawaii data center for cold archive)	IP ownership per MOU Section 12; Cloudflare R2 + UH data center; Charter SSC v5 Art. on sovereignty

12 Stones Global Inc. · jimlangford@me.com · Maui, Hawaii
REGENERATE · EDUCATE · GOVERN · LIBERATE
CONFIDENTIAL — MauiOS Government Architecture Document · April 2026 · Version 1.0

Sources and Legal References

Maui County Charter (2023 Edition)
https://www.mauicounty.gov/DocumentCenter/View/139187/CHARTER---January-2023
HRS Chapter 46 — County Organization
https://law.justia.com/codes/hawaii/title-6/chapter-46/
HRS Chapter 92 — Sunshine Law
https://files.hawaii.gov/luc/docs/hrs_92.pdf
HRS Chapter 91 — Administrative Procedure
https://www.hawaii.edu/ohelo/statutes/HRS091/HRS091.html
HRS Chapter 174C — State Water Code
https://files.hawaii.gov/dlnr/cwrm/regulations/Code174C.pdf
HRS Chapter 205 — Land Use Commission
https://files.hawaii.gov/luc/docs/hrs_chapter205_web.pdf
HRS Chapter 603 — Circuit Courts
https://www.capitol.hawaii.gov/hrscurrent/vol13_ch0601-0676/HRS0603/HRS_0603-.htm
HRS Chapter 604 — District Courts
https://www.capitol.hawaii.gov/hrscurrent/vol13_ch0601-0676/HRS0604/HRS_0604-.htm
HRS Chapter 27G — Digital Government
https://data.capitol.hawaii.gov/sessions/session2025/bills/HB726_.HTM
Maui County FY2027 Budget ($1.616B)
https://mauinow.com/2026/03/25/mayor-bissen-proposes-1-616-billion-budget-for-fiscal-2027/
Hawaii Data & AI Summit 2026
https://events.govtech.com/Hawaii-Data-AI-Summit
Maui County Council — Bills Log
https://mauicounty.us/laws/
Guide to Government in Hawaii (LRB)
https://lrb.hawaii.gov/wp-content/uploads/GuideToGovernmentInHawaii.pdf
Second Circuit Court — Maui
https://www.courts.state.hi.us/25993244-2

BFED Committee

From: Brian Mercer <bmercer78@gmail.com>
Sent: Sunday, April 5, 2026 10:37 PM
To: BFED Committee
Subject: Written testimony for the BUDGET, FINANCE, AND ECONOMIC DEVELOPMENT COMMITTEE (2025-2027) meeting on Monday, April 6, 2026 at 9:00 a.m.
Attachments: Brian Mercer testimony.pdf

[You don't often get email from bmercer78@gmail.com. Learn why this is important at https://aka.ms/LearnAboutSenderIdentification]

This written testimony is for the BUDGET, FINANCE, AND ECONOMIC DEVELOPMENT COMMITTEE (2025-2027) meeting on Monday, April 6, 2026 at 9:00 a.m.

TESTIMONY IN SUPPORT OF FUNDING FOR LA‘AKEA VILLAGE HOUSING PROJECT
Before the Maui County Council Budget, Finance, and Economic Development Committee

Aloha Chair Sugimura, Vice Chair Batangan, and Members of the Committee,

My name is Brian Mercer, resident of Paia, and I live in Skill Village neighboring the proposed housing project. I am writing in strong support of the County of Maui Affordable Housing Fund request of **$4.5 million** for the La‘akea Village supported housing project in Pā‘ia.

My husband and I are personal trainers and have volunteered in the past at La’akea Village teaching fitness classes to their participants. We’ve seen firsthand how empowering their program for adults with IDD is. We’ve also heard firsthand from their participants how isolating life can sometimes be at home, away from their La’akea community. Not only would supporting the project provide much needed affordable and safe housing, it would continue to provide that sense of community after the day program has ended.

Particularly as adults with IDD age out of family homes or lose caregivers, finding housing that is both accessible and affordable can be a struggle. This project adds much needed ADA-accessible housing inventory to Maui with dedicated units for adults with disabilities and integrated housing for support staff and other income-qualified residents.

We have lived across the street from La’akea for over 11 years now and have seen through our time volunteering as it has grown and added a commercial kitchen, a store, and developed a farm that grows produce and flowers and now has horses. We’ve seen how well everyone does there and watched the individuals who take part in their program grow. We’ve seen their lives enriched. We’d love to see La’akea Village continue to grow and thrive with a housing community.

I am in favor of the proposed development and urge the committee to fully fund the requested $4.5 million so that this project can move forward to development.

Mahalo for your time and consideration.

Sincerely,
Brian Mercer
666 Pili Loko St.
Paia

BFED Committee

From: Chris Mercer <chrs_mercer@yahoo.com>
Sent: Sunday, April 5, 2026 10:41 PM
To: BFED Committee
Subject: Written testimony for the BUDGET, FINANCE, AND ECONOMIC DEVELOPMENT COMMITTEE (2025-2027) meeting on Monday, April 6, 2026 at 9:00 a.m.
Attachments: cid24212BED-3FFB-4F8A-ACA0-F5A33A60D536.pdf

[You don't often get email from chrs_mercer@yahoo.com. Learn why this is important at https://aka.ms/LearnAboutSenderIdentification]

This written testimony is for the BUDGET, FINANCE, AND ECONOMIC DEVELOPMENT COMMITTEE (2025-2027) meeting on Monday, April 6, 2026 at 9:00 a.m.

TESTIMONY IN SUPPORT OF FUNDING FOR LA‘AKEA VILLAGE HOUSING PROJECT
Before the Maui County Council Budget, Finance, and Economic Development Committee

Aloha Chair Sugimura, Vice Chair Batangan, and Members of the Committee,

My name is Chris Mercer, resident of Paia, and I live in Skill Village neighboring the proposed housing project. I am writing in strong support of the County of Maui Affordable Housing Fund request of **$4.5 million** for the La‘akea Village supported housing project in Pā‘ia.

My husband and I are personal trainers and have volunteered in the past at La’akea Village teaching fitness classes to their participants. We’ve seen firsthand how empowering their program for adults with IDD is. We’ve also heard firsthand from their participants how isolating life can sometimes be at home, away from their La’akea community. Not only would supporting the project provide much needed affordable and safe housing, it would continue to provide that sense of community after the day program has ended.

Particularly as adults with IDD age out of family homes or lose caregivers, finding housing that is both accessible and affordable can be a struggle. This project adds much needed ADA-accessible housing inventory to Maui with dedicated units for adults with disabilities and integrated housing for support staff and other income-qualified residents.

We have lived across the street from La’akea for over 11 years now and have seen through our time volunteering as it has grown and added a commercial kitchen, a store, and developed a farm that grows produce and flowers and now has horses. We’ve seen how well everyone does there and watched the individuals who take part in their program grow. We’ve seen their lives enriched. We’d love to see La’akea Village continue to grow and thrive with a housing community.

I am in favor of the proposed development and urge the committee to fully fund the requested $4.5 million so that this project can move forward to development.

Mahalo for your time and consideration.

Sincerely,
Chris Mercer
666 Pili Loko St.
Paia

BFED Committee

From: County Clerk
Sent: Monday, April 6, 2026 7:26 AM
To: BFED Committee
Subject: Fw: Letter for the ag and business internship ship

From: Eliza Brooks <eliza.brooks.555@gmail.com>
Sent: Sunday, April 5, 2026 11:06 AM
To: County Clerk <County.Clerk@mauicounty.us>
Subject: Letter for the ag and business internship ship

You don't often get email from eliza.brooks.555@gmail.com. Learn why this is important

Hi my name is Eliza Brooks i go by liza I was an intern for the ag and business internship. i interned at hale auka garden and farm eco retreat in haiku. I learned how to use and start a hydroponic system and I learned more about the Aina I didn't know before. I did up to 120 hours for this internship learning new things as I went like how to check if eggs are fertilized or not and how to trim feather so the birds don't fly away we did weeding feeding animals and we picked the vegetables and fruit for the stores and online. I also learned that clean a fish pond takes time and effort you have to make sure no fish get damaged in the process of getting them out and transferred then you have to drain the pond and make sure that their air circulation pump isn't clogged or anything like that and clean the filter then fill the pond back up . My experience for this whole internship was great I got a lot out of and it might be something im going to do more in the future. These experience give you more then just knowledge it gives you something you look forward to and it keeps youu active, sometimes it might be hard to continue to work but if you find the joy in it you will find your way .

BFED Committee

From: County Clerk
Sent: Monday, April 6, 2026 7:26 AM
To: BFED Committee
Subject: Fw: Maui County Funding Support - STEMworks
Attachments: STEMS MEDB Testimony (Pellegrino).pdf

From: Alana <akaopuiki@gmail.com>
Sent: Sunday, April 5, 2026 6:37 PM
To: County Clerk <County.Clerk@mauicounty.us>
Cc: lalaine@medb.org <lalaine@medb.org>; Mrs. Austin <20091595@k12.hi.us>
Subject: Maui County Funding Support - STEMworks

You don't often get email from akaopuiki@gmail.com. Learn why this is important

Aloha,

Our family‘s written testimony in support of Maui STEMworks is attached.

Mahalo,
Alana Kaopuiki-Pellegrino

4/5/26

Aloha mai kākou,
‘O mākou ka ‘Ohana Pellegrino. On behalf of myself Hōkūao Pellegrino, wife Alana Ka‘ōpūiki-Pellegrino, and daughter Kawaihua Pellegrino, we are submitting written testimony in strong support for funding the Maui Economic Development Board’s Economic Development Programs" specifically STEMworks Programs. Our ‘ohana engage in these programs that exist both at schools as well as on our farm as part of the Ag Internship Program.

As parents, we were thrilled that Kalama offered a plethora of afterschool enrichment opportunities at no cost. Not only were the offerings available creative and interesting, but they also helped promote advanced skill building and mentorship opportunities. Our daughter, Kawaihua is a 7th grader at Kula Kaiapuni ‘o Kalama (Kalama Intermediate School) and partakes in two MEDB Stemworks Afterschool programs: Papa Hula and Volleyball. Through these programs, she has developed new friendships with other STEMsters and strengthened her self-confidence with the help of positive and encouraging staff. We know our daughter is in a safe, stable, and nurturing environment extending her education instead of being home alone after school before we get home. STEMworks Programs respect students‘ maturity and agency while still providing structure and guidance.

We are grateful this year that she was also selected to attend the STEMworks Hawaii Conference. Not only is this a great opportunity for our daughter to learn about different STEM careers, but it provides her with enriching off-island experiences. She is excited to have such a large venue to share the ‘ike she learns from Ka Pā Hula o Kalama with the rest of the conference. By sponsoring the travel costs, it keeps our family from having to stress about fundraising and creates an opportunity for us to teach her about the importance of giving back. Kalama School provides a Kalama Cares night that hosts a hō‘ike showcasing the learning that occurred in the year. It is a fantastic event for our ‘ohana to be a part of and learn from STEM representatives in our own community.

Our farm, Noho‘ana Farm also engages with STEMworks Programs by being a host and mentorship site for the Ag Internship Program. We have been working with this program now for 9 years and we have had tremendous success in providing a unique skillset around food production, food processing and culture-based education with over a dozen high school students. We currently have 2 high school students who started in the fall. They help with numerous tasks, like milling poi for various community events. We would not have been able to hit the many goals we have set forth at our farm without the assistance of this program.

We feel blessed and lucky to be a part of the STEMworks community. We hope that other keiki will have the opportunity to attend these great programs in the future.

Mahalo,

Hōkūao Pellegrino, Alana Ka‘ōpūiki-Pellegrino, Kawaihua Pellegrino, and Keālia Pellegrino

BFED Committee

From: County Clerk
Sent: Monday, April 6, 2026 7:27 AM
To: BFED Committee
Subject: Fw: MEDB STEMworks written testimony
Attachments: MEDB STEMworks written testimony.pdf

From: Carlys Higuchi <carlyshiguchi@gmail.com>
Sent: Friday, April 3, 2026 7:23 PM
To: County Clerk <County.Clerk@mauicounty.us>
Cc: Lalaine STEM Medb <lalaine@medb.org>
Subject: MEDB STEMworks written testimony

You don't often get email from carlyshiguchi@gmail.com. Learn why this is important

Hi,

Please see the attachment that my daughter, Kalena Lee, wrote asking for continued support of the MEDB Stemworks program.

Mahalo for your consideration.

Carlys Higuchi

4/3/26

Aloha, my name is Kalena Lee and I'm a 5th grader at Pukalani Elementary School. I have been in STEM for 3 years, starting in 2nd grade. Some of the STEM courses that I have done are: Digital art, Intro to STEM, Coding, Robotics, STEM Research Team and Technovation Girls. Now I am here to tell you a few reasons why you should keep supporting the Maui Economic Development Board STEM Works Program.

One of the things I've learned in STEM is how to be a good leader by showing up and always giving it my all even when I'm not up for it. I learned how to be a good teammate by not putting others down, giving everyone a chance to be a leader and by helping each other. I learned that it's good to keep an open mind and try new things even if I didn't want to at first because it can end up being a new experience in the end. I think that if I didn't do STEM, I would not be confident in myself and I would be a lot more shy.

STEMworks has given me the opportunity to go on fun memorable field trips such as the How to Save a Life program at the Maui Memorial Medical Hospital and the trip to Tokyko, Japan to participate in the Japan Cup with the robotics team.

Another thing that STEMworks has opened up to me is being able to teach others, both older and younger people, what STEM is and why it's important. One thing I did was, I helped teach a club of retired teachers how to use Canva to make greeting cards and other things. I also taught kindergarten through 5th grade students at Pukalani Elementary school the importance of STEM and why they should join.

In conclusion, these are my reasons why I think you should continue to support MEDB Stemworks program. Mahalo for listening.

Kalena Lee
Pukalani Elementary School
5th Grade

BFED Committee

From: Sandra Abe <sandra.abe@gmail.com>
Sent: Thursday, April 2, 2026 10:05 PM
To: County Clerk
Cc: mountainbikemaui@gmail.com
Subject: Letter from Mary Charles for Maui Mountain Bike Coalition
Attachments: Maui Mountain Bike Coalition Ltr.pdf

Please see the attached letter from Mary Charles.
Thank you.

April 2, 2026

Aloha Chair & Council Members:

My name is Mary Charles and am writing in support of funding for the Maui Mountain Bike Coalition. I am a long-time resident of Honolulu and a part-time resident of Lanai. I have been riding bikes since I was four-years old. I rode my bike to school, to meet friends and for the pure joy of being outside and free! My first three years of living in Honolulu when I moved here in 1970, I rode my bike to work and to meet friends arriving at the airport who came to visit. It has always been a way of life for me. I took up mountain biking later in life which has been an enriching experience for me to go "off road".

Sadly, as I look at our youth today, they are more and more dependent on their parents, cars, and mobile devices. We need to support the Maui Mountain Bike Coalition to give our kids the opportunity to get outside and thrive. Get exercise, learn skills in navigating and traversing trails may help in other areas of their lives. Make new friends, learn from the biking mentors and so much more than just riding a mountain bike!

Please support the efforts for our youth. It is so desperately needed for their mental and physical health.

With Appreciation & Support,

Mary Charles

Mary Charles
300 Kaunaoa Drive
Lanai City, Hawaii 96763
m: 808-479-5577

BFED Committee

From: Ronjini Joshua <Ronjini@thesilvertelegram.com>
Sent: Friday, April 3, 2026 12:02 PM
To: County Clerk
Cc: lalaine@medb.org
Subject: Testimony in Support of MEDB STEMworks Funding

You don't often get email from ronjini@thesilvertelegram.com. Learn why this is important

Aloha Chair and Councilmembers,

My name is Ronjini Joshua, and I am a parent of three children who have participated in the MEDB STEMworks programs over the past 3 years. I am writing today in strong support of continued funding for STEMworks through the County of Maui.

As a parent, I have seen firsthand the impact this program has had on my children. Before participating, math and science were simply subjects in school. Through STEMworks, they have become passions. My children are now excited about problem-solving, curious about how things work, and eager to explore careers in science, technology, and engineering.

What stands out most is their enthusiasm. They genuinely love the program. Every session, every activity, every opportunity has deepened their interest and confidence. In fact, we make it a priority to register as soon as enrollment opens because they don't want to miss it.

With three children, access to high-quality STEM enrichment like this would not be financially feasible for our family without the support of this grant. STEMworks has made these opportunities accessible, equitable, and impactful for so many families across Maui.

Programs like this create an important foundation for learning and thinking outside the box. They are shaping the next generation of thinkers, innovators, and leaders right here in our community.

I respectfully urge you to continue funding MEDB STEMworks so that families like mine can continue to benefit from these life-changing opportunities.

Mahalo for your time and for supporting the future of our keiki and our community.

--

Ronjini Joshua
ronjini@thesilvertelegram.com
(949) 295-9779 Mobile/WhatsApp

BFED Committee

From: County Clerk
Sent: Monday, April 6, 2026 9:45 AM
To: BFED Committee
Subject: Fw: Bill 55 section 6 item 12: Funding needs for Pa'ia Youth Council, Inc. / PYCC!

From: Billy Jalbert <billy@mauirealestate.com>
Sent: Monday, April 6, 2026 9:40 AM
To: County Clerk <County.Clerk@mauicounty.us>
Cc: Dean Wong <dwong@discoverimua.com>; K. Kauanoe Batangan <k.kauanoe.batangan@mauicounty.us>; Alice L. Lee <Alice.Lee@mauicounty.us>; Yukilei Sugimura <Yukilei.Sugimura@mauicounty.us>; Thomas M. Cook <Thomas.Cook@mauicounty.us>; Gabe Johnson <Gabe.Johnson@mauicounty.us>; Tamara A. Paltin <Tamara.Paltin@mauicounty.us>; Keani N. Rawlins <Keani.Rawlins@mauicounty.us>; Shane M. Sinenci <Shane.Sinenci@mauicounty.us>; Nohe M. Uu-Hodgins <Nohe.Uu-Hodgins@mauicounty.us>
Subject: Bill 55 section 6 item 12: Funding needs for Pa'ia Youth Council, Inc. / PYCC!

You don't often get email from billy@mauirealestate.com. Learn why this is important

Aloha County Council Members,

Mahalo for your service and support to Maui County. My name is Billy Jalbert, I am the President of the Board of the Pa'ia Youth & Cultural Center. I have served PYCC for 24 years because I care deeply about our community and believe that PYCC's mission of Building Community Through Our Youth helps make a better Maui for all of us.

PYCC serves Maui's youth in a facility that sits right on the ocean adjacent to Baldwin Beach park – which is both a blessing and a challenge. Besides providing a safe haven for youth PYCC provides a place where our young members connect, learn life skills in radio broadcasting, the multimedia arts, culinary training, the skate park and of course the ocean. But that salt air? It's tough on everything...

We have requested an increase in County support for FY27 and FY 28 to $554,976.94 Let me break down why this investment matters and why we need your help now more than ever.Simply stated every single line item in our budget costs more than before.

This budget supports 10 positions – seven full-time youth specialists and three part-timers. These aren't just staff members; they're mentors, role models, and sometimes the only stable adults in these kids' lives. Running our radio station, skate park, culinary program, multimedia lab, the girl's club – keeping kids engaged, learning life skills, and staying out of trouble. Our staff are all underpaid and overworked. Their passion and commitment to our mission and youth keeps the doors open. It's been several years since we requested an increase in support.

But here's the thing – your $554,976.94 investment only covers 60% of our actual costs. We have to raise the other 40% – nearly $386,000 – through private donations and fundraising. So, this isn't a handout; it's a partnership and we work hard to deliver real results for Maui.

Now, about timing. Our long-term Executive Director, Susun White retired at the end of 2024. We were fortunate that we had Benjamin Rachunas ready to step up. He's been with us for 20 years and knows this organization inside and out. Ben knows how to run PYCC and serve our youth. What we need now is stability and time to build those external funding relationships. **Without stable County support during this transition, we risk losing other funding sources** because donors get nervous when they see instability.

After our former ED retired we discovered gaps in our administrative systems – things managed informally, processes not documented, financial controls that needed strengthening. **That's when Imua Family Services stepped in to support us.**

The $43,869 in administrative costs from the County – that's our partnership with Imua. They handle financial reconciliations, payroll processing, tax compliance, audit preparation, grant monitoring and reporting, fundraising support – all the critical infrastructure that keeps programs running legally and effectively. **Without Imua's support, we honestly don't know if PYCC would still be operating at full capacity today.** They created a solid foundation so Ben can focus on leading programs and building funding relationships.

Let me put it simply: COVID hit us hard. The wildfires hit us hard. Rising maintenance costs hit us hard. Losing our ED created uncertainty. **But our kids? They keep showing up. Every day that we are open!** They still need us. They count on PYCC to be their safe haven.

Without this investment right now, we face a domino effect where reduced County support signals to other funders that maybe PYCC isn't stable, maybe they should pull back too. And then where are our kids?

This budget isn't padding – it's what it actually costs to run quality youth programs in an oceanfront facility on Maui in 2027. Every dollar goes toward either the people serving our youth or the infrastructure keeping those programs running.

I'm asking you to continue to invest in Maui's future by investing in PYCC. Give us the stability to transition leadership successfully. Give us the foundation to continue serving the young people who need us most.

Our kids are counting on us. And we are counting on you.

Mahalo for your consideration and support,

Billy Jalbert
Board President
Pa'ia Youth Council Inc. (DBA Pa'ia Youth & Cultural Center).

Union Calendar No. 600

99th Congress, 2d Session - - - - - - - - - - - - - - - House Report 99-1016

INVESTIGATION OF THE CHALLENGER ACCIDENT

REPORT

OF THE

COMMITTEE ON SCIENCE AND TECHNOLOGY HOUSE OF REPRESENTATIVES

NINETY-NINTH CONGRESS

SECOND SESSION

OCTOBER 29, 1986.—Committed to the Committee of the Whole House on the State of the Union and ordered to be printed

U.S. GOVERNMENT PRINTING OFFICE

64-420 O WASHINGTON : 1986

COMMITTEE ON SCIENCE AND TECHNOLOGY

DON FUQUA, Florida, *Chairman*

ROBERT A. ROE, New Jersey
GEORGE E. BROWN, JR., California
JAMES H. SCHEUER, New York
MARILYN LLOYD, Tennessee
TIMOTHY E. WIRTH, Colorado
DOUG WALGREN, Pennsylvania
DAN GLICKMAN, Kansas
ROBERT A. YOUNG, Missouri
HAROLD L. VOLKMER, Missouri
BILL NELSON, Florida
STAN LUNDINE, New York
RALPH M. HALL, Texas
DAVE McCURDY, Oklahoma
NORMAN Y. MINETA, California
BUDDY MacKAY,** Florida
TIM VALENTINE, North Carolina
HARRY M. REID, Nevada
ROBERT G. TORRICELLI, New Jersey
RICK BOUCHER, Virginia
TERRY BRUCE, Illinois
RICHARD H. STALLINGS, Idaho
BART GORDON, Tennessee
JAMES A. TRAFICANT, JR., Ohio
JIM CHAPMAN, Texas

MANUEL LUJAN, JR.,* New Mexico
ROBERT S. WALKER, Pennsylvania
F. JAMES SENSENBRENNER, JR., Wisconsin
CLAUDINE SCHNEIDER, Rhode Island
SHERWOOD L. BOEHLERT, New York
TOM LEWIS, Florida
DON RITTER, Pennsylvania
SID W. MORRISON, Washington
RON PACKARD, California
JAN MEYERS, Kansas
ROBERT C. SMITH, New Hampshire
PAUL B. HENRY, Michigan
HARRIS W. FAWELL, Illinois
WILLIAM W. COBEY, JR., North Carolina
JOE BARTON, Texas
D. FRENCH SLAUGHTER, JR., Virginia
DAVID S. MONSON, Utah

HAROLD P. HANSON, *Executive Director*
ROBERT C. KETCHAM, *General Counsel*
REGINA A. DAVIS, *Chief Clerk*
R. THOMAS WEIMER, *Republican Staff Director*

CHALLENGER ACCIDENT REPORT STAFF TASK GROUP

ROBERT C. KETCHAM, *General Counsel*
RADFORD BYERLY, Jr.
RONALD E. WILLIAMS
PATRICIA G. GARFINKEL
A. SCOTT CROSSFIELD
CAROLYN C. GREENFELD
DAVID D. CLEMENT
SHIRLEY J. WATSON

ROBERT E. PALMER
GEORGE S. KOPP
HARRY S. DAWSON, Jr.
HARRIET J. SMITH
NELSON L. MILDER
CAROLYN J. RADABAUGH
JAMES PAUL
GLORIA J. DUNDERMAN

---

*Ranking Republican Member.
**Serving on Committee on the Budget for 99th Congress.

LETTER OF SUBMITTAL

U.S. HOUSE OF REPRESENTATIVES,
Washington, DC, October 29, 1986.

Hon. THOMAS P. O'NEILL, Jr.,
The Speaker of the U.S. House of Representatives, Washington, DC.

DEAR MR. SPEAKER:

By direction of the Committee on Science and Technology, I hereby submit the Committee's investigative report on the Challenger accident. The report was approved by the Committee on October 7, 1986. The report was carried out under the direction of the Ranking Majority Member, Robert A. Roe, who chaired the hearings and instructed the Committee staff assigned to the investigation.

Sincerely,

DON FUQUA, *Chairman.*

Enclosure.

LETTER OF TRANSMITTAL

U.S. HOUSE OF REPRESENTATIVES,
COMMITTEE ON SCIENCE AND TECHNOLOGY,
Washington, DC, October 29, 1986.

To Robert A. Roe, Ranking Majority Member, Committee on Science and Technology.

DEAR MR. CHAIRMAN:

I am forwarding this investigative report on the Challenger Accident prepared at your request.

On your instructions, the staff has carefully reviewed the information made available to the Committee and prepared the necessary findings and recommendations. The task was enormous and has been done in a careful and painstaking manner. I am especially grateful to all the staff which is listed on the inside cover of the report for their professional and thorough application to this important report. We worked in a collegial manner and on a completely bipartisan basis. I particularly want to thank the three group leaders, Ron Williams, Terry Dawson and Harriet Smith who headed up the respective subgroups for the Accident, the Technical, and the Management Issues.

Sincerely,

ROBERT C. KETCHAM, *General Counsel.*

CONTENTS

	Page
I. INTRODUCTION	1
II. CONCLUSIONS	3
III. COMPILATION OF ISSUES, FINDINGS, AND RECOMMENDATIONS	9
IV. BACKGROUND	35
V. THE ACCIDENT	39
A. Introduction	39
B. Solid Rocket Motors	50
1. History	50
2. Summary of Casing Joint Design	60
3. Testing and Certification	64
4. Manufacturing	64
5. Stacking Operations	65
6. Summary of Launch Operations	68
7. Retrieval, Transportation, and Refurbishment	73
C. External Tank	74
D. Crew Survival	76
E. Sabotage	79
F. Additional Avenues of Investigation	79
VI. DISCUSSION OF CRITICAL ISSUES	85
A. Technical Issues	85
1. Hardware Development and Production	85
a. Problems in Hardware Certification	85
b. Recurrent Hardware Problems	92
c. Other Engineering Concerns	99
d. Desirable Tests Not Yet Approved	101
e. Production/Refurbishment Issues	105
f. Review of NASA's Redesign/Recertification Plan	109
2. Operations	114
a. Shuttle Processing Issues	114
b. Pressures on Shuttle Operations	119
c. Impact of Pressures on Shuttle Operations	122
d. Other Safety Issues	132
B. Management Issues	138
1. Technical Management	138
a. Risk Management Issues	138
b. Launch Decision Process	144
c. Technical Expertise of Personnel	154
d. Change Control Process	161
2. Organization and Policy Management	165
a. Management Structure	165
b. Communication	170
c. Safety, Reliability, and Quality Assurance	174
d. Contractor Incentives	179
VII. CASING JOINT DESIGN	183
VIII. LAUNCH OPERATIONS	207
IX. DEFINITION OF TERMS AND ACRONYMS	253
X. APPENDICES	255
V-A	255
V-B	260
V-C	273
V-D	275
V-E	277
V-F	283

	Page
V-G	285
V-H	286
V-I	288
V-J	290
VI-A	292
VI-B	319
VI-C	321
VI-D	327
VII-A	369
VII-B	376
VII-C	278
VIII-A	280
VIII-B	384
VIII-C	387
VIII-D	393
VIII-E	395
VIII-F	406
VIII-G	417
VIII-H	425
VIII-I	428
VIII-J	434
VIII-K	438
VIII-L	442

Union Calendar No. 600

99TH CONGRESS *2d Session* | HOUSE OF REPRESENTATIVES | REPORT 99-1016

INVESTIGATION OF THE CHALLENGER ACCIDENT

OCTOBER 29, 1986.—Committed to the Committee of the Whole House on the State of the Union and ordered to be printed

Mr. FUQUA, from the Committee on Science and Technology, submitted the following

REPORT

I. INTRODUCTION

On January 28, at 11:39 a.m., the Space Shuttle Challenger and its crew suffered a tragic accident during launch. That same day the House of Representatives adopted H. Res. 361 which expressed the profound sorrow of the House for the tragedy and offered condolences to the families of the Challenger crew members.

During consideration of the resolution Chairman Fuqua informed the full House of Representatives that, in conformance with its oversight responsibilities, the Committee on Science and Technology would conduct a comprehensive investigation into the cause of this accident.

This report is the result of the Committee's inquiry. It contains the best efforts of the Committee to review the work of the Presidential Commission on the Space Shuttle Challenger Accident (hereafter referred to as the Rogers Commission) and the work of the National Aeronautics and Space Administration (NASA) in investigating the causes of the accident, and reviewing the recommendations to resume safe flight.

In addition to reviewing the five volumes of the Rogers Commission Report, the Committee also had direct on-line access to the entire Rogers Commission data base, which included full-text and document retrieval capability.[1]

The findings and recommendations contained in this report are the product of the Committee's own extensive hearing record,

[1] *Report of the Presidential Commission on the Space Shuttle Challenger Accident,* Volumes I-V, Washington, D.C., June 6, 1986. (Hereafter referred to as Rogers Commission Report.)

which includes materials submitted for the record, staff investigations, interviews, and trips.

It should be understood that the role of this Committee is different from that of the four-month Rogers Commission. The Committee, which authorized the funds and reviewed the lengthy development process which led to the successful Shuttle program, has a responsibility to insure that the tragic accident, and those events that led up to it, are understood and assimilated into all levels and activities of NASA so that safe manned space flight can be resumed.

In carrying out its annual authorizing responsibilities, the Committee endorses the programs and activities of NASA, and functions as a key player in the legislative activities of our federal system. As part of the fulfillment of this role, the Committee has reviewed the report of the Rogers Commission, called upon numerous witnesses, and utilized many members of its staff to prepare and review the material that has produced this report.

The Committee has been most fortunate in its work due to the diligent and thorough investigation undertaken by the Rogers Commission and the NASA investigation panels that supported the Commission. The Commission's exhaustive efforts to achieve completeness as it came to grips with a very complex technical and management system are very commendable, and will serve as a model for future Presidential Commissions.

The Committee wishes to express its appreciation for the assistance of the House Administration Committee, the Rogers Commission staff, and the Justice Department's Office of Litigation Support, Civil Division. Each of these groups was very cooperative and helpful in providing the access to, and equipment for, the Challenger accident data base needed by the Committee to do its work. In addition, the Committee very much appreciates the assistance of NASA personnel who responded to numerous requests for briefings and documents during the course of the investigation.

II. CONCLUSIONS

In execution of its oversight responsibilities, the Committee on Science and Technology has conducted a thorough investigation of the Challenger accident. Although the Committee's concern and evaluation in this report are related specifically to the safe and effective functioning of NASA's Space Shuttle program, it should be understood that our larger objective and greater responsibility are to insure that NASA, as the Nation's civilian space agency, maintains organizational and programmatic excellence across the board.

What we as a Committee, NASA as an agency, and the Nation as a whole, also must realize is that the lessons learned by the Challenger accident are universally applicable, not just for NASA but for governments, and for society. We hope that this report will serve this much larger purpose.

The Committee's investigation included: ten formal hearings involving 60 witnesses; an extensive review of the report of the Rogers Commission along with its voluminous supporting appendices and related reports by the investigation panels at NASA, as well as numerous briefings and interviews with NASA officials, contractor personnel, outside experts, and other interested parties.

From the outset, the focus of the Committee's investigation has been on understanding each of the following:

> What was the cause, or causes, of the Challenger accident?
>
> Are there other inherent hardware or management-related deficiencies that could cause additional accidents in the future?
>
> What must be done to correct all of these problems so that the Space Shuttle can be safely returned to flight status?

The Committee found that NASA's drive to achieve a launch schedule of 24 flights per year created pressure throughout the agency that directly contributed to unsafe launch operations. The Committee believes that the pressure to push for an unrealistic number of flights continues to exist in some sectors of NASA and jeopardizes the promotion of a "safety first" attitude throughout the Shuttle program.

The Committee, the Congress, and the Administration have played a contributing role in creating this pressure. Congressional and Administration policy and posture indicated that a reliable flight schedule with internationally competitive flight costs was a near-term objective.

Pressures within NASA to attempt to evolve from an R&D agency into a quasicompetitive business operation caused a realignment of priorities in the direction of productivity at the cost of safety.

NASA management and the Congress must remember the lessons learned from the Challenger accident and never again set unreasonable goals which stress the system beyond its safe functioning.

The Committee commends the work of the Rogers Commission and its supporting panels at NASA. Their investigation and the reports that document their efforts are very broad in scope and exceptionally detailed considering the time that was available to accomplish their task.

As a rule, the Committee agrees with the findings reached by the Rogers Commission. However, there are areas where the Committee either disagrees with a Rogers Commission finding or with the relative importance that the Rogers Commission attached to that finding.

Like the Rogers Commission, the Committee concluded that the Challenger accident was caused by a failure in the aft field joint on the right-hand Solid Rocket Motor. Additionally, we agree with the Rogers Commission that this tragic accident was not caused by the Orbiter, the Space Shuttle Main Engines, the External Tank, the onboard payloads, the ground support equipment, or the other elements of the Solid Rocket Boosters. We also agree that the failure of the joint was due to a faulty design, and that neither NASA nor Thiokol fully understood the operation of the joint prior to the accident. Further, the joint test and certification programs were inadequate, and neither NASA nor Thiokol responded adequately to available warning signs that the joint design was defective.

In concurrence with the Rogers Commission, the Committee confirms that the safety, reliability, and quality assurance programs within NASA were grossly inadequate, but in addition recommends that NASA review its risk management activities to define a complete risk management program. The Committee also agrees that a thorough review must be conducted on all Criticality 1 and 1R items and hazard analyses; a study should be conducted on how to provide Space Shuttle crews with a means of escape during controlled gliding flight; and NASA's Shuttle management structure, safety organization, communications procedures, and maintenance policies should be carefully scrutinized and improved.

In other areas, the Committee reached somewhat different conclusions than the Rogers Commission:

> The Rogers Commission concluded that NASA's decision-making process was flawed. The Committee does agree that the Marshall Space Flight Center should have passed along to higher management levels the temperature concerns that Thiokol engineers raised the night before the launch of Mission 51-L. However, the Committee feels that the underlying problem which led to the Challenger accident was not poor communication or inadequate procedures as implied by the Rogers Commission conclusion.[1] Rather, the fundamental problem was poor technical decision-making over a period of several years by top NASA

[1] For the purpose of this report, a procedure is a formal set of instructions designed to guide and assist in the performance of a technical or management function.

and contractor personnel, who failed to act decisively to solve the increasingly serious anomalies in the Solid Rocket Booster joints.

Information on the flaws in the joint design and on the problems encountered in missions prior to 51-L was widely available and had been presented to all levels of Shuttle management. Despite the presence of significant amounts of information and the occurrence of at least one detailed briefing at Headquarters on the difficulties with the O-rings, the NASA and Thiokol technical managers failed to understand or fully accept the seriousness of the problem. There was no sense of urgency on their part to correct the design flaws in the SRB. No one suggested grounding the fleet, nor did NASA embark on a concerted effort to remedy the deficiencies in O-ring performance. Rather, NASA chose to continue to fly with a flawed design and to follow a measured, 27-month, corrective program.

The Committee has more concerns than those expressed by the Rogers Commission about the relative safety of the Space Shuttle Main Engine. We are impressed by the sophistication and performance of the Main Engine, but are concerned that it may have inadequate safety margins to ensure continued safe operation. The Committee is also concerned by the presence of persistent operating problems with the engine (e.g., cracked turbine blades and defective hydraulic actuators and temperature sensors), and believes that NASA should give serious consideration to not allowing the Main Engine to be operated (except in emergency situations) at a thrust level greater than the standard 104 percent. On the other hand, should NASA determine that a higher engine thrust setting is needed for programmatic reasons, the Committee believes that the space agency should take whatever actions are required to ensure that adequate operating margins are present to maintain safety.

The Committee has gone beyond the Rogers Commission in recommending a new system specification to overcome the inadequacies of the landing gear, tire, wheel, brake and nose wheel steering systems. The Committee also concluded that orbiter landings appear to be high risk even under ideal conditions, which seldom occur.

The Rogers Commission stated that "there appears to be a departure from the philosophy of the 1960s and 1970s relating to the use of astronauts in management positions."[2] In contrast, after taking testimony from several former and current astronauts, the Committee could find no evidence that astronauts are denied the opportunity to enter management if they so choose. On the other hand, prior to the STS 51-L accident, astronauts were not encouraged to enter management.

---

[2] Rogers Commission Report, Volume I, p. 199.

In still other areas, the Committee has raised concerns that do not appear to have been addressed sufficiently by the Rogers Commission. We are concerned that:

There are numerous other recurrent hardware problems that are either not fully understood by NASA or have not been corrected.

The existing internal communication system is disseminating too much information, often with little or no discrimination in its importance. Accordingly, recipients have difficulty "separating the wheat from the chaff."

Existing contract incentives used by NASA do not adequately address or promote safety and quality concerns—most emphasis is placed on meeting cost and schedule requirements.

NASA does not yet understand how or why the deficiencies in Solid Rocket Motor testing and certification went undetected in spite of the very comprehensive processes and procedures used by the agency to conduct and oversee these activities. The Committee is concerned that without such an understanding, NASA will not be able to protect against a similar breakdown in its system of checks and balances in the future.

The Committee has concerns regarding the safety of the Filament Wound Case Solid Rocket Booster now under development by NASA, and recommends that the agency consider moving the heaviest Space Shuttle payloads to expendable launch vehicles so that there will be no need to use Filament Wound Case Boosters.

The Committee is not assured that NASA has adequate technical and scientific expertise to conduct the Space Shuttle program properly. NASA has suffered staffing reductions in key areas over several years. Moreover, it loses a significant number of technical/scientific personnel due to an imbalance between the government salary schedule and that of the private sector. The salary structure also inhibits NASA's ability to recruit top technical talent to replace its losses. The record is not sufficient to warrant a formal finding on this matter. However, the Committee intends to conduct an in-depth review of NASA technical ability in the next Congress.

On July 14, 1986, NASA submitted to the President a report on what actions the space agency plans to take in response to the recommendations of the Rogers Commission.[3] The Committee believes that the plans contained in this report are a step in the right direction. When fully implemented, these plans should substantially improve the safety of Space Shuttle flight operations. The Committee also endorses NASA's decision to move the proposed date for the next Space Shuttle launch beyond June 1987. This is a realistic and responsible decision that has removed some unnecessary pressure

[3] NASA, "Report to the President: Actions to Implement the Recommendations of the Presidential Commission on the Space Shuttle Challenger Accident," July 14, 1986. (Hereafter referred to as NASA Response to Rogers Commission.)

from the government and contractor personnel who must ensure that all hardware will be in readiness to reinstitute safe flight operations.

Throughout the remainder of this report, the Committee addresses dozens of specific issues that relate to the Challenger accident. The Committee makes many recommendations for actions to be taken on the part of NASA to correct the problems that we have identified. The Committee directs NASA to report back to us by February 15, 1987, on how it is responding to each recommendation contained in this report.

In closing, the Committee would like to state that it continues to believe in and remains committed to a vigorous civilian space program. The Committee also continues to believe that the Space Shuttle is a critically important element of that program. The Committee's purpose, as NASA's primary overseer in the House, must be to monitor, understand, and help correct where necessary the patterns in NASA which lead to weakened and ineffective operation.

We are at a watershed in NASA's history and the Nation's space program. NASA's 28-year existence represents its infancy. We must use the knowledge and experience from this time to insure a strong future for NASA and the U.S. space program throughout the 21st century.

This Committee has long been proud of the many awe-inspiring achievements of NASA and understands the importance of NASA's programs to the future well-being of this country. We as a Committee have perhaps exhibited the human inclination to accept the successful completion of a flight or event as an indication of the overall strength of all aspects of its planning and execution. Perhaps it is arrogant to dissect and interrogate relentlessly projects and programs that bring home repeated A's for achievement and accomplishment. However, all of us—NASA, the Committee, the Congress and the Nation—have learned from the Challenger tragedy that it is wisdom to do so, and it is a reflection of respect for the human fallibility that we all possess.

We have no doubt that through the hard work and dedication of the men and women at NASA and its supporting contractors, the Space Shuttle will be safely returned to flight status—and will once again continue to impress people around the world with its many important accomplishments.

As has been said many times since the January 28th tragedy, space flight is a high risk undertaking. The Committee accepts this fact and applauds those men and women who, in spite of this risk, have chosen manned space flight as a career. Though we grieve at the loss of the Challenger crew, we do not believe that their sacrifice was in vain. They would not want us to stop reaching into the unknown. Instead, they would want us to learn from our mistakes, correct any problems that have been identified, and then once again reach out to expand the boundaries of our experience in living and working in outer space.

III. COMPILATION OF ISSUES, FINDINGS, AND RECOMMENDATIONS

This compilation is taken from the body of the report. In order to facilitate the reader's ability to refer to specific sections within the report, the outline in the following compilation corresponds to the Table of Contents and the body of this report:

V. THE ACCIDENT

A. INTRODUCTION

Discussion Only

B. SOLID ROCKET MOTORS

1. History

Issue

Was there sufficient time to correct the problems with the Solid Rocket Motor?

Findings

1. Problems with the joints which connect the Solid Rocket Motor casings were recognized for many years. While attempts were made to correct these problems, the measures taken were insufficient to provide a reliable joint.

2. The joint seal problem was recognized by engineers in both NASA and Morton Thiokol in sufficient time to have been corrected by redesigning and manufacturing new joints before the accident on January 28, 1986. Meeting flight schedules and cutting cost were given a higher priority than flight safety.

2. Summary of Casing Joint Design

Issue

Why did the aft field joint between the steel containers that hold the Solid Rocket Motor propellant fail to contain the burning gases of the propellant during lift-off and flight operations?

Findings

1. The design of the field joint was unsatisfactory and could not reliably contain the burning propellant gases under the range of operating conditions to be expected during the lift-off and flight phases.

2. The O-ring materials and putty used in the design of the joint were unsatisfactory as used on the Shuttle, particularly during the winter months. Furthermore, neither NASA nor its contractor, Morton Thiokol, can adequately control the quality or consistency of these kinds of materials, which are made from recipes known

only by the manufacturer and which can be changed without certification and approval.

Recommendations

1. NASA should write and issue a new and more accurate performance specification which would cover the full range of thermal and structural requirements for the Solid Rocket Motors, with an adequate factor of safety for unusually low temperatures.
2. The Committee concurs with the Rogers Commission Report Recommendations on new joint design, but believes it is more appropriate to be more explicit in identifying the weaknesses in the joint design that need correction.
3. The field joints of the Solid Rocket Motors should be redesigned to account for the following features while providing a significant factor of safety:
   a. Movement in the joint,
   b. Proper spacing between tang and clevis,
   c. Seals made to withstand high and low temperatures under all dynamic thermal and structural loadings,
   d. Adequate sealing without the use of putty,
   e. Protection against insulation debonding and propellant cracking.

3. Testing and Certification

Discussion Only

4. Manufacturing

Discussion Only

5. Stacking Operations

Issue

Was there any damage to the casing joints or contamination that occurred during the stacking operations when the Shuttle was assembled in the Vehicle Assembly Building (VAB) that could have contributed to the failure?

Finding

There was no evidence of joint contamination, fracture or other damage from foreign objects or due to casing ovality that contributed to the joint failure. Although certain problems occurred during stacking and the procedures were violated once, there was no evidence that these events contributed to the Flight 51-L accident.

6. Summary of Launch Operations

Issue 1

How was the decision to launch STS 51-L arrived at and why was it wrong?

Findings

1. The Flight Readiness Review for STS 51-L was conducted in accordance with established procedure.
2. The decision to launch STS 51-L was based on a faulty engineering analysis of the SRM field joint seal behavior.

3. Compounding this erroneous analysis were serious ongoing weaknesses in the Shuttle Safety, Reliability and Quality Assurance Program which had failed to exercise control over the problem tracking systems, had not critiqued the engineering analysis advanced as an explanation of the SRM seal problem, and did not provide the independent perspective required by senior NASA managers at Flight Readiness Reviews.

4. The initial response of Marshall managers to the attempts of Thiokol engineers to raise the issue of temperature effects on the SRM seals caused Thiokol management to discount proper technical concerns and engineering judgment in their recommendation to launch.

5. The Director of Marshall's Shuttle Projects Office may have violated NASA's Flight Readiness Review policy directive by failing to report the results of the January 27th teleconference to the Associate Administrator for Space Flight.

6. The decision of the STS Program Manager to launch despite the uncertainty represented by ice on the Fixed Service Structure was not a prudent effort to mitigate avoidable risks to the Shuttle.

7. The Launch Director failed to place safety paramount in evaluating the launch readiness of STS 51-L.

8. No launch should have been permitted until ice was cleared from the platform leading to the pad escape system.

9. Ice Team personnel and Rockwell contractors properly conveyed their inability to predict the post-ignition behavior of ice.

10. Post-flight analysis indicates that ice did not exhibit the behavior predicted by analysis, and that ice traversed a distance sufficient to strike the Shuttle during liftoff.

11. Failure to enforce a clear requirement for definite readiness statements contributed to failures in communication between NASA and its contractors during launch preparations.

Issue 2

Should firing room personnel be allowed to waive launch commit criteria or equipment redlines during a launch countdown without a well-developed technical reason for doing so?

Finding

NASA's management waived its own launch commit criteria on January 28, 1986, without a valid technical reason for doing so.

7. Retrieval, Transportation and Refurbishment

Issue

Were the motor casings used on STS 51-L damaged as a result of the retrieval, transportation and refurbishment operations following previous launches?

Finding

There was no evidence of damage to the casings or joint due to prior use or preparation for reuse.

C. EXTERNAL TANK

Issue

The External Tank was obviously involved in the accident. Was that involvement a cause or an effect?

Findings

1. The Committee adopts the "Finding" of the Rogers Commission that: "A review of the External Tank's construction records, acceptance testing, pre-launch and flight data and recovered hardware, does not support anything relating to the External Tank which caused or contributed to the cause of the accident."
2. The External Tank ruptured under the forces of a failed Solid Rocket Booster motor. These forces were far outside of any possible design considerations that could have been applied to the External Tank.

D. CREW SURVIVAL

Issue

Was the accident of STS 51-L on January 28, 1986, survivable?

Finding

In the case of the tragic loss of the Space Shuttle Challenger and her crew on January 28, 1986, the Committee is convinced that the accident was not survivable.

E. SABOTAGE

Issue

Could the accident have been caused by sabotage, terrorism, or foreign covert action?

Finding

The Committee is convinced that there is no evidence to support sabotage, terrorism or foreign covert action in the loss of the Challenger.

F. ADDITIONAL AVENUES OF INVESTIGATION

Issue

Could the accident have been caused by some failure other than failure of the joint between the casings?

Finding

As of September 15, 1986, the Committee has not found any credible evidence to support any cause of the Challenger accident, other than the failure of the aft casings joint in the right-hand Solid Rocket Booster. Nor has there been any substantial evidence of a secondary or parallel failure on Flight 51-L.

VI. Discussion of Critical Issues

A. TECHNICAL ISSUES

1. Hardware Development and Production

a. Problems in Hardware Certification

Issue 1

Have all elements of Space Shuttle flight hardware been adequately certified?

Findings

1. The overall design and certification processes prescribed by NASA for each major element of Space Shuttle flight hardware are very comprehensive.
2. Prior to the STS 51-L accident, in spite of the comprehensive nature of NASA's prescribed design and certification processes, insufficient testing had been conducted to permit an adequate understanding by either Morton-Thiokol or NASA regarding the actual functioning of the Solid Rocket Motor joint. Also, the Solid Rocket Motor had not been adequately certified to meet the natural and induced environmental conditions that are stated in NASA's design standards. The issue of whether or not standards were adequate is discussed in Section VII.
3. The deficiencies in Solid Rocket Motor testing and certification persisted in spite of many reviews of the program by panels of experts: (1) within the manufacturer; (2) within NASA; and (3) from independent, outside groups.
4. These deficiencies in testing and certification of one major element of the Space Shuttle system raise the possibility that other elements of flight hardware (or other sub-elements of the Solid Rocket Motor) could have similar deficiences.
5. If NASA is unable to explain why the deficiences in Solid Rocket Motor testing and certification went undetected by the existing comprehensive set of processes and procedures, the agency will not be able to protect against a similar breakdown in its system of checks and balances in the future.

Recommendations

1. NASA should devote more attention to determining why the deficiencies in Solid Rocket Motor testing and certification went undetected, so that appropriate action can be taken to uncover latent problems in existing hardware and to prevent similar problems in future development programs.
2. NASA and its contractors should thoroughly reassess the adequacy of all the testing and certification that has been conducted to date on each element of Space Shuttle flight hardware. Where deficiencies are found, they must be corrected.

Issue 2

Does the Space Shuttle Main Engine have adequate operating margins, and is the "fleet leader" concept adequate to ensure safe operation?

Findings

1. The Space Shuttle Main Engine is an impressive, technological achievement. However, it also is one of the higher risk elements of the Space Shuttle system. Anomalous component performance or premature engine shutdown could prove catastrophic to the Space Shuttle and its crew.
2. Some NASA officials familiar with the Space Shuttle Main Engine believe that it should be operated at a throttle setting of 109 percent only in an emergency; others believe the engine could be safely operated at 109 percent on a routine basis.
3. It is widely accepted that the Space Shuttle Main Engine would be safer if its operating margins (for temperature, pressure, operating time, etc.) were increased.
4. The Committee agrees with the sense of Dr. Feynman's concerns with respect to NASA's current "fleet leader" concept for certifying Space Shuttle Main Engine components, such as high pressure turbopumps, for flight.
5. On a case by case basis, NASA regularly violates its own certification requirements by permitting individual engine components to be used for flight even though they have accumulated an operating time in excess of 50 percent of the two fleet leaders (i.e., in violation of the "2X" rule).

Recommendations

1. NASA should continue its active development program for the Space Shuttle Main Engine. The program should be focused more on increasing operating margins.
2. Because of the safety concerns raised by some knowledgeable officials, NASA should give serious consideration to restricting use of the 109 percent engine throttle setting to emergency situations only. If NASA decides that it needs to use the 109 percent throttle setting for other than emergency situations, the space agency should take whatever actions are required to ensure that adequate margins are present to maintain safety.
3. NASA should closely scrutinize each of the concerns raised by Dr. Feynman regarding the agency's "fleet leader" concept for certifying Space Shuttle Main Engine components. The agency should also closely reassess its practice of selectively violating its "2X" rule for some Main Engine flight hardware elements.

b. Recurrent Hardware Problems

Issue

What resolutions of inadequacies revealed in the landing gear, tires, wheels, brakes, and nose wheel steering of the landing and deceleration system are required?

Findings

1. The Orbiter landing gear, tires, wheels, brakes, and nose wheel steering, as a system, is experimental, designed to criteria outside any other experience, and uses unique combinations of materials. The original design performance specifications for speed and landing weights are routinely exceeded. The original design

did not consider asymmetrical braking for cross wind steering as the normal case, although it has become standard practice. Stresses which were not taken into account in the design have surfaced in as yet a very small real world sample.

2. As a consequence, Orbiter landings appear high risk even under ideal conditions, which seldom occur. Exceptional procedural and skill demands are placed upon the pilots to nurse the brakes and tires through every landing. Landing rules have had increasing constraints imposed that hamper operational flexibility and usefulness of the Orbiter.

3. Brake and tire damage have been evident since early on in the program. The Rogers Commission seems very correct in finding the current landing gear system unacceptable. Resolution of landing gear system problems can no longer be put off.

Recommendations

The Committee recommends that NASA:

1. Assemble all of the fragmented studies, analyses, and conclusions on landing gear problems and integrate them into one engineering description of the system as it is now intended to be used. This should include consideration of the basic strength of the struts themselves and their attachments.

2. Write a new system specification and match the proposed design improvements to an acceptable reliability and certification specification.

3. Design a test and certification program adequate to meet criteria to fly and to continue well into future operations until understanding and confidence in the landing gear system is attained.

4. In anticipation of requirements for a new brake specification, accelerate a program to provide:

Increased brake mass and/or heat sink,

Substantial increase in energy absorption,

Evaluation which weighs the experimental nature of the proposed 65 million foot pound carbon brake and its impact on the system against the penalty of weight of known materials (e.g. steel) for operational confidence.

5. Write updated subsystem specifications to upgrade the landing gear system to acceptable levels of performance to respond to the Rogers Commission's recommendations.

Issue 2

What actions should be taken relative to other recurrent problems with flight hardware?

Finding

There have been many instances of in-flight anomalies and failures of other elements of Space Shuttle hardware, some involving mission critical pieces of equipment. Some of these past problems have been corrected while others have not.

Recommendation

NASA should ensure that before reinstituting Space Shuttle flight operations, it fully understands and has corrected all in-

stances of serious in-flight anomalous behavior or failures involving mission critical pieces of flight hardware.

c. Other Engineering Concerns

Issue

What action should be taken relative to other engineering concerns regarding critical elements of Space Shuttle flight hardware?

Finding

In recent years, serious engineering concerns have been raised regarding the safety of some elements of Space Shuttle flight hardware, such as the 17 inch flapper valve and the heat exchanger feeding the liquid oxygen tank.

Recommendations

1. NASA should ensure that, as a part of its current review of Space Shuttle safety, it identifies, thoroughly evaluates, and then takes appropriate action on all serious engineering concerns raised regarding mission critical elements of Space Shuttle flight hardware.
2. NASA should give special attention to both the cost and risks of using Filament Wound Case Solid Rocket Boosters for very heavy Space Shuttle payloads versus the cost and programmatic impacts of simply transferring those payloads to expendable launch vehicles.

d. Desirable Tests Not Yet Approved

Issue 1

Is the current ground test program for the SSME adequate to provide a complete understanding of the engine's operating characteristics and safety margins?

Findings

1. The Committee supports the Findings and Conclusions of the Development and Production Team concerning the SSME, particularly the concern that "Hardware availability and the potential of damage to hardware and facilities resulting from tests malfunctions have constrained . . . [full margin] . . . testing during the ground test program."
2. The Committee shares Dr. Feynman's concern that there has been a slow shift toward decreasing safety in the SSME program.
3. There is not a sufficient understanding of SSME blade cracks and fractures.

Recommendations

1. The Committee concurs with the Development and Production Team conclusion that overtesting, limits testing, and malfunction-testing in the SSME program should be re-emphasized to demonstrate full engine capability.
2. NASA should prepare and submit to the Committee a cost-benefit analysis of testing a SSME to destruction including: (a) uti-

lizing additional SSME test stands; (b) utilizing additional hardware for the ground test program; and (c) the value of such a test.

3. A vigorous study of fracture behavior should be conducted to minimize the hazard of cracked SSME blades and to increase the reliability and safety margin of blades. New blades and/or new policies for duration of blade use should be incorporated prior to the next Shuttle launch.

Issue 2

Is the leak/combustion threat of the External Tank's hydrogen pressure valve a hazard warranting testing?

Findings

1. The Committee supports the Rogers Commission concern regarding the hazard posed by the liquid hydrogen vent and relief valve.

2. The Committee supports the intent of the ET prime contractor, Martin Marietta, to pursue outdoor wind tunnel testing to eliminate the liquid hydrogen vent/relief valve hazard.

Recommendation

NASA, in conjunction with the appropriate contractor, should consider designing and conducting an ET liquid hydrogen leak/burn test to determine if corrective actions should be taken prior to the next Shuttle flight.

Issue 3

Does the present Range Safety System (RSS) on the External Tank present an unreasonable risk?

Finding

There is substantial controversy over the relative benefits and risks of the present RSS on the External Tank.

Recommendation

The Committee believes the Administrator should prepare and submit to the Committee a comprehensive review of RSS requirements.

e. Production/Refurbishment Issues

Issue 1

Should 100 percent X-ray inspection of the propellant and insulation for the Solid Rocket Motors (SRM) be resumed?

Findings

1. Previous X-ray inspection led to only one SRM being rejected for Shuttle use.

2. There is no non-destructive inspection method which can guarantee a defect-free SRM. X-ray inspection cannot detect "kissing" voids in which the SRM insulation is touching the SRM steel casing but is not bonded to it. Debonded insulation at the end of an SRM segment could provide burning propellant gases with a path to the SRM steel casing and could result in loss of vehicle and

crew. X-ray inspection can detect propellant cracks and large voids which if undetected could also result in a catastrophic situation.

3. Although there is no guarantee that X-ray inspection has been a particularly effective method of detecting propellant and insulation SRM flaws, it remains one of the best available methods to monitor the SRM manufacturing process.

Recommendations

1. NASA should consider reinstating full X-ray inspection of the propellant and insulation for all motors used on succeeding flights until new, more accurate inspection methods can be developed and implemented and there is unquestionable confidence in the SRM production process.

2. NASA, in conjunction with the appropriate contractors, should investigate the development of new, more accurate inspection techniques which can detect "kissing" voids and other potential defects that cannot be detected by X-ray inspection.

Issue 2

Are all production and other activities involving Criticality 1 and 1R hardware at prime and secondary contractor facilities labeled as "critical" processes?

Findings

1. Critical processes are formally identified and controlled by NASA. All processes are classified and controlled by the contractor's Process Change Control Board.

2. The O-ring used in the case joint is critical to the sealing integrity of the joint, yet it is not designated as a "critical" process by either the Parker Seal Co. or Hydrapack, the manufacturer and supplier respectively. This raises the possibility that other Criticality 1 and 1R hardware components are also not appropriately designated by their manufacturer as "critical" processes.

Recommendations

1. NASA should require the manufacture of critical items, such as the O-rings, to be designated "critical" processes. Contractors should formally notify their employees involved in critical manufacturing processes of the serious nature of particular production processes.

2. NASA should conduct a thorough review to ensure that all manufacturing processes involving Criticality 1 and 1R hardware components of prime and secondary contractors are appropriately designated "critical" processes.

Issue 3

Do O-ring repairs compromise safety?

Finding

The Committee supports the Development and Production Team Finding and Conclusion that the "limit of five repair joints per O-ring is an arbitrary number" and that "repair of inclusions and voids in the rubber . . . appears to be an area of potential problem."

Recommendation

NASA should review its O-ring repair policy and contractor repair practices in terms of their effects on O-ring performance and safety. Such review should be completed prior to the resumption of Shuttle flights if, as anticipated, the new SRB joint design uses O-rings.

Issue 4

What impact does growth of SRM case size have upon booster and Shuttle performance and safety?

Finding

The Committee concurs with the Development and Production Team Finding that "Remeasurement of two used SRM case segments indicated both tang and clevis sealing surfaces have increased in diameter beyond the anticipated design limits."

Recommendation

NASA and the appropriate contractor should resolve through analysis and testing prior to the next Shuttle flight the cause of SRM case size growth and its impact upon booster and Shuttle performance, reliability of refurbished SRM case segments, and safety.

f. Review of NASA's Redesign/Recertification Plan

Issue

Is NASA's SRM redesign and hardware recertification plan a viable and realistic one which will result in a safer, more reliable Space Transportation System?

Findings

1. NASA's SRM redesign plan is a step in the right direction. Moving the proposed launch date beyond June 1987 is a responsible and realistic decision. The membership of the SRM Redesign Team is representative of qualified individuals in and outside of NASA. With the expert assistance of the specially appointed National Research Council (NRC) Independent Oversight Group, the new SRM design should be a significantly safer and more reliable Shuttle element.

2. NASA's current hardware recertification plan is also a step in the right direction. The use of independent review contractors distinguishes this recertification plan from earlier reviews. However, given the failure of previous reviews to discover the deficient SRB joint certification, the Committee is concerned there is still the possibility that the recertification effort may not reveal other certification deficiencies, if indeed they exist. The plan also raises concern about the qualifications of independent reviewers to evaluate certain elements given the uniqueness of particular Shuttle components.

3. The joint was never fully tested as a separate element of the SRM. The various forces that act on the joint during stacking, launch, and flight are difficult, if not impossible, to duplicate in a

test of the joint under all conditions that could be experienced during launch and flight.

4. It is unclear what function the new Safety Office will perform in the redesign of the SRB field joint and other critical elements of the Shuttle, as well as NASA's recertification plan.

Recommendations

1. The Committee recognizes the national need to return the Shuttle to flight status as soon as reasonably possible. As noted in NASA's July 14, 1986, report to the President, safety will determine the launch schedule. However, NASA should consider the proposed launch date of early 1988 as a flexible one which should be slipped further if necessary. The Shuttle should not be launched again until NASA can assure that safety criteria have been met.

2. In establishing a test program to certify the new Solid Rocket Motor design, NASA should consider the feasibility of including in combination and in the proper sequence all of the thermal and structural loads expected to be experienced by the Solid Rocket Motor during ignition, lift-off, and flight.

3. The independent review contractors participating in the hardware recertification plan should utilize sufficient specific technical expertise to insure adequate recertification of all elements of the STS.

4. The Committee requests that the new Office of Safety, Reliability and Quality Assurance conduct an independent assessment of the SRB field joint redesign efforts. In addition, the new office should also be integrally involved in reviewing all other critical component redesign efforts and NASA's recertification plan.

2. Operations

a. Shuttle Processing Issues

Issue

In 1983, NASA consolidated fifteen separate contracts and awarded a single Shuttle Processing Contract (SPC) encompassing all ground processing related to launch and landing of the Space Shuttle. There are two issues associated with this contract: (1) How should is the concept of a unified SPC; and (2) How well has the SPC contractor actually performed? A related issue is the quality of essential logistical support, especially spare parts, provided to the contractor by NASA.

Findings

1. Performance under the SPC has improved since the inception of the contract. However, up to the time of the Challenger accident, contractor performance continued to be plagued by excessive overtime, persistent failures to follow prescribed work procedures, and inadequate logistical support from NASA.

2. High overtime rates have hampered SPC performance. Overtime rates had increased significantly during the six months prior to the Challenger launch, to the point that critical personnel were working weeks of consecutive workdays and multiple strings of 11 and 12-hour days. Fatigue resulting from work patterns of this sort can constitute a threat to safety. In fact, worker fatigue was a con-

tributing factor in a mission-threatening incident on Flight 61-C, the mission immediately prior to the January 28 Challenger launch.

3. There are numerous documented cases where contractor employees failed to comply with guidelines for carrying out assigned duties, including specific "Operations and Maintenance Instructions" (OMIs). Such failures contributed to both of the major mishaps in 1985 involving Shuttle processing—namely, the November 8, 1985, "handling ring" episode which led to significant damage to a Solid Rocket Motor segment slated for use on STS 51-L, and the March 8, 1985, "payload bay access platform" episode which led to significant damage to bay payload bay door. Failure to follow an OMI also led to improper (and mission-threatening) handling of the hydrogen disconnect valve during the 51-L launch operations. All of these incidents show a lack of discipline, both with respect to following prescribed procedures and with respect to reporting violations of these procedures.

4. At the time of the Challenger accident, the lack of spare parts caused a degree of cannibalization (i.e., the removal of a part from one Orbiter to satisfy a need for a spare part on another Orbiter), which was the highest in the history of the Shuttle program and which was a threat to flight schedule and flight safety. Excessive cannibalization leads to multiple installations, retesting, added documentation, delayed access to parts, and increased damage potential. As a result, cannibalization contributes directly to excessive overtime.

5. There is no clear evidence whether or not greater involvement of the development contractors would improve Shuttle operations.

Recommendations

1. Because of the serious quality and safety concerns surrounding the contract, NASA should conduct a careful review of Shuttle processing, the SPC contract, and the relationship of flight hardware contractors and report its findings, recommendations, and proposed contract modifications to the Committee. NASA's reexamination should include a comparison of efficiency and safety under the SPC versus efficiency and safety during pre-1983 Shuttle processing operations, which heavily involved the development contractors.

2. NASA should examine the issues of spares availability and cannibalization and provide the Congress with a management and budgetary plan for correcting previous logistical problems.

3. NASA should stop routine cannibalization and develop guildelines (including appropriate control and review procedures and roles for the SR&QA office) governing permissible cannibalization.

4. The Committee recommends that NASA provide its re-invigorated safety office with the authority to enforce scheduling that leads to safe overtime rates.

b. Pressures on Shuttle Operations

Issue

Was NASA under pressure to fly more flights? How did this pressure originate? Will it recur?

Findings

1. The Congress and the Executive Branch jointly developed the policy that the Space Shuttle should, in a reliable fashion and at an internationally competitive cost, provide for most of the Free World's space launch needs. By and large, both Branches failed to appreciate the impact that this policy was having on the operational safety of the system.

2. NASA was under internal and external pressure to build its Shuttle flight rate to 24 per year, primarily to reduce costs per flight, but also to demonstrate and achieve routine access to space. NASA has never achieved its planned flight rate.

Recommendations

1. NASA must not attempt to achieve a flight rate beyond that which (1) can be supported by the budget and staff resources available; and (2) is consistent with the technical maturity of the Shuttle and the flexibility desired and needed in scheduling payloads. Management should ensure efficient use of resources but should not impose a flight rate on the system.

2. Once operation of the Space Shuttle resumes, the Committee should maintain a close and continuous oversight of Shuttle flight rate, planning, and operations. The Committee should ensure both that flight rate flows logically from the resources provided and that flight safety is not compromised beyond acceptable limits.

c. Impact of Pressures on Shuttle Operations

Issue

Did operating pressures adversely affect the safety of the Shuttle program?

Findings

1. The pressure on NASA to achieve planned flight rates was so pervasive that it undoubtedly adversely affected attitudes regarding safety.

2. The pressure to achieve planned flight rates was compressing mission preparation as earlier missions were delayed due to unforeseen problems. Had the accident not occurred there would soon have been a collision between planned launch dates and mission preparation needs which could not have been met by overtime, cannibalization, or other undesirable practices. Operating pressures were causing an increase in unsafe practices.

3. The schedule of payloads planned to fly on the Shuttle (the manifest) was frequently changed. Each change rippled through the NASA Shuttle organization and through the manifest and, especially if made shortly before launch, would increase the demands on personnel and resources in order to achieve the planned flight rate.

4. The Space Shuttle has not yet reached a level of maturity which could be called operational as that term is used in either the airline industry or the military. Each Shuttle flight is fundamentally unique, and requires unique preparations. Therefore, small changes in a mission can cause significant perturbations of mission planning and crew training.

Recommendations

1. The new Associate Administrator for Safety, Reliability and Quality Assurance must assure that any pressures to increase the Shuttle flight rate do not adversely influence mission preparation. The Associate Administrator must have the authority not only to stop a particular flight, e.g., at a Flight Readiness Review, but to stop the whole mission planning process if necessary.

2. Where appropriate, NASA should take steps to make the mission planning process standard and routine to reduce the time and resources needed to plan a mission. Before requesting more resources for the existing mission planning process (manpower, facilities, equipment), NASA should identify ways to improve the process.

d. Other Safety Issues

Issue 1

What is the criticality of landing safety associated with programmed and abort landing sites and their local characteristics?

Findings

1. The Committee finds that many of the normal and abort landing safety problems will be alleviated when the Rogers Commission's and the Committee's (Section VI. A. 1. b., this report) recommendations to upgrade the landing gear system are implemented. When the landing gear system is understood, straightforward calculations and operational rules will determine acceptable runway dimensions and conditions.

2. The Committee found no reason to fault NASA's current procedure on launch constraints based upon operational judgment and conservative rules on local conditions at planned abort and landing sites. However, since an obvious finding is that the Orbiter is a developmental system, it is axiomatic that unanticipated "dicey" circumstances will arise.

3. It was found that for the least landing gear system stress, runway preference is Edwards Air Force Base (EAFB) (concrete), KSC, and Rogers Dry Lake (EAFB "lake bed") in that order. No reason was found to invalidate the KSC runway design. The reasons for the "dry" course surface still prevail over concern about wear on tires designed for one landing. Additional constraints at KSC because of lesser lateral stabilized overrun area may be needed to bring its safety to the level of the EAFB runway.

4. The NASA Landing Safety Team's proposal to provide standard landing aids and arresting barriers at all sites and their emphasis on runway surface characteristics for repetitive tire use takes on a new dimension that is in addition to the Rogers Commission's recommendations.

5. Weather, by far, is the most significant factor governing operational decisions, Orbiter damage, and landing safety. The constraint is simply that acceptable weather must be forecast with confidence within the time frame needed. Ultra-conservative rules prevail because of the predictable unpredictability of Cape weather. New and innovative local weather analysis and forecasting re-

search is a high priority. The African Coast and southwestern United States sites enjoy more stable and predictable weather.

Recommendations

The first priority to achieve an acceptable degree of landing safety and to have a sensible base to work from for improvement is to implement the recommendations of the Rogers Commission and the Committee on the landing gear system improvements to attain an operational capability. Then:

Instrument the system, and schedule all landings at Edwards runway for systematic concurrent testing until the landing gear system is understood.

Write a clean sheet set of rules based on results.

Determine the risk of accident with the B–747 Shuttle Carrier Aircraft (SCA) and its impact upon the Shuttle program.

Extend every reasonable effort to assure a mission planning process to minimize the need for abort site landings.

Reevaluate and determine the degree of risk acceptable at abort site landings and bring abort site capability up to meet that risk level.

Expand astronaut matched team flight landing practice to cover all known exigencies. Propose additional training craft if necessary.

Join in a venture with NOAA to invent new technology and techniques to learn new ways to understand the dynamics of Cape Kennedy weather phenomena to supplant current inadequacy to forecast two hours ahead.

Issue 2

Has adequate provision been made for crew safety in case of in-flight emergencies? That is, has adequate provision been given to launch abort options and crew escape options?

Findings

1. Crew escape options were considered when the Shuttle was originally designed and the basic situation has not changed. Many initially attractive options do not significantly reduce risk to the crew either because they may not reduce exposure to the principal hazards or because they add risks of their own.
2. A crew escape system for use in controlled gliding flight might be feasible and worthwhile.
3. Crew escape during the ascent phase appears infeasible.
4. Launch abort during SRB burn appears impossible but it may be possible to decrease risk to the crew after SRB separation, primarily through mission design.

Recommendation

NASA should continue to respond to the recommendations of the Rogers Commission regarding (i) crew escape during controlled gliding flight and (ii) increasing the possibility of successful emergency runway landings. NASA should reexamine all crew survival options and report to the Committee on its findings.

B. MANAGEMENT ISSUES

1. Technical Management

a. Risk Management Issues

Issue

Is there a coordinated and effective risk management program in the NSTS?

Findings

1. NASA does not explicitly use a centralized program that coordinates all the factors that encompass an adequate risk management program.
2. As a result of the accident, NASA is reexamining the Failure Modes and Effects Analyses (FMEA) and Hazard Analyses (HA) to reassess risks associated with the designs of Shuttle subsystems.
3. NASA's lack of statistical data on the performance of certain components will limit the usefulness of sound engineering judgment in much the same way as it limits the usefulness of probabilistic risk assessment.

Recommendations

1. NASA should develop and provide to the Committee a description of an overall risk management program as it relates to the Space Shuttle. This effort should include a determination of whether or not a more centralized coordination of a risk management program and issuance of direct risk management guidance directives are needed.
2. NASA should review analytical methods utilized in the performance of risk assessment, including statistical analyses, trend analyses and probabilistic risk assessment methodologies to determine their applicability to the NSTS program. Assistance from the National Academy of Sciences, or other appropriate organizations with expertise in these matters, may be required to adequately perform this review.
3. NASA should review its certification testing to ensure that all critical items are adequately tested. Data obtained from these tests should be used when appropriate in conducting a formal risk assessment.

b. Launch Decision Process

Issue 1

Is the process for establishing launch constraints and dealing with them effective?

Findings

1. There is no clear understanding or agreement among the various levels of NASA management as to what constitutes a launch constraint or the process for imposing and waiving constraints.
2. Launch constraints were often waived after developing a rationale for accepting the problem rather than correcting the problem; moreover, this rationale was not always based on sound engineering or scientific principles.

64-420 O - 86 - 2

Recommendations

1. NASA should establish rigorous procedures for identifying and documenting launch constraints. The individual(s) responsible for implementing this procedure should be clearly identified, and well defined and understood criteria for waiving them should be established.
2. NASA should exercise extreme caution in waiving launch constraints before correcting the problem that led to the launch constraint. The rationale should be based on rigorous scientific/engineering analyses or tests and should be understood and accepted by the program manager.

Issue 2

Are the Launch Commit Criteria procedures adequate to ensure the safety of the mission?

Findings

1. The procedure used for developing launch commit criteria is systematic and thorough; however, violations of the criteria do not necessarily mean "no go". Therefore, NASA sometimes has relied on engineering judgments made during the terminal countdown in determining whether to launch.
2. Launch commit criteria were sometimes waived without adequate engineering analysis or understanding of the technical reasons for establishing the criteria.

Recommendations

1. NASA should review the launch commit criteria procedures, especially those for dealing with violations, to lessen the reliance on engineering judgments under stress.
2. When situations arise where "real time" judgments are unavoidable, NASA should adopt a more conservative approach to waiving previously established criteria. In no case should a criterion be waived without a thorough understanding of the rationale for the establishment of the criterion.

Issue 3

Are launch readiness review procedures and communications adequate?

Finding

The Committee finds that the review procedures and communications used to assure flight readiness were systematic, thorough, and comprehensive and provided ample opportunity for surfacing hardware problems prior to flight. Level I FRRs are usually recorded (audio); however, there is often no record made of other key prelaunch meetings.

Recommendation

NASA should make every reasonable effort to record meetings where key decisions might be made; in particular, all formal Flight Readiness Reviews, including the L–1 and the Mission Management Team meeting should be recorded, where feasible by video.

Issue 4

Was the failure to inform the Level I or Level II Program Managers of the Teleconference involving NASA and Morton Thiokol on the eve of the launch a factor in the decision to launch?

Findings

1. The Committee finds that Marshall management used poor judgment in not informing the NSTS Program Manager or the Level I Manager of the events that took place the night before the launch, specifically the stated concerns of the Thiokol engineers. However, the Committee finds no evidence to support a suggestion that the outcome would have been any different had they been told.

2. The Committee finds the efforts of Thiokol engineers to postpone the launch commendable; however, Thiokol had numerous opportunities throughout the normal flight readiness process following flight 51-C in January 1985 to have the new minimum temperature criteria established.

Issue 5

Do the principal contractors have an appropriate role in the launch decision making process?

Finding

The principal contractors have an active role throughout the decision making process right up to the launch; however, the lack of a firm requirement for their concurrence at the time of launch does partially relieve them of responsibility for mission success.

Recommendation

Principal contractors should be required to make a clear, unambigous statement concerning launch readiness just prior to launch.

Issue 6

Are astronauts adequately represented in the decision making process?

Finding

The astronauts believe they currently have the opportunity to make inputs into the process and are reluctant to assume a greater responsibility for the decision to launch.

c. Technical Expertise of Personnel

Issue

Does NASA have an adequate level of in-house technical expertise to manage the Shuttle Program properly?

Findings

1. During the last decade NASA has had significant decreases in manpower. A disproportionate reduction may have occurred in the safety, reliability and quality assurance staff at NASA headquarters and at the Marshall Space Flight Center. Additionally during the period preceding the Challenger accident, the Office of Space

Flight also suffered a decline in staff. The decreases may have limited the ability of those offices to perform their review functions.

2. The information presented to NASA headquarters on August 19, 1985 was sufficient to require immediate and concentrated efforts to remedy the joint design flaws. The fact that NASA did not take stronger action to solve this problem indicates that its top technical staff did not fully accept or understand the seriousness of the joint problem.

Recommendations

1. NASA should review the numbers and qualifications of key staff in technical and management positions and should consider additional training and recruitment of individuals to further the quality and safety of NASA's missions.

2. The Committee should maintain on-going oversight of this analysis and conduct an in-depth examination upon the conclusion of NASA's review.

d. Change Control Process

Issue 1

Has the pressure to maintain operational flight rates and schedules for the Shuttle compromised the hardware Change Control Process?

Findings

1. When NASA declared the Space Shuttle to be an operational system, additional pressure to increase flight rates impacted other aspects of the overall program such as the ability to implement, evaluate, test, and certify changes in hardware design.

2. As a result of attempting to operate the Shuttle at increased flight rates, controlling other aspects of the program such as the flight production process and manifest also became a more complex and difficult aspect of program administration.

Recommendations

1. NASA must reconsider its efforts to categorize the Shuttle as an operational transportation system.

2. The Configuration Management System designed to control such changes must be reexamined by NASA as to its effectiveness in assuring that all hardware changes take place in a safe and reliable fashion.

Issue 2

Is the change control process sufficiently defined for all elements of the Shuttle system?

Findings

1. The NSTS engineering and process change guidelines are, for the most part, sufficiently well-defined for the majority of the subsystems that comprise the Space Shuttle.

2. NASA gives the same level of scrutiny to changes involving a minor component (such as moving velcro strips in the Orbiter) as those involving mission critical elements of flight hardware.

Recommendation

NASA should review its change control process to determine the usefulness of differentiating between minor changes and significant changes.

2. Organization and Policy Management

a. Management Structure

Issue 1

Does the management of the Shuttle Program adequately define the lines of authority and are managers given authority commensurate with their responsibilities?

Finding

The management of the Shuttle Program is complex and diversified and it is not always clear who has authority or responsibility. NASA's "lead center" concept has resulted in placing the management of the program at JSC, one of three centers participating in the program; however, because Johnson does not have control of the other centers' resources, the NSTS Program Manager's authority to manage the program is limited and the responsibility is unclear.

Recommendation

NASA should restructure the Shuttle Program management to define clear lines of authority and responsibilities. This restructuring should take into account the special role each center must play and be especially sensitive to the need for the cooperation and support of all the participants to achieve a common goal. NASA should give special consideration to moving the Program Manager to NASA Headquarters to avoid the confusion and inter-center rivalry that result from having a large multi-center program managed out of one of the participating centers.

Issue 2

Are astronauts adequately represented in management?

Finding

The Committee finds no evidence that astronauts are denied the opportunity to enter management if they so choose.

b. Communication

Issue 1

Are there adequate opportunities to communicate problems within the Shuttle Program management structure?

Finding

There are many regularly scheduled meetings and telecons at all levels of management throughout the Shuttle Program. In addition, "special" meetings and telecons are routine. No evidence was found to support a conclusion that the system inhibited communication or that it was difficult to surface problems.

Issue 2

Is too much information being disseminated so that important information is lost?

Finding

Large amounts of information are disseminated on a routine basis, often with little or no indication of its importance to all of the recipients.

Recommendation

NASA management should review the process of providing information on significant actions so that awareness by concerned managers is assured.

Issue 3

Are communications filtered so that important information is prevented from reaching the decision makers?

Finding

NASA managers delegated the responsibility for making technical judgments to lower level managers or assistants. Therefore, the information that reached the top decision makers was "filtered" in that it was interpreted by others that were presumed to have more specialized experience or expertise in a given area. There is no evidence that middle level managers suppressed information that they themselves deemed to be significant. In fact, as discussed in the section on technical expertise, the failure was not the problem of technical communications, but rather a failure of technical decision making.

c. Safety, Reliability and Quality Assurance

Issue 1

Is NASA's decision to establish a new Office of Safety, Reliability and Quality Assurance appropriate and, if so, what should its role be?

Finding

The Committee finds that the Rogers Commission recommendation that NASA should establish an Office of Safety, Reliability and Quality Assurance that reports directly to the Administrator is indeed appropriate. However, it is not clear what the activities of this office will encompass.

Recommendations

1. The Associate Administrator for Safety, Reliability and Quality Assurance (SR&QA) should provide to the Committee the agency's draft plan delineating the organization, goals, implementation strategies and resource requirements of the Office of SR&QA.
2. After the Office of SR&QA is fully operational, the Committee will wish to continue oversight over its activities.

Issue 2

Has NASA applied sufficient resources to support adequate SR&QA efforts within the NSTS program?

Findings

1. The Committee finds that reductions in NASA civil service personnel that have occurred over the past decade have adversely impacted the agency's ability to maintain the appropriate level of oversight and control of the SR&QA activities within the NSTS.

2. NASA has become increasingly dependent upon outside SR&QA support from the Department of Defense (Defense Contract Administration Services (DCAS) and Air Force Plant Representative Office [AFPRO]) and contractors.

3. NASA has reduced or reassigned to other program areas in-house safety, reliability and quality assurance tasks such as testing, analyses and instrumentation and has reduced or shut down in-house facilities for performing SR&QA research and technology development. The degree to which these factors have adversely impacted the safety, reliability and quality assurance activities within the NSTS program has not been adequately assessed.

Recommendations

1. NASA should establish and maintain a strong and effective SR&QA program. Continuing support for such a program must come directly from the Administrator.

2. Although it is appropriate to establish strong contractor capabilities in the areas of SR&QA, the internal oversight responsibilities and coordination of SR&QA tasks must be the responsibility of NASA itself. In order to assure that the appropriate interfaces among the various subsystem elements that comprise the NSTS are maintained, a sufficient complement of NASA SR&QA management and support staff must be available to perform the necessary oversight and coordination tasks.

Issue 3

Are the responsibilities of safety engineers and design engineers adequately specified within NASA's "risk management" program?

Finding

The roles of safety, design as well as reliability engineers are not adequately and uniformly defined throughout the NSTS program. In some cases, the Committee learned that safety engineers were not participating in major decisions related to flights of the Shuttle.

Recommendation

It should be the responsibility of the new Associate Administrator for SR&QA to fully specify the roles of safety and reliability engineering as well as quality assurance personnel within the NSTS program, so that all critical aspects of the program and decisions related to the adequacy of hardware and subsystem performance are fully reviewed by these disciplines.

Issue 4

Does the SR&QA program require improved coordination between centers, contractors and NASA Headquarters?

Findings

1. Although guidelines have been published that describe the responsibility of contractors in the areas of SR&QA, NASA's guidelines do not adequately distinguish these various activities as distinct disciplines requiring specialized skills and centralized coordination.

2. In its review of the agency's reliability and quality assurance programs as they relate to the Space Shuttle, the Committee found there was little commonality among the cognizant officials at MSFC, JSC, KSC, and Headquarters in the perception of the various responsibilities associated with these separate and distinct disciplines.

Recommendations

1. It is important that a clear delineation of responsibilities for the separate SR&QA disciplines be appropriately documented. It is also essential that the relative importance of each of the three separate disciplines be established as an integral part of the NSTS program. These functions are the responsibility of NASA Headquarters.

2. NASA must carefully review the staff and resources devoted to the SR&QA function within NASA and contractor organizations for adequacy. The Administrator shall report to the Committee with his findings and recommendations.

d. Contractor Incentives

Issue

Key Shuttle contracts (e.g., Solid Rocket Booster Production Contract and the Shuttle Processing Contract (SPC)) provide incentives both for reliability, integrity, and safety of products and services on the one hand, and for cost and schedule on the other. Do these contracts provide an appropriate balance between the two types of incentives? That is, does NASA utilize contracts to reward and promote operational safety?

Findings

1. The SPC provides far greater incentives to the contractor for minimizing costs and meeting schedules than for features related to safety and performance. SPC is a cost-plus, incentive/award fee contract. The amount of the incentive fee is based on contract costs (lower costs yields a larger incentive fee) and on safe and successful launch and recovery of the Orbiter. The award fee is designed to permit NASA to focus on those areas of concern which are not sensitive to the incentive fee provisions, including the safety record of the contractor. However, the incentive fee dwarfs the award fee—while the maximum value of the award fee is only one percent of the value of the SPC, the incentive fee could total as much as 14 percent of the SPC.

2. During the developmental phases of the Thiokol contract for Solid Rocket Booster production (1980–1983), the contractor received consistent ratings of "Excellent-Plus" or "Superior" under the cost-plus, award-fee contract. NASA contracted with Thiokol on a cost-plus, incentive-fee (CPIF) basis beginning in July 1983. The CPIF contract pays strictly on the basis of costs, although penalties may be invoked for delays in delivery or for Shuttle accidents due to SRB failure. At the time of the Challenger accident, Thiokol was eligible to receive a very large incentive fee, probably on the order of $75 million.

Recommendations

1. NASA should reexamine all Shuttle contracts and report to the Committee with its findings and recommendations on whether more incentives for safety and quality can be built into these contracts. This report should address, *inter alia,* the SRB Production Contract and the SPC.

2. NASA's new Office of SR&QA should be involved in the procurement and award fee processes, both to establish reasonable guidelines and rewards in new contracts and to judge performance of ongoing contracts.

IV. BACKGROUND

Initial Events Following the Accident

On January 28, Chairman Fuqua stated on the floor of the House that the Committee would conduct comprehensive hearings and prepare its report on the Challenger accident and its implications after the National Aeronautics and Space Administration had completed its immediate investigation. NASA's effort was to follow the same investigative approach it had taken after the Apollo 204[1] fire.

In preparation for this time, Mr. Fuqua working with Mr. Lujan, the Ranking Republican Member, appointed a steering group of Committee Members two days following the accident to guide the Committee's work. This group consisted of:

Don Fuqua	Manuel Lujan, Jr.
Harold Volkmer	Robert Walker
Bill Nelson	Ron Packard

However, this plan and timetable were changed when President Reagan, by Executive Order, established a Presidential Commission on the Space Shuttle Challenger Accident on February 3, 1986. The order directed the Rogers Commission to make its final report to the President and the Administrator of NASA within 120 days. The order directed the Commission to: "(1) Review the circumstances surrounding the accident to establish the probable cause or causes of the accident; and (2) Develop recommendations for corrective or other action based upon the Commission's findings and determinations."

With this important new development, the Committee Steering Group met and decided to modify its earlier approach of investigating NASA's inquiry to that of reviewing the Rogers Commission's investigation. It was determined that the Committee's formal work would begin as soon as practicable after the Rogers Commission issued its report.

Committee Preparation

On February 5, the Chairman, Mr. Fuqua, and Mr. Lujan, wrote a letter to Chairman William P. Rogers stating their support for the serious task which was ahead of the Presidential Commission. In that letter Messrs. Fuqua and Lujan also outlined the Committee's approach, saying:

> We would like to begin our oversight process by asking you to establish procedures for providing us with progress reports as appropriate so that we can be kept advised of

[1] On January 27, 1967, astronauts Virgil Grissom, Edward White II and Roger Chafee were killed when their Apollo spacecraft was destroyed by fire on the launch pad.

the activities of your Commission. At the conclusion of the Commission's work, we will undertake a thorough review of your report; we expect that this review will be similar to the review and hearings held after the Apollo 204 fire and the Apollo 13 incident.[2]

It is our understanding that the Commission is tasked with completing its report in 120 days. In light of this fact, we would like to request your appearance before the Science and Technology Committee during the first week in June, or within one week of your final report, should you complete it sooner.

The letter to Chairman Rogers also noted that a similar letter had been sent that same day to the NASA Acting Administrator, Dr. William R. Graham. It stated that the Committee also planned, after hearing from the Commission, to take testimony from NASA management on the accident, and "closely review NASA proposed management plans designed to implement the Commission's recommendations."

Chairman Fuqua and Chairman Rogers then worked out an informal arrangement for the Committee Steering Group so that when there was sufficient reason to meet, in the opinion of the two chairmen, Chairman Rogers would brief the Steering Group on the progress of the investigation.

By April 22, the Steering Group felt it had heard sufficient information to brief the Members of the full Committee on Science and Technology. This was done in a closed meeting that day.

On May 16, 1986, Chairman Fuqua sent a memorandum to all Members stating that he had asked Congressman Robert Roe, the Ranking Majority Member, to chair the Committee hearings on the Challenger accident, stating that "there is a distinct possibility that follow-through activities related to the hearings will carry over into the next Congress in which I shall not serve."

Committee Trip

When it appeared that the Rogers Commission would be able to meet its 120-day deadline, Mr. Roe arranged to take a group of Committee Members and key staff to the Kennedy Space Center on June 6, 1986. At the Center the Members heard detailed accident briefings, took a tour of the Vehicle Assembly Building where a set of Solid Rocket Motors and External Tank was examined, and viewed the recovered debris from the Challenger spacecraft.

The Hearings

The Rogers Commission report was released on June 9, 1986. Immediately thereafter, the full Committee began its inquiry under the direction of Mr. Roe. The Committee heard from 60 witnesses during 10 days of hearings, for a total of 41 hours. A compilation follows:

---

[2] On April 13, 1971, Apollo 13's Command and Service Modules were disabled by an oxygen tank explosion en route to the Moon. The crew was recovered safely.

CHALLENGER INVESTIGATION HEARINGS

[Witnesses and total days of hearings]

Days of hearings	Total witnesses	Witnesses
June 10, 1986	3	Hon. William P. Rogers, Chairman, Presidential Commission on the Space Shuttle Challenger Accident.
		Neil Armstrong, Vice Chairman.
		Dr. Alton Keel, Executive Director.
June 11, 1986	7	Dr. James C. Fletcher, Administrator, NASA.
		RADM Richard Truly, Associate Administrator for Space Flight, NASA, accompanied by Mr. Edward Aldridge, Secretary of the Air Force, Washington, DC.
		Arnold D. Aldrich, Manager, National Space Transportation System, NASA.
		Dr. William Graham, Deputy Administrator, NASA.
		Dr. Dan Germany, Leader, Photo and TV Analysis Team, NASA.
		Capt. Robert L. Crippen, Astronaut, NASA.
June 12, 1986	12	Dr. James C. Fletcher, Administrator, NASA.
		Arnold D. Aldrich, Manager, National Space Transportation System, NASA.
		Mr. L. Michael Weeks, Deputy Associate Administrator (Technical), Office of Space Flight, NASA.
		Dr. Milton A. Silvera, Chief Engineer, NASA.
		RADM Richard Truly, Associate Administrator for Space Flight, NASA.
		J.R. Thompson, Vice Chairman, NASA, STS 51–L Design and Data Analysis Task Force, NASA.
		Dr. William Graham, Deputy Administrator, NASA
		REPRESENTATIVES FROM NASA'S TASK TEAMS
		Dan Germany, Deputy Manager, Space Station Project Office, NASA, Johnson Space Center.
		John Thomas, Manager Ad Hoc, Solid Rocket Motor Design Team, Marshall Space Flight Center.
		Thomas J. "Jack" Lee, Deputy Director, Marshall Space Flight Center.
		Thomas Holloway, Chief, Flight Director Office, NASA, Johnson Space Flight Center.
		Thomas Utsman, Deputy Director, NASA, Kennedy Space Center.
June 17, 1986	10	Charles S. Locke, Chairman of the Board and Chief Executive Officer, Morton Thiokole.
		Joseph C. Kilminster, Vice President, Space Operations, Morton Thiokol.
		Allan J. McDonald, Director, SRM Verification Task Force, Morton Thiokol.
		Roger M. Boisjoly, Staff Engineer, Morton Thiokol.
		U. Edwin Garrison, President, Aerospace Group, Morton Thiokol.
		Carver G. Kennedy, Vice President, Space Booster Programs, Morton Thiokol.
		Arnold R. Thompson, Supervisor, Structures Design, Morton Thiokol.
		Dr. William Graham, Deputy Administrator, NASA, accompanied by Thomas Moser, Deputy Administrator, Office of Space Flight, NASA.
		RADM Richard Truly, Associate Administrator for Space Flight, NASA.
		Dr. William Lucas, Director, Marshall Space Flight Center, accompanied by J. Wayne Littles, Deputy Director of Science and Engineering, Marshall Space Flight Center.
		Lawrence Mulloy, Assistant to the Director for Science and Engineering, Marshall Space Flight Center.
		Gerald W. Smith, Manager, Solid Rocket Booster Project, Marshall Space Flight Center.
		Stanley Reinartz (no comment), Manager, Special Projects Office, Marshall Space Flight Center.
		William Sneed, Assistant Director, Policy and Review, Marshall Space Flight Center.
June 18, 1986 [1]	5	U. Edwin Garrison, President, Aerospace Group, Morton Thiokol.
		Joseph C. Kilminster, Vice President, Morton Thiokol.
		Carver G. Kennedy, Vice President, Space Booster Program, Morton Thiokol.
		Allan J. McDonald, Director, SRM Verification Task Force, Morton Thiokol.
		Roger M. Boisjoly Staff Engineer, Morton Thiokol.
		Arnold R. Thompson, Supervisor, Structures Design, Morton Thiokol.
June 25, 1986	6	Maj. Donald "Deke" K. Slayton, Brig. Gen. James A. McDivitt, Capt. John W. Young, Col. Henry W. Hartsfield, Jr., Comdr. Robert L. Gibson, Gen. Thomas Stafford.
July 15, 1986	3	George Jeffs, President, North American Space Operations, Rockwell International.
		Richard M. Davis, President, Martin Marietta Michoud Aerospace.
		George Murphy, Executive Vice President and General Manager, United Technology Booster Production Co.
July 16, 1986	7	E.D. Sargent, President, Lockheed Space Operations Co. and Program Manager, Shuttle Processing Contract, accompanied by Fred Haise, President, Grumman Technical Services Division.
		Carver Kennedy, Vice President, Space Booster Programs, Morton Thiokol.
		David Owen, Lockheed Space Operations Co. and Deputy Program Manager, Kennedy Space Center.

CHALLENGER INVESTIGATION HEARINGS—Continued

[Witnesses and total days of hearings]

Days of hearings	Total witnesses	Witnesses
		Charlie Floyd, Systems Engineer, Lockheed Space Operations Co.
		James R. Dubay, President and General Manager, EG&G Florida, Inc., accompanied by Dr. Donald Kerr, Senior Vice President, EG&G Florida, Inc.
		George R. Faenza, Vice President and General Manager, McDonnell Douglas Astronautics Co.
July 23, 1986	2	Alan M. Lovelace, Vice President and General Manager, Space Systems Division, General Dynamics Corp.
		John F. Yardley, President, McDonnell Douglas Astronautics Co.
July 24, 1986	5	Lt. Gen. James A. Abrahamson, Director, Strategic Defense Initiative, Department of Defense.
		Jesse W. Moore, Director, Johnson Space Center, NASA/Houston, Texas.
		Robert F. Thompson, Vice President, Space Stations, McDonnell Douglas Astronautics Co.
		G.S. Lunney, President, Satellite Systems Division, Rockwell International.
		Arnold Aldrich, Manager, National Space Transportation System, Lyndon B. Johnson Space Center, NASA/Houston, Texas.
Total	60	

[1] Continued from June 17, Morton Thiokol only.

V. THE ACCIDENT

A. INTRODUCTION

This section as well as Sections VII and VIII identify what happened, as well as what did not happen, to cause the loss of the Challenger. This section also discusses why the accident happened in an effort to prevent future catastrophes.

By the time the Rogers Commission had completed its report, it had been learned that many items investigated by the Commission did not contribute to the accident. Consequently, this section is directed toward a more narrow range of possible contributing causes.

There were human as well as technical failings that combined on the morning of January 28, 1986, to cause the Challenger accident. Most of NASA's personnel were not involved in the Solid Rocket Motor program while there were others outside of NASA, such as the media, the Congress and the Administration, who were involved through their influence on the Shuttle program.

It should also be recognized that this report has the advantage of hindsight. Our investigation indicates that the decision to launch Challenger on January 28 suffered equally from a lack of information, misinterpretation of the information that was available, and a complex interplay of personalities among the principals involved. We are equally convinced, however, that the resulting decision to launch was arrived at as a logical conclusion of faulty premises, coupled with a failure to recognize the effect of temperature on the design.

We hope the lessons learned from this accident will lead to design improvements in the Shuttle Program. Just a few years ago, the collapse of the Hartford Civic Center contributed to the improvement of engineering design techniques to accommodate the unique secondary forces inherent in long-span structures. The Gothic cathedrals of the fourteenth century were constantly improved after their early failures were studied.

We hope this section, as well as Sections VII and VIII, properly identify the mistakes that led to the Challenger accident. It is the intent of the Committee to identify these mistakes so that NASA will regain its former level of excellence. The Committee has confidence that the men and women of the National Aeronautics and Space Administration will meet the challenge, improve the Shuttle and their management methods, and go on to explore new frontiers in space. This assumes, however, that the agency will now receive resources adequate to support the programs it is authorized to carry out by the Congress and the President.

For the benefit of those who may not be familiar with the Space Transportation System, the Shuttle consists of an Orbiter (51-L's Orbiter, the Challenger, was one of a four-vehicle fleet), an External Tank (ET), and two Solid Rocket Boosters (SRBs). (See Figure V-1.) A brief description of the Solid Rocket Booster and the Solid Rocket Motors is included to familiarize readers with these systems.

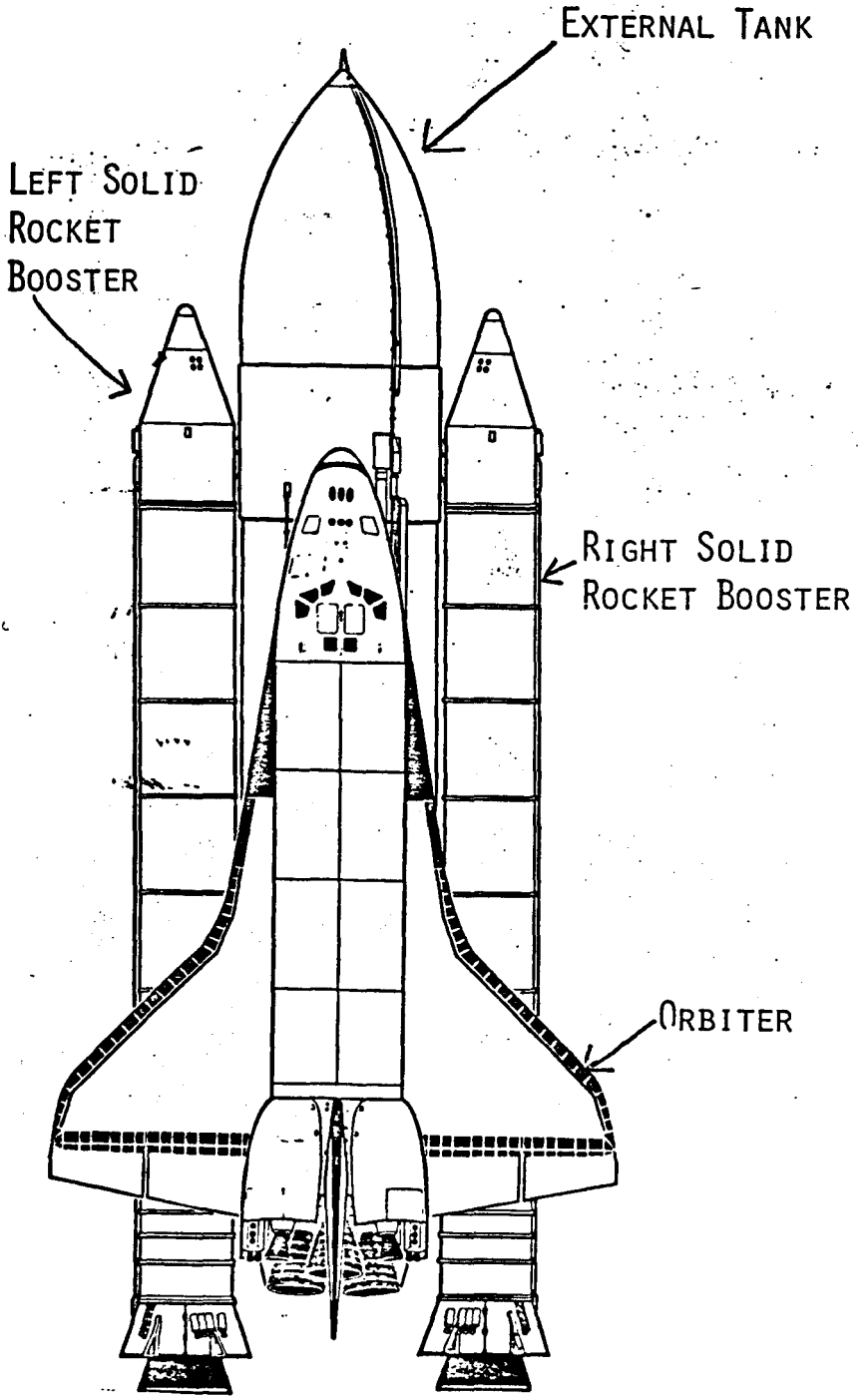
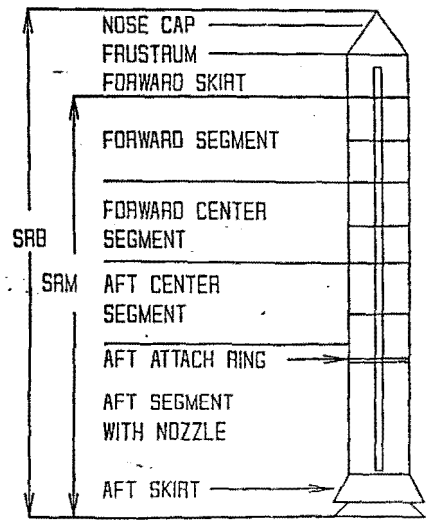

EXTERNAL TANK
LEFT SOLID ROCKET BOOSTER
RIGHT SOLID ROCKET BOOSTER
ORBITER

FIGURE V-1

The Solid Rocket Boosters operate in parallel with the main engines for the first two minutes of flight to provide the additional thrust needed if the Orbiter is to escape the gravitational pull of the Earth. At an altitude of approximately 144,000 feet (24 nautical miles), the SRBs separate from the Orbiter/External Tank, descend on parachutes, and land in the Atlantic Ocean. They are recovered by ships, returned to land, and refurbished for reuse.

The heart of the booster is the Solid Rocket Motor (Figure V-2). It is the largest solid propellant motor ever developed for space flight and the first built to be used on a manned craft. Larger solid motors have been test-fired but have never been carried through complete development to actual use in flight. The huge Solid Rocket Motor is composed of a segmented motor case loaded with solid propellant, an ignition system, a movable nozzle, and the necessary instrumentation and integration hardware.



STATISTICS FOR EACH BOOSTER

THRUST AT LIFT-OFF (2,650,000 pounds)

PROPELLANT
Atomized aluminum powder (fuel), 16 percent

Ammonium perchlorate (oxidizer), 69.83 percent

Iron oxide powder (catalyst), 0.17 percent (varies)

Polybutadiene acrylic acid acrylonitrile (binder), 12 percent
Epoxy curing agent, 2 percent

WEIGHT
Empty: (193,000 pounds)
Propellant: (1,107,000 pounds)
Gross: (1,300,000 pounds)

FIGURE V-2

Each motor case is made of 11 individual weld-free steel segments (Figure V-3). Averaging approximately 1.27 centimeters (0.5 inch) thick, the steel is a high-strength formulation. Each segment is heat-treated, hardened, and machined to the exact dimensions required. The 11 segments are held together by 177 high-strength steel pins at each case segment joint. The clevis-type joints are wrapped with reinforced fiberglass tape and sealed with a rubber seal band that is bonded to the case with adhesives.

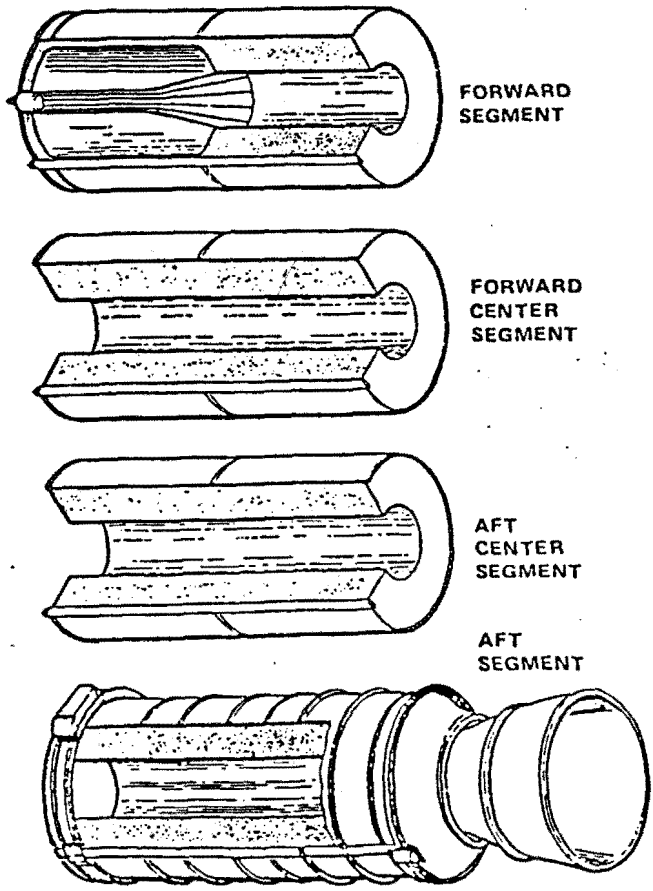


FIGURE V-3

In this report there are many references to the joint design, erosion and O-ring seals. There are several different joint designs used in the Solid Rocket Motor. The joint that failed on the last Challenger flight, the aft field joint, was not the one that had been giving NASA the most trouble. More O-ring erosion had been experienced on nozzle joints, the design of which is significantly different than the aft field joint. However, since NASA treated erosion as a problem that impacted both the nozzle and field joints, the data on erosion in this section includes that obtained from the nozzle joint.

Whenever a temperature is specified, it is essential that it be related to a specific medium such as air (or ambient temperature), rocket propellant, or casing joints, for example. The temperature of the joints, air and propellant can all be different at the same time,

just as the ocean temperature at the beach on a 90-degree day could be 75 degrees.

Much of this discussion concerns heat, or the absence thereof. For example, if an O-ring had given up heat during the night, it would very likely be at a lower temperature than the temperature of the air in the morning after the sun had risen. This was the situation at the time Flight 51-L was launched. The heat gained by the joint in the time after sunrise was not sufficient to raise the temperature of the O-ring material to a level where Thiokol engineers believed the O-ring could respond and seal the joint under ignition pressures.

The following chart describes the principal steps in the evolution, flight, and reconditioning of the Solid Rocket Motors (Figure V-4).

SOLID ROCKET MOTOR

PRINCIPAL STEPS IN THE EVOLUTION, FLIGHT AND RECONDITIONING OF SOLID ROCKET MOTORS

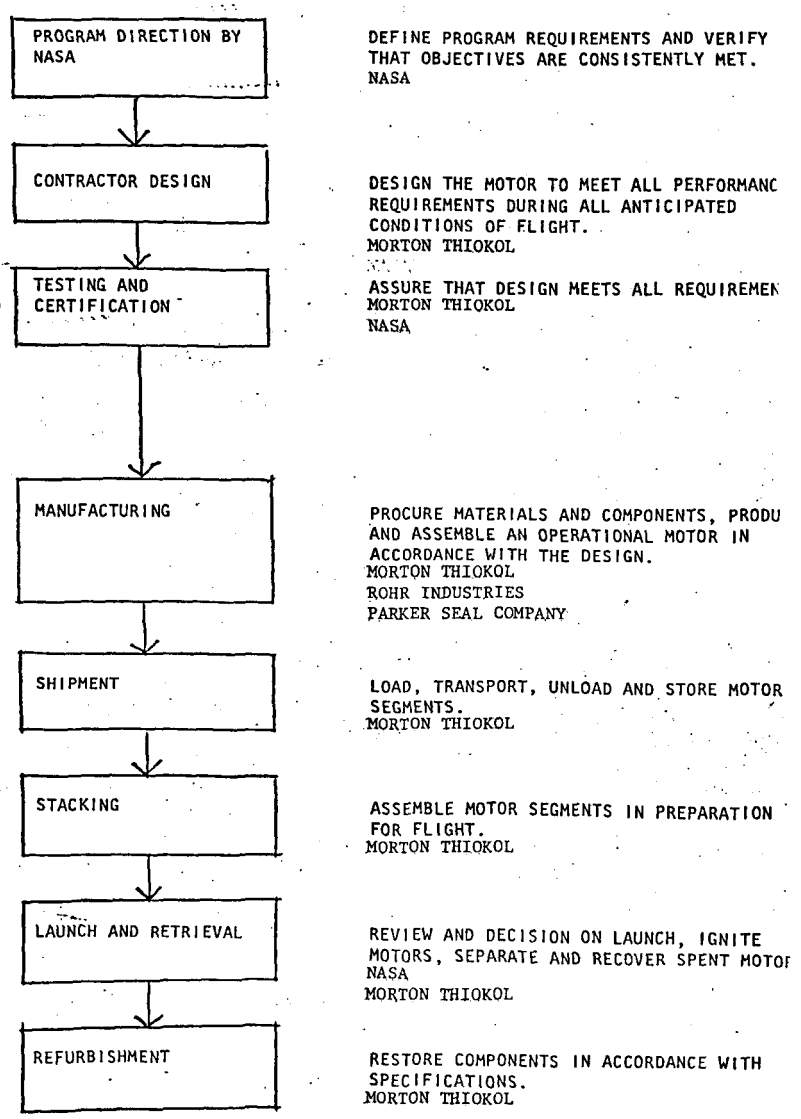


FIGURE V-4

FIGURE V-4

Because of the difficulty the reader may find in understanding the NASA Flight Readiness Review for the Solid Rocket Booster for Flight 51-L and the terms used to describe the steps in the process, the following chart describes the level of review, office conducting the review, and the scope of the review. In addition to the following meeting chart, there were numerous other ad hoc meetings on the SRMs including the meeting between NASA and Thiokol personnel during the evening before the launch of Flight 51-L.

TABLE I.—FLIGHT READINESS REVIEWS

[STS-51L]

Level and date	Reviewing office	Scope of review
IV—Dec. 11, 1985	Thiokol Wasatch	Conducted by Thiokol Solid Rocket Motor program managers in preparation for presentations to Marshall Space Flight Center (MSFC).
III—Dec. 17, 1985	SRM Office	Conducted by Larry Wear, Manager of the Solid Rocket Motor Program Office, MSFC. Material presented by Thiokol personnel.
III—Jan. 3, 1986	SRB Project Office	Conducted by Larry Mulloy, Manager of the Solid Rocket Booster Project Office. This is a combined briefing on the SRM and the elements making up the booster assembly, which, when integrated make up the Shuttle Solid Rocket Boosters.
III—Jan. 9, 1986	Shuttle Projects	Conducted by Stanley Reinartz, Manager, Space Shuttle Projects Office, MSFC. This review discusses all elements of the Shuttle managed by Marshall.
III—Jan. 13, 1986	Center Board	Conducted by Dr. William Lucas, MSFC Director. Final discussion of Marshall hardware in preparation for review by the Space Transportation System Program Manager.
II—Jan. 14, 1986	STS Program	Conducted by Arnold Aldrich, Space Transportation System Program Manager. First review dealing with the flight vehicle and associated ground support in its entirety.
I—Jan 15, 1986	Space Flight	Conducted by Jesse Moore, Associated Administrator for Space Flight. Remaining items that impact launch are discussed and assigned for disposition. Certificate of Flight Readiness is signed.
I—Jan. 25, 1986	L-1 Review	Meeting of the Mission Management Team to receive reports on action items remaining from the Flight Readiness Review. All action items should be closed by this time.

Considerable reference will be made to the "joint design" throughout this section of the report. Consequently, the following description of the joint is provided. (See Figures V-5 thru V-7.)

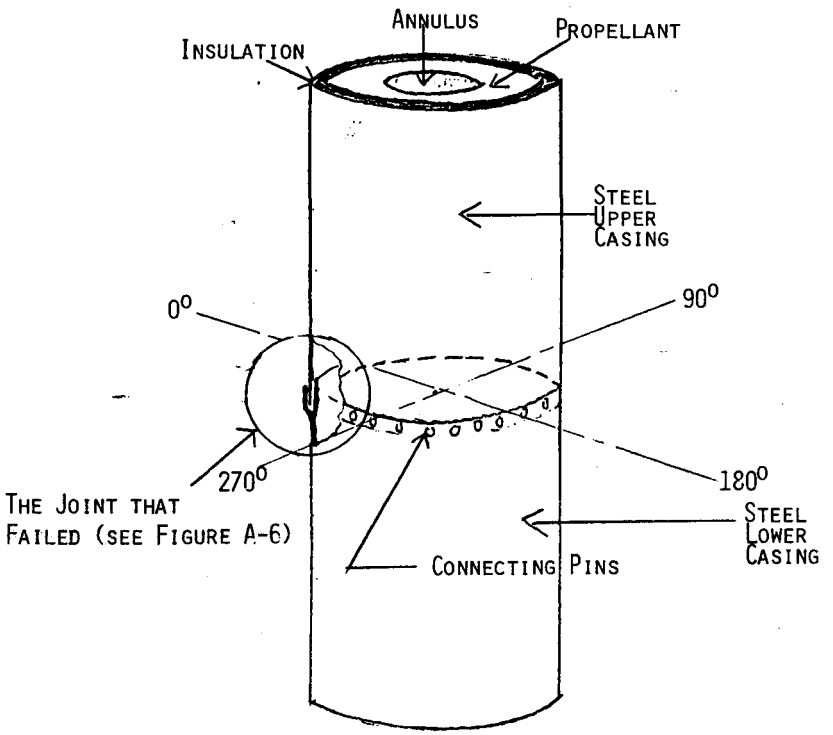
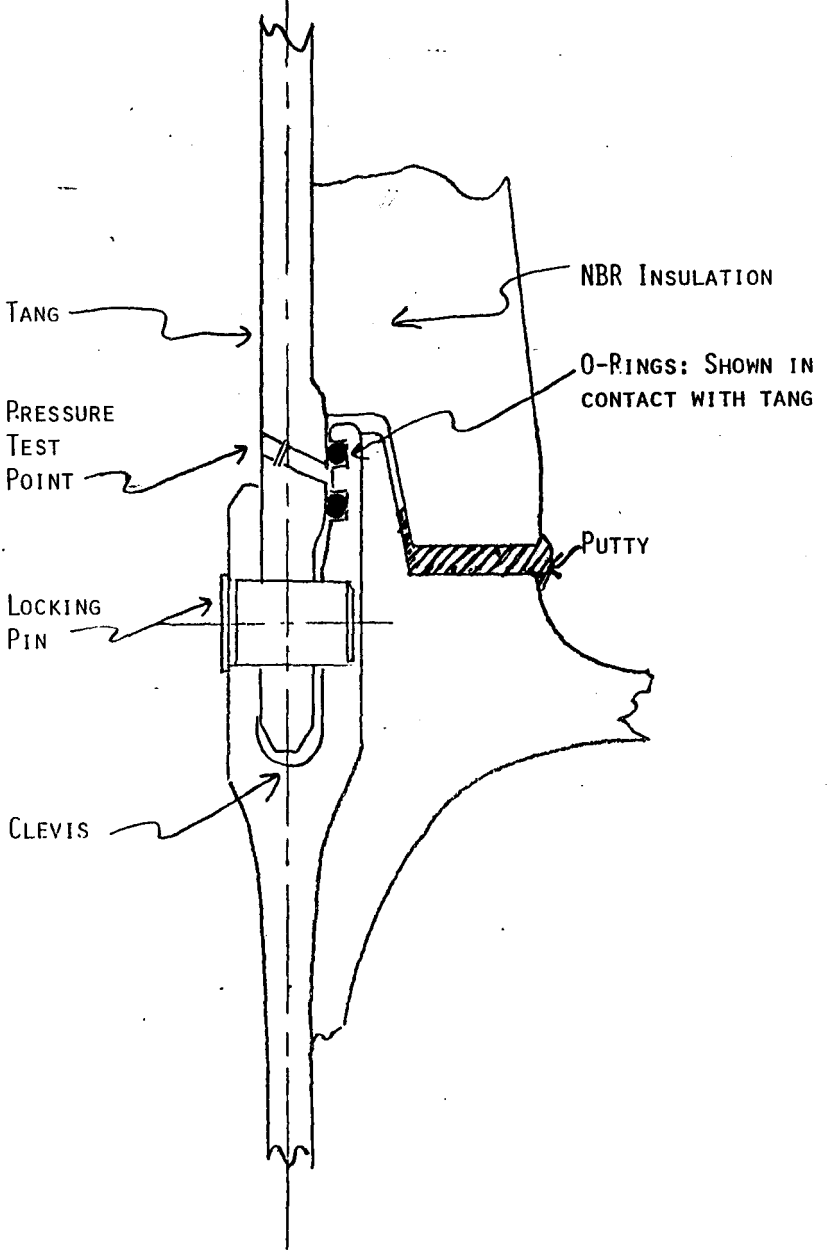

ANNULUS
PROPELLANT
INSULATION
STEEL
UPPER
CASING
0°
90°
180°
270°
THE JOINT THAT
FAILED (SEE FIGURE A-6)
CONNECTING PINS
STEEL
LOWER
CASING

FIGURE V-5

A. JOINT IN NORMAL ALIGNMENT
(NO GAPS BETWEEN O-RINGS AND TANG)



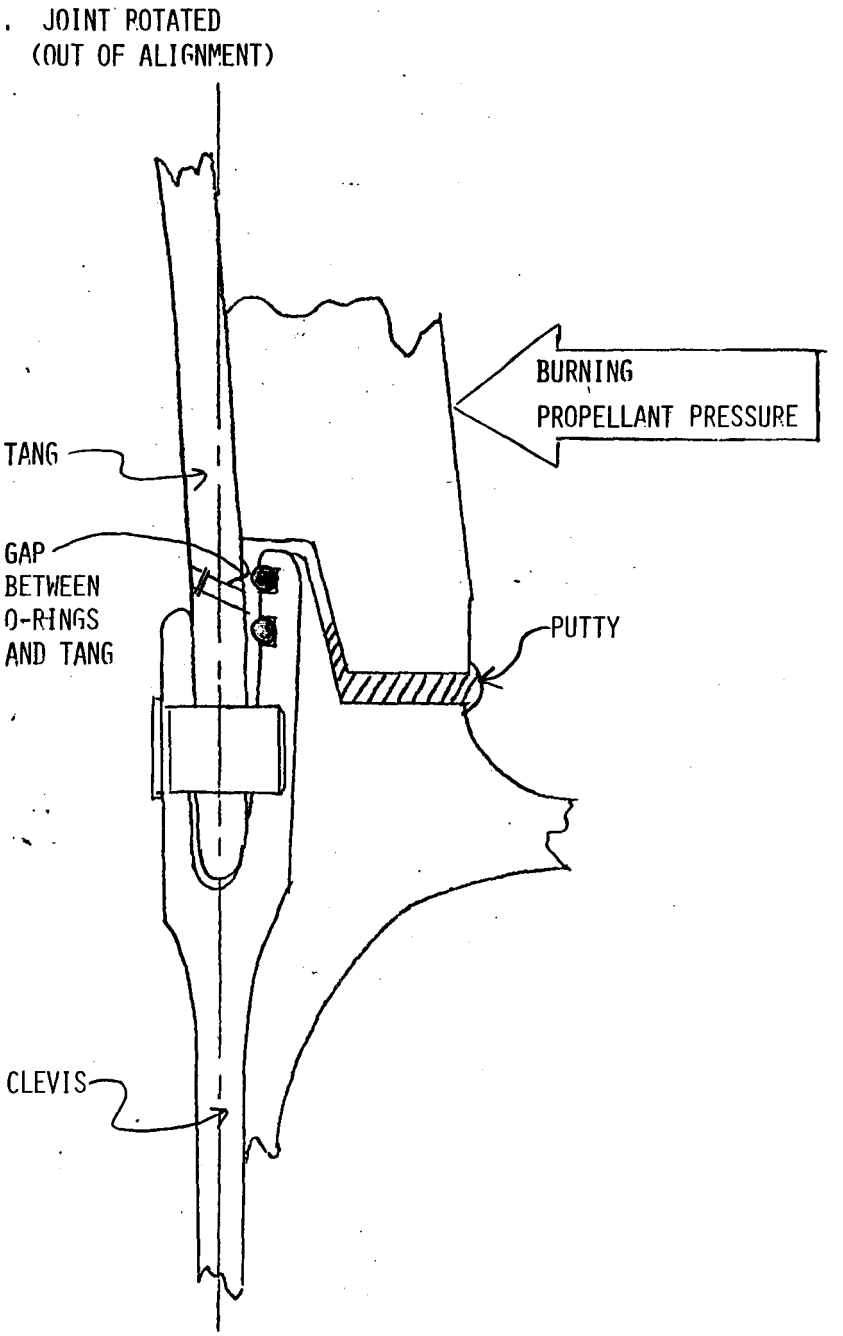

B. JOINT ROTATED
(OUT OF ALIGNMENT)
BURNING
PROPELLANT PRESSURE
TANG
GAP
BETWEEN
O-RINGS
AND TANG
PUTTY
CLEVIS

FIGURE V-7

B. Solid Rocket Motors

1. HISTORY

Issue

Was there sufficient time to correct the problems with the Solid Rocket Motor?

Findings

1. Problems with the joints which connect the Solid Rocket Motor casings were recognized for many years. While attempts were made to correct these problems, the measures taken were insufficient to provide a reliable joint.

2. The joint seal problem was recognized by engineers in both NASA and Morton Thiokol in sufficient time to have been corrected by redesigning and manufacturing new joints before the accident on January 28, 1986. Meeting flight schedules and cutting cost were given a higher priority than flight safety.

Discussion

At seven different times in the Shuttle Program, NASA and Thiokol managers made poor technical decisions that ultimately permitted continued flight of an unsafe Solid Rocket Motor design.

1. NASA's issuing of a performance specification that did not adequately take into account the known weather conditions that occur in Florida during the winter months.

2. Accepting the new joint design without sufficient certification and testing.

3. Failure to accept John Miller's [1] recommendations to redesign the clevis joint on all on-coming hardware at the earliest date.

4. Establishing a specific value for the upper limit of erosion that could be tolerated in flight on the basis of a "computer program model" instead of recognizing the erosion itself as a failure of the joint.

5. Proceeding through more than four years of Shuttle flights with continuing joint/seal problems without designing, testing and incorporating a new type of field joint and nozzle joint as well.

6. NASA's permitting Thiokol to continue making Solid Rocket Motors without conducting full scale tests as had been requested by NASA 14 months previously.

7. Mr. Mulloy's description of joint failures as being within "their experience base." In other words, if it broke before and the size of the recent break was no bigger than those before, then there was no problem. Even when the erosion surpassed all previous experience, NASA then went on and expanded its "experience base."

What follows is a list of events and documents which relate to the cause of the accident. They are included here to demonstrate that there was adequate experience and information available before the accident and that this information should have been sufficient to cause the initiation of corrective action before the launch of Flight 51-L.

---

[1] John Q. Miller is Chief, Solid Motor Branch, Marshall Space Flight Center, NASA.

July 16, 1973.—NASA issues a Request for Proposal (RFP) for the Space Shuttle Solid Rocket Motor project. Under "Scope of Proposal," the RFP stated in part:

> NASA considers that a prime contractor's use of established expertise in the private sector is an essential approach toward the objective of maximum economic effectiveness. Proposals from joint ventures will not be accepted, and the development of new expertise by a prime contractor, either in-house or elsewhere in the private sector, is to be avoided to the extent possible, since the latter course detracts from the stated objective.
>
> This RFP is specifically directed toward the design, development, test, production, acceptance, operation, and refurbishment of the Solid Rocket Motor and its ancillary equipment, post-flight analysis, and support functions. It is imperative in all considerations of the proposal and its subsequent implementation, that effort be made to minimize production and operating costs while maintaining reasonable DDT&E costs. The minimization of these costs entails the utilization of design and production approaches that will result in the lowest possible cost per flight consistent with the Space Shuttle Program early year funding constraints and the design, performance and reliability requirements.[1a]

August 27, 1973.—Thiokol, in the Executive Summary to its response to the RFP, addressed NASA concerns regarding SRM reliability (Appendix N, RFP 8-1-4-94-98401). Among other failure modes identified by NASA, Thiokol described the steps it had taken to prevent O-ring seal failure. These included:

DESIGN FEATURES

Redundant seals;
Protection of mating surfaces;
Assure proper environment and capability.

TEST AND CONTROL FEATURES

Functional leak check of dual seals prior to test or use;
Material migration/compatibility tests to demonstrate suitability.[1b]

November 19, 1973.—In its report to NASA Administrator James C. Fletcher, the Solid Rocket Motor Source Evaluation Board (SEB) evaluated the proposals generated by the Solid Rocket Motor RFP. Thiokol scored 124 out of a possible 200 points for its motor design, the lowest score among the four competitors. The only design strength identified by the Board: "Case joint leakcheck capability increases reliability and improves checkout operations."[1c]

[1a] NASA, Marshall Space Flight Center, "Request for Proposal: Space Shuttle Program Solid Rock Motor Project," RFP 8-1-4-94-98401, Volume 1, July 16, 1973, pp. 1-3, 1-4.

[1b] Thiokol, "Executive Summary: Proposal for Solid Rocket Motor Project for the Space Shuttle Program," Publications No. 0873-73270-1, Volume 1, August 27, 1973, p. 3-10 (Table 3-4).

[1c] NASA, Marshall Space Shuttle Center, "Presentation to NASA Administrator: Solid Rocket Motor Project Source Evaluation Board," RFP 8-1-4-94-98401, November 19, 1973, Chart D-4.

November 1973.—The SEB Design, Development and Verification Team Report rejected proposals by Aerojet and United Technologies Corporation to test motor performance at 40° and 90° Fahrenheit. The report stated that "The temperature conditioning of two motors to verify the motor performance over the range of 40° to 90° is not required, as this data can be obtained from the normal variation in ambient conditions."[1d]

December 12, 1973.—NASA Administrator James C. Fletcher announced selection of Morton Thiokol as contractor for Design, Development, Test and Evaluation (DDT&E) of the Solid Rocket Motors. In the source selection statement, "Selection of Contractor for Space Shuttle Program, Solid Rocket Motors," a statement was included that indicates that Thiokol ranked fourth out of the four bidders in the design category (See Appendix V–A). NASA, however, placed greater importance on cost reduction and Thiokol had an attractive cost proposal.[2]

January 9, 1978.—Major problems with the joint design were identified when Mr. John Miller of NASA sent a memo to Mr. Eudy. In it Miller stated, "Calculations performed by MSFC [Marshall Space Flight Center personnel] and agreed to by Thiokol show that distortion of the clevis joint tang for any joint can be sufficient to cause O-ring/tang separation. Data from DMT–1 [Development Motor Test-1] showed that this condition could be created by joint movement . . ." Miller continued, "All situations which could create tang distortion are not known, nor is the magnitude of movement known." Miller also noted that 15 percent industry recommended a compression value of 15 percent for adequate O-ring performance. He also cited a Thiokol test report dated August 15, 1977, TWR–11507, which showed a maximum compression of 5.8 to 7.0 percent for O-ring material and spliced joints. Finally, Miller also recommended a redesign of clevis joints on all on-coming hardware at the earliest possible effectivity to preclude unacceptable, high risk, O-ring compression values.

November 7, 1978.—Ten months later it would appear that there was nothing to worry about when a letter from E. G. Dorsey of Thiokol to Mr. George Hardy of MSFC contained the statement, "The extrusion data presented in the review and mentioned in the minutes have confirmed the capability of the O-rings to prevent leakage under the worst hardware conditions." Mr. Dorsey attached the Thiokol TWR–12019, dated October 6, 1978 to his letter.

February 2, 1979.—Mr. Eudy and Mr. Ray of NASA visited the Parker Seal Company. A trip report was sent to Messrs. Hardy/Rice/McCool of NASA which contained the following statement: "Parker experts would make no official statements concerning reliability and potential risk factors associated with the present design however, their first thought was that the O-ring was being asked to perform beyond its intended design and that a different type of seal should be considered. The need for additional testing of the present design was also discussed and it was agreed that tests which more

[1d] NASA, Marshall Space Flight Center, "Design, Development and Verification Team Report: Solid Rocket Motor Project," November 1973, p. 50.

[2] NASA, "Selection of Contractor for Space Shuttle Program, Solid Rocket Motors," January 2, 1974.

closely simulated actual conditions should be done." This report also referred to the O-ring extrusion gap being larger than Parker had previously experienced. (See Appendix V-I.)

November 12, 1981.—During STS-2, the second Shuttle flight, erosion of the primary O-ring was discovered in the 90 degree location of the aft field joint of the right hand Solid Rocket Motor. The 0.053 inch erosion was not discussed in the STS-3 Flight Readiness Reviews.[3] This was the deepest O-ring erosion that would be discovered in any case field joint.

February 25, 1983.—Employees of Thiokol discussed joint "gap size" and "O-ring compression" at a briefing at the Marshall Space Flight Center (MSFC).[4]

March 17, 1983.—Mr. Lawrence Mulloy, MSFC Solid Rocket Booster (SRB) Project Manager, informed NASA Level 1 (meaning the Associate Administrator for Space Flight), of the pending change in criticality from 1R to 1, which meant that a single seal failure could result in the loss of the Shuttle and crew. That change was approved on March 28, 1983.[5]

April 4, 1983.—STS-6 was the first flight to use the "lightweight case." It was also the first flight where a criticality factor of 1, instead of 1R, was assigned to the joint. After the flight, "blowholes" in the nozzle to case joints, not the case field joints, were found in both the left and right Solid Rocket Motors. These observations were not discussed in the Flight Readiness Reviews for STS-7.[6]

December 6, 1983.—An internal Marshall Space Flight Center (MSFC) memo from Mr. Miller to Mr. Horton highlighted the seal leak detection and zinc chromate putty problems. (See Appendix V-D.)

February 22, 1984.—Marshall Space Flight Center memorandum from Ben Powers to Horton requested that post-flight and post-static firing inspection on specific joints be made. The memo expressed concern about adhesion life of the zinc chromate sealant after installation on the SRM. See Appendix V-F.

March 2, 1984.—Thiokol personnel described the erosion discovered in the 351 degree location of the left Solid Rocket Motor forward field joint of STS-41B at a Flight Readiness Review. The erosion extended over three inches with a maximum depth of 0.040 inches. This was the first time the subject of O-ring erosion sustained on flights STS-2 and STS-6 was discussed as a technical issue at a Flight Readiness Review.[7]

March 8, 1984.—The notion of ACCEPTABLE EROSION was mentioned at a meeting of the Shuttle Projects Office Board for STS-41-C. Even though the joint was now classified as Criticality 1, which meant that failure of the joint could lead to the loss of the Shuttle and crew, the concept of "maximum possible" erosion, 0.090 inches, was accepted as an absolute value based on a comput-

---

[3] Thiokol Report, TWR-300209, Brian Russell, "Narrative History of SRM Seal Erosion and Blowby Incidents," June 25, 1986, p. 3.

[4] Rogers Commission Report, Volume II, p. H-1.

[5] Ibid.

[6] Ibid. [NOTE.—The nozzle to case joint design is significantly different than the case field joint design which caused the Challenger accident. However, it is cited here because some of the problems are relevant to the failure of the aft field joint.]

[7] Ibid.

er program which was supported by limited data. Furthermore, the 0.090 inch value was based on the concept that the O-ring would seal at 3 times the actual motor pressure even if the erosion extended to 0.095 inches thereby giving comfort in continuing with a known problem.[8]

March 1984.—Thiokol submitted their "Performance Charateristics of the SRM O-ring Assembly Test Plan," TWR-14336, which contained the following statement: "O-ring seals in rocket motors in general and the Space Shuttle SRMs in particular can suffer thermal degradation because of exposure to the high temperature motor chamber gases. Although none of the SRM primary O-rings to date have failed to perform their design function, there is some concern because of isolated events which show localized erosion as high as 0.053 inches. The postulated scenario for this thermal degradation effect is a short time duration impingement of a high energy jet which is induced during ignition pressurization by a combination of voids in the protective vacuum putty and the filling of available free volumes created by the tolerances of mating parts and the O-ring slots."

March 20, 1984.—Acceptable erosion was again discussed at the Flight Readiness Review briefing to the Marshall Center Board.[9]

March 27, 1984.—Mr. Mulloy discussed O-ring erosion at the Level 1 Flight Readiness Review for STS41-C. As a result, he received an "action item" to review the case and nozzle seals. The action did not have to be completed before the flight of STS41-C.[10]

Mr. Lawrence Wear, the Solid Rocket Motor (SRM) Element Manager, then directed Thiokol to establish a plan and a test program to investigate the issue. Thiokol was directed to determine if the O-ring erosion was acceptable and if so, why?[11]

April 6, 1984.—Heat degradation of the O-ring in the left SRM aft field joint of STS41-C was found, along with "blowholes" in the putty.

April 12, 1984.—In an internal Marshall Space Flight Center memorandum, John Q. Miller told Mr. Horton that "stacking difficulties and observed O-ring anomalies" were increasing with the use of Randolph putty. The former supplier, Fuller O'Brien, had discontinued producing the putty previously used in the Shuttle program. Accordingly, putty was ordered from Randolph Products. The memo requested expedited development of a putty with the characteristics of the Fuller O'Brien putty used prior to STS-8.

May 4, 1984.—Morton Thiokol prepared a Program Plan for the protection of Space Shuttle SRM primary motor seals. Thiokol's objective was to isolate the joint problem and to eliminate damage to the motor seals, the O-rings. The plan called for analysis and testing of O-rings, putty and associated lubricants.[12] See Appendix V-B.

---

[8] Ibid.
[9] Ibid.
[10] Ibid.
[11] Ibid.
[12] Morton Thiokol, "Program Plan, Protection of Space Shuttle SRM Primary Seals," TWR-14359, May 4, 1984.

May 23, 1984.—NASA responds to Thiokol's plan, endorsing the Program Plan, supplementing it and expressing continued concerns about zinc chromate putty performance.[13] See Appendix V-C.

May 30, 1984.—A presentation by Thiokol personnel at the SRM Preboard Flight Readiness for STS41-D described the problems with STS41-C.[14]

June 8, 1984.—The Marshall Center Board review for STS41-D took place without mention of the SRM problems found on STS41-C, even though Thiokol had prepared briefing charts for the review.[15]

June 18, 1984.—MSFC memorandum from Miller to Horton mentioned zinc chromate putty installation discrepancies and recalled eroded/heat exposure O-ring experiences on QM-4, STS-2, STS-6, STS-11 (41B) and STS-13 (41C).

June 18, 1984.—The Level 1 Flight Readiness Review for STS41-D took place, but again without mention of the O-ring problems discovered on STS41-C.

June 29, 1984.—Scenario of hot gas jet impingement against O-ring is substantiated in a teleconference between Thiokol and MSFC.

August 30, 1984.—STS41-D was launched. Upon disassembly of the SRM casings at the Kennedy Space Center (KSC). O-ring erosion was found in both the right-hand forward field joint and the left-hand nozzle joint. The field joint erosion was 0.028 inches deep and extended over a 3 inch span in the 275 degree location.[16]

September 12, 1984.—Thiokol personnel discussed the problems with STS41-D at the STS41-G SRM Preboard Review.[17]

September 19, 1984.—For the first time, at the STS41-G Shuttle Projects Board review, Mr. Mulloy mentioned the term "allowable erosion." He used the same briefing charts on September 20 at the Marshall Center Board Review.[18]

Flights STS41-G and STS51-A successfully flew without O-ring damage, a fact that was mentioned in the SRM Preboards for STS51-A and STS51-C.[19]

January 24, 1985.—With a calculated O-ring temperature of 53 degrees F, STS51-C suffered erosion and blow-by in the two case field joints. The primary O-ring in the left-hand forward field joint was eroded 0.010 inches over a span of 4.25 inches at the 163 degree location, with a considerable amount of soot between the primary and secondary O-rings. The primary O-ring in the right-hand center field joint was eroded 0.038 inches over a 12.5 inch space at the 354 degree location. There was soot behind the primary O-ring over a 110 degree arc and the secondary O-ring was heat damaged over a span of 29.5 inches.

January 31, 1985.—At the STS51-E Preboard review, Thiokol personnel described the previous O-ring damage in detail as well as

---

[13] NASA, letter from John Miller, MSFC, "Evaluation of TWR-14359, 'Program Plan, Protection of Space, Shuttle SRM Primary Motor Seals,'" EP 25 (84-49), May 23, 1984.
[14] Rogers Commission Report, Volume II, p. H-1.
[15] Ibid.
[16] Ibid.
[17] Ibid., p. H-2.
[18] Ibid.
[19] Ibid.

joint performance. They also showed analytical predictions of the "maximum expected erosion." [20]

February 12, 1985.—Mr. Mulloy and Thiokol personnel presented a summary of STS 51-C O-ring related problems during a briefing to the Shuttle Projects Office Board. A portion of the problem summary on the briefing charts referred to a field joint O-ring blow-by problem as being an "acceptable risk." In this briefing the secondary O-ring was referred to as a "redundant seal using actual hardware dimensions" even though the field joint had been officially classified as Criticality 1 for two years.[21]

February 14, 1985.—Mr. Mulloy addressed the Marshall Center Board but did not comment on the STS 51-C O-ring problems in detail.[22]

March 7, 1985.—MSFC Memo to Mr. Mulloy from Mr. McCool. McCool was concerned that 14 months had elapsed since full scale diameter tests to provide data on zinc chromate putty behavior as it related to its effect on joint leak checks were requested. McCool pointed out that the only positive response from Thiokol was the Program Plan submitted on May 4, 1983.[23] (See Appendix V-E.)

April 4, 1985.—A letter from MSFC to Mr. Joseph Kilminster of Morton Thiokol requested specific sub-scale and full-scale tests on effects of zinc chromate putties on O-ring sealing integrity.

April 12, 1985.—STS 51-D was launched and, upon disassembly, erosion of the primary O-rings in both nozzle joints was discovered. The right-hand nozzle primary O-ring eroded to a depth of 0.068 inches over a 6 inch span at the 116 degree location. The left-hand nozzle primary O-ring eroded to a depth of 0.011 inches over a 2.12 inch span at the 14 degree location. There was no blow-by past either nozzle O-ring.

April 17, 1985.—The Shuttle Projects Board for STS 51-B was held without mention of seal problems. There was also no mention of seal problems associated with STS 51-C or 51-D at the Level 1 Flight Readiness Review on April 23, 1985.[24]

April 22, 1985.—Thiokol's evaluation of a second source for putty is issued. The evaluation states: "The Randolph Products putty is the only material presently qualified for use on the Space Shuttle Program. It is the desire of Morton Thiokol to evaluate and qualify a second source for a joint filler material." The evaluation went on to state, "The material has demonstrated poor processing characteristics and is moisture sensitive."[25]

April 24, 1985.—Problem Assessment System Record Number A07934, tracking damage to the field joint seals, contains the following entry: "At NASA request, a solution for O-ring erosion will not involve a radical design change. Therefore, the possible solutions under current investigation are linked to: (1) new O-ring [ma-

---

[20] Ibid.
[21] Ibid.
[22] Ibid.
[23] NASA, MSFC, Memo from Mr. Alex McCool, "Request for Initiation of Testing to Provide Data for Resolving the Burned O-Ring Seal Problem on the Space Shuttle SRM," EPOI (85-48), March 7, 1985.
[24] Rogers Commission Report, Volume II, p. H-2.
[25] Thiokol, S.H. Cardall, "SRM Evaluation of Second Source Joint Filler Material," TWR-14946, April 22, 1985, pp. 1-2.

terials] and/or diameter and (2) new vacuum putty and/or layup procedure.

April 29, 1985.—Flight STS 51-B was launched and when the Solid Rocket Boosters were recovered, it was found that the worst O-ring erosion to date had occurred. The left-hand nozzle primary O-ring eroded to a depth of 0.171 inches over a 1.50 inch space at the 54 degree location. There was evidence of considerable blow-by. The secondary O-ring was eroded to a depth of 0.032 inches over a 3 inch span which was also at the 54 degree sector. The right-hand nozzle O-ring eroded to a depth of 0.005 inches over a 3.50 inch span at the 14 degree location.

May 8, 1985.—Blowholes through the putty in one field joint from each of the STS 51-B SRMs was mentioned before the SRB Board for Flights STS 51-F and STS-51-G.

May 29, 1985.—The STS 51-G Shuttle Project Board took place without mentioning O-ring problems.[26]

May 13, 1985.—The Center Board Review for STS 51-G took place without mentioning O-ring problems.[27]

June 11, 1985.—The Level 1 Flight Readiness Review for STS 51-G took place without mentioning O-ring problems.[28]

June 17, 1985.—STS 51-G was launched, experiencing blow-by and erosion in both nozzle joints. The right-hand nozzle primary O-ring was eroded in two different places. The left-hand nozzle primary O-ring was also eroded and there was blow-by associated with all three locations.

July 1, 1985.—A combined Flight Readiness Review for the Marshall SRM Preboard, SRB Board, Shuttle Project Office Board, and Marshall Center Board was held at which Thiokol personnel presented an extensive analyses of the problems discovered on Flight STS 51-B.[29]

July 2, 1985.—Mr. Mulloy briefed the Level I Flight Readiness Review for STS 51-F and presented the STS 51-B O-ring erosion problem as a "closed item." Mr. Mulloy based this resolution on the use of a higher 200 psi leak check stabilization pressure and introduced, for the first time, a rationale for accepting secondary O-ring erosion. The Roger Commission would not find any reference to O-ring problems in any Flight Readiness Review associated with Flight STS 51-D or STS 51-G.[30]

July 19, 1985.—An attempt to form an SRM Erosion team at Thiokol "virtually failed" according to Mr. Roger M. Boisjoly because of lack of commitment on the part of Thiokol personnel.

July 22, 1985.—One of the engineers who appreciated the joint problem was Mr. Boisjoly of Morton Thiokol. In a "Progress Report" he wrote, "This problems has escalated so badly in the eyes of everyone, especially our customer, NASA, that NASA has gone to our competitors on a proprietary basis and solicited their experiences on their joint configuration."[31] (See Appendix V-G.)

---

[26] Ibid.
[27] Ibid.
[28] Ibid.
[29] Ibid.
[30] Ibid., p. H-3.
[31] Thiokol, Roger Boisjoly, "Progress Report: Applied Mechanics Center," July 22, 1985.

64-420 O - 86 - 3

July 29, 1985.—STS 51-F was launched without O-ring erosion problems. However, there was a blowhole through the putty in the right-hand SRM nozzle and the primary O-ring was affected by heat.

July 31, 1985.—Boisjoly wrote an interoffice memo to R.K. Lund, Morton-Thiokol's Vice President of Engineering: On it, he warned that the rationale for flying the joint design was now suspect as a result of the secondary O-ring erosion on STS 51-B.[31a] See Appendix V-J.).

August 7, 1985.—The Shuttle project review for STS 51-I was conducted, followed on August 13 by the Marshall Center Board and on August 15 by the Level 1 Flight Readiness Review. The O-ring damage was noted at these reviews.

August 19, 1985.—Thiokol gave a presentation to Mr. Weeks, NASA Deputy Associate Administrator for Flight (Technical), and others at NASA Headquarters, which contained the following chart.

SUMMARY OF SIGNIFICANT OBSERVATIONS

All joints:
- Seal damage always has associated putty blowhole
- Putty blowholes exist without resultant seal damage
- Soot blowby can occur away from a putty blowhole
- Frequency of O-ring damage has increased since incorporation of:
  - Randolph putty;
  - Higher stabilization pressures in leak test procedure;
  - High performance motors.
- Randoph putty is more susceptible to environmental conditions such as humidity and temperature.
  - Can become leathery in dry conditions;
  - Becomes extremely sticky in moist conditions and in some cases begins to disintegrate.[32]

August 20, 1985. A Thiokol interoffice memo mentioned that a Nozzle O-ring Investigation Task Force had been formally instituted, stating, "As you are aware, we have experienced O-ring damage on a random basis in the case field joints and prevalently in the case/nozzle joint on the Space Shuttle Booster Motors. The frequency had increased in recent flights. While we have not compromised the performance of any motor to date, the result of a leak at any of the joints would be catastrophic."

August 27, 1985. Flight STS 51-I was launched, after which it was discovered that there was primary O-ring erosion in two locations on the left-hand SRM nozzle joint. At the reviews for STS 51-J, which occurred on September 9, 1985, September 17, 1985, September 19, 1985, and September 26, 1985, the O-ring erosion noted on STS 51-I was merely itemized as, "left-hand nozzle to case primary O-ring erosion within experience base." There was no O-ring damage on Flight STS 51-J.

August 30, 1985. One year and four months after the original drafting of Thiokol's Program Plan TWR-14359, for improvement of Space Shuttle SRM Motor Seals, the revised version of the plan was issued.

[31a] Thiokol, Roger Boisjoly, "SRM O-Ring Erosion/Potential Failure Criticality," Memo 2870: FY 86: 073, July 31, 1985.

[32] Rogers Commission Report, Volume II, p. H-73.

October 1, 1985. R.V. Ebeling of Morton Thiokol submitted a weekly activities report to A.J. McDonald, Director, Solid Rocket Motor Project, with copies to J. Kilminster and others, which included the following statements:

> *Executive Summary.* HELP! The seal task force is constantly being delayed by every possible means. People are quoting policy and systems without work-around. MSFC is correct in stating that we do not know how to run a development program.
>
> * * * * *
>
> 5. The allegiance to the O-ring investigation task force is very limited to a group of engineers numbering 8–10. Our assigned people in manufacturing and quality have the desire, but are encumbered with other significant work. Others in manufacturing, quality, procurement who are not involved directly, but whose help we need, are generating plenty of resistance. We are creating more instructional paper than engineering data. We wish we could get action by verbal request but such is not the case. This is a red flag.

(See appendix V–H.)

October 4, 1985. Roger Boisjoly's Activity Report identified problems in obtaining support from Mr. Kilminster for the O-Ring Investigation Task Force.

October 30, 1985. STS 61–A experienced erosion of the right-hand nozzle primary O-ring to a depth of 0.075 inches over a 13 inch space at the 97 degree location. There was also blow-by past the primary O-rings in the center and aft field joints on the left-hand SRM. But these problems were not discussed at the STS 61–B SRB Board Review on November 4, 1985. However, Mr. Mulloy included a note at the Shuttle Project Board Review on November 6, 1985, "SRM Joint O-ring performance within experience base."

November 18, 1985. Mr. Mulloy briefed the Level 1 Flight Readiness Review stating, "Post flight inspection of SRM revealed hot gas erosion of primary nozzle/case joint-O-ring on right-hand SRM—Within previously accepted experience."

November 20, 1985. Thiokol briefing document TWR–15349, "SRM O-ring Task Force Status and QM–5" Recommendations, presented new seal design concepts and recommendations for Qualification Motor-5 testing on February 20, 1986.

November 26, 1985. STS 61–B experienced primary O-ring erosion in both nozzle joints. There was also blow-by past the primary O-ring in the left-hand nozzle joint. These observations were noted at the STS 61–C SRB Board Flight Readiness Review on December 2, 1985.

December 4, 1985. At the STS 61–C Shuttle Project Board, Mr. Mulloy noted "SRM joint O-ring performance within experience base." The Commission's copy of the December 9, 1985, Marshall Center Board briefing was incomplete; however, at the December 11, 1985, Level I Flight Readiness Review, it was reported that there were "No 61–B flight anomalies."

December 11, 1985. Thiokol management holds a Solid Rocket Motor Flight Readiness Review for STS 51-L. No discussion of O-ring anomalies occurs.

December 17, 1985. Larry Wear holds a Flight 51-L Solid Rocket Motor Project Flight Readiness Review at Marshall Space Flight Center.

January 3, 1986. The Level III Flight Readiness Review for Flight 51-L takes place at Marshall. SRB recovery system changes are the primary point of discussion.

January 9, 1986. Larry Mulloy makes his Flight 51-L presentation at the MSFC Shuttle Projects Office Readiness Review. SRB parachutes are discussed. O-rings are not.

January 12, 1986. STS 61-C experienced nozzle Joint O-ring erosion and blow-by and a field joint O-ring was eroded 0.011 inches over an 8 inch span at the 162 degree location. There was blow-by past the primary O-ring in the left-hand nozzle joint between the 255 degree and 335 degree locations. The primary O-ring in the left SRM aft field joint was eroded 0.004 inches over a 3.5 inch span at the 154 degree location.

January 13, 1986. Marshall Space Flight Center 51-L Readiness Review. Mulloy again does not mention O-ring anomalies.

January 14, 1986. Mulloy's Flight 51-L presentation to the Level II Flight Readiness Review indicates there were "no 61-C flight anomalies."

January 15, 1986. During the STS 51-L Level I Flight Readiness Review, Mr. Mulloy noted that there were "No 61-C Flight Anomalies," and that there were "No major problems or issues."

January 25, 1986. According to Mr. McDonald, Mr. Mulloy mentioned that 61-C had suffered O-ring erosion "within experience base" at the STS 51-L L-1 Flight Readiness Review.

January 26, 1986. The *Orlando Sentinel* printed an article titled, "Bitter freeze is expected to clobber state Tuesday." [32a]

January 27, 1986. Thiokol and Marshall personnel spend approximately three hours in a teleconference debating the effect that predicted low temperatures will have on the performance of the O-ring seals.

January 28, 1986. The ice/frost evaluation team visits Launch Complex 39B at 1:45 a.m., 6:45 a.m. and 10:30 a.m. Meeting with Rockwell personnel concluded with a decision to continue the launch countdown.

January 28, 1986. STS 51-L was launched at approximately 11:38 a.m. Eastern Standard Time.

2. SUMMARY OF CASING JOINT DESIGN

Issue

Why did the aft field joint between the steel containers that hold the Solid Rocket Motor propellant fail to contain the burning gases of the propellant during lift-off and flight operations?

[32a] John Wark, "Bitter Freeze is Expected to Clobber State Tuesday," *The Orlando Sentinel,* Jan. 26, 1986, p. B-3.

Findings

1. The design of the field joint was unsatisfactory and could not reliably contain the burning propellant gases under the range of operating conditions to be expected during the lift-off and flight phases.

2. The O-ring materials and putty used in the design of the joint were unsatisfactory as used on the Shuttle, particularly during the winter months. Furthermore, neither NASA nor its contractor, Morton Thiokol, can adequately control the quality or consistency of these kinds of materials, which are made from recipes known only by the manufacturer and which can be changed without certification and approval.

Recommendations

1. NASA should write and issue a new and more accurate performance specification which would cover the full range of thermal and structural requirements for the Solid Rocket Motors, with an adequate factor of safety for unusually low temperatures.

2. The Committee concurs with the Rogers Commission Report Recommendations on new joint design, but believes it is more appropriate to be more explicit in identifying the weaknesses in the joint design that need correction.

3. The field joints of the Solid Rocket Motors should be redesigned to account for the following features while providing a significant factor of safety:

a. Movement in the joint;

b. Proper spacing between tang and clevis;

c. Seals made to withstand high and low temperatures under all dynamic thermal and structural loadings;

d. Adequate sealing without the use of putty;

e. Protection against insulation debonding and propellant cracking.

Discussion

This section is a summary of Section VII, Casing Joint Design. For details and substantiation of the statements made in this summary, refer to Section VII.

The evidence, consisting of recovered pieces of the right Solid Rocket Motor casings, photographs of smoke and flame emanating from the right Solid Rocket Motor and telemetry data transmitted from STS 51-L back to Mission Control at the Johnson Space Center verify the failure of the aft field joint of the motor.

As mentioned earlier, NASA's performance specifications did not anticipate operations at temperatures below 31 degrees, a temperature that might occur in Florida during the winter months. The design of the joint was unsatisfactory to provide for the low temperatures or water in the joints that existed on January 28. While it was based on an existing similar rocket casing joint design that had been successful, the design was changed to accommodate the manufacturing requirements of the larger sized shuttle rocket motors. There were even some features of the revised design that indicated the changes were an improvement. It was easier to assemble in the field and it had a second O-ring. The designers

thought if the first O-ring failed, the second would surely hold the propellant gases.

The casing joints, as described in the Introduction, have to withstand various structural loads, which change dramatically as the shuttle is assembled, through launch operations, separation of the Solid Rocket Booster and retrieval from the ocean. The joint is dynamic; the components move under these loads. The loads carried by the aft field joint are different from those carried by other joints. The design, based on these loads and 24 successful missions, appeared satisfactory.

One of the loads, however, that of the propellant gas pressure, was not adequately accommodated. The zinc chromate putty, intended to protect the O-rings from this high temperature and relatively high pressure gas, frequently failed and permitted the gas to erode the primary O-rings.

Instead of redesigning the joint, NASA and Thiokol persisted in trying to fix the problem by changing leak-test pressures, changing the size of the O-rings, and trying to control proper spacing between the tang and clevis where the O-rings were located.

Complicating this problem, two of the materials used in the joint, the putty and the fluorocarbon elastomer O-rings, were not suited to the task of containing the propellant gas under the full span of Shuttle operating conditions. The behavior of the fluorocarbon elastomer O-rings was something of a mystery to NASA and its contractor. The material was "proprietary," meaning that the constituents used were known only to the manufacturer. Fluorocarbons are expensive, so fillers are frequently added to reduce the cost of the material. These materials behave unlike most other materials. The particular material used in the manufacture of the shuttle O-rings was the wrong material to use at low temperatures.[33] Nitrile or silicon based materials would have demonstrated better performance characteristics.

It became necessary to find a new putty when the original supplier, Fuller O'Brien, stopped making it because it contained asbestos. The characteristics of the new putty changed substantially in response to the quantity of water in the air and it was difficult to apply in both the dry climate of Utah and the dampness of Florida. Its performance in use was highly unpredictable. Again, NASA and its contractor tried to make up for the unsatisfactory material by storing it under refrigeration prior to application in Florida.

After ignition of the solid propellant in the SRM, it was learned that the O-ring could be seated by the motor's gas pressure yet still suffer erosion as the hot gases came in contact with it. As mentioned, O-ring erosion was noted after various flights and tests. Also seen was damage given the name "blow-by", a condition where erosion was not necessarily present but where there was evidence that the propellant gas had bypassed the primary O-ring. But rather than identify this condition as a joint that didn't seal, that is, a joint that had already failed, NASA elected to regard a certain degree of erosion or blow-by as "acceptable." To make matters worse, confidence was mistakenly obtained from a mathemati-

[33] NASA's primary concern was having a very durable material with excellent high temperature performance characteristics.

cal model which suggested that if the erosion did not exceed a specific depth, the O-ring would still seal that joint. In cases where the erosion did exceed the maximum predicted by the model, NASA expanded its experience base to cover this increased damage.

As the joint seals continued to exhibit erosion or blow-by or both, more research illustrated the importance of maintaining proper gap spacing between the tang part of the joint and the O-ring face of the inner clevis leg. Too little space, and the O-rings would not seal. Too much space, and again the seals would fail. Since the joint opens, or "rotates," when the Solid Rocket Motor is ignited, maintaining proper spacing was difficult if not impossible. The maintenance of such close tolerances in spacing, on the order of 20 thousandths of an inch, while joining 300,000 lb. segments that have been bent during shipment, was not sufficiently provided for in the design. Months passed until, in 1985, engineers at NASA recognized that the design was unsatisfactory. In fact, NASA had written to several other contractors soliciting help with the joint problems. Unfortunately, in the quest to meet schedule and budget, the warnings of the engineers were not heeded.

Based on the above conditions and the evidence, the Committee has endeavored to determine the way in which the joint failed; recognizing that such a determination is difficult, if not impossible, to make with 100% certainty.

The following is the most probable sequence of the joint failure:

1. The failure occured in the lower assembly joint near a strut that connects the Solid Rocket Booster to the External Tank.
2. At that location, the spacing between the two casings was too small to facilitate a tight seal.
3. Also, at that location, there probably existed a hole through the insulating putty, which would act as a conduit concentrating the hot propellant gas on the primary O-ring.
4. The freezing temperatures reduced the capability of the O-rings to seal. Worse, at this particular location, near the connecting strut, the joint was made even colder by the further loss of heat caused by the direct connection to the liquid hydrogen fuel, at 423 degrees below zero, in the external tank.
5. When the Solid Rocket Motors were ignited, the pressure from the motor changed the spacing between the casings. Among other effects, this can prevent the secondary O-ring from sealing.
6. Seven inches of rain fell while the shuttle was being prepared for launch. Water very likely penetrated the joints and froze. Ice in the joints could have dislodged the secondary O-ring even if the change in spacing, coupled with a cold and stiff O-ring, did not.
7. Smoke at ignition occurred at a location near the connecting strut to the external tank. At that location, the primary O-ring was either unseated or eroded and the secondary O-ring was unseated.
8. The primary O-ring was sealed at other locations around the motor casings.
9. The breach in the primary O-ring clogged with burned char and aluminum oxide from the propellant in less than 3 seconds, causing the smoke to stop.
10. At 37 seconds, 45 seconds and 58 seconds into the flight, the Space Shuttle encountered heavy turbulence, which forced the

steering controls to cycle through changes more severe than previous flights.

11. After throttling back to 65% power as planned, at 57 seconds, power was increased to 104%.

12. The combined effect of the turbulence and the increase in power caused the material which clogged the joint to break free, reopening the joint.

13. A flame from the right Solid Rocket Motor was seen at the location near the connecting strut.

14. This flame burned through the external tank and caused the destruction of the shuttle.

Since the technical faults in the joint design must be corrected if safe shuttle flight is to resume, this subject has been discussed in more detail in Section VII.

3. TESTING AND CERTIFICATION

Discussion

In developing the Solid Rocket Motor, Thiokol concentrated most of their efforts and concerns on the proper design and performance of the propellant. There is no question that this is where the emphasis on safety and performance is required. The propellant is a high performance material, dangerous to manufacture and handle and which must be prepared to the highest quality standards. Consequently, testing and certification of the propellants, as well as its performance, was carefully controlled. This does not mean that the design of the casings was ignored. Considerable attention was paid to the design of the casings because they were larger than seen on any previous Solid Rocket Motor, because this Solid Rocket Motor would be used on a manned flight system, and because these particular motors would be brought back, refurbished and reused. Given this background, the testing of the joint was included in static firing tests. While there were no special tests conducted to confirm and certify the joint as a separate item, analysis was performed to assure that the joint was adequate. Later, during the operation of the Solid Rocket Motor, it was discovered that the performance of the joint was unsatisfactory.

4. MANUFACTURING

The Solid Rocket Motor is 126 feet long and 12 feet in diameter. The propellant weighs 1.9 million pounds and the average thrust is 2.3 million pounds. Fifty of these motors have been produced. The segmented Solid Rocket Motor case is roll formed from D6AC steel. The case is weld-free and consists of eleven segments. The propellant is made in batches at 135 degrees F and it takes 40 to 43 of these batches to load one casting segment. One segment includes two steel cases which are joined in the factory. The content and quality of the materials used to make the propellant is inspected prior to mixing. The motor is designed for a short burn time (122 seconds) and therefore has a high mass flow which requires a large burning surface. In manufacture, either new steel casings or previously used casings are employed. The first step is to apply the rubber insulation liner around the inside of the casings. The insu-

lation is removed from a roll and spread around the inside of the casings with special tooling. After application it is cured in place in an autoclave. After the casings have been insulated, they are placed in a casting pit. The propellant is then poured into the casings under vacuum. The propellant is then cured and the casings are removed from the pit. There is no indication that there were any manufacturing defects that contributed to the loss of the Challenger.

5. STACKING OPERATIONS

Issue

Was there any damage to the casing joints or contamination that occurred during the stacking operations when the Shuttle was assembled in the Vehicle Assembly Building (VAB) that could have contributed to the failure?

Finding

There was no evidence of joint contamination, fracture, or other damage from foreign objects or due to casing ovality that contributed to the joint failure. Although certain problems occurred during stacking and the procedures were violated once, there was no evidence that these events contributed to the Flight 51-L accident.

Discussion

The discussion of the assembly of the aft field joint on the right hand Solid Rocket Booster is drawn from the "STS 51-L SRB Joint Mate Review Team" report [33a] The report was provided to Committee staff during the Committee's trip to KSC on June 6, 1986.

There were 24 Solid Rocket Booster sets (48 SRBs) stacked prior to STS 51-L. The stacking experience of the technicians involved in STS 51-L ranged from 5 to 20 stacking operations. Sixty percent of the technicans and all of the supervisory personnel, including lead technicians, had participated in the 14 stacking operations performed since the Shuttle processing contract was awarded to Lockheed. Thiokol managed the stacking operations for Lockheed under a subcontract. The NASA Accident Review Team found that all personnel assigned to the stacking of STS 51-L were experienced and qualified to perform their assigned tasks.

Aft segment receiving inspection and processing in the Rotation, Processing and Surge Facility (RPSF) was normal. No problems were reported relative to the aft segment clevis during offload from the railcar, mate to the aft skirt, aft booster assembly, or in preparation for transfer to the Vehicle Assembly Building (VAB) for stacking. Some surface defects were identified on non-sealing surfaces of the aft segment clevis, but were found not to exceed the specification in the Operations and Maintenance Requirements Specification (OMRS) document. There were no defects identified in the clevis O-ring grooves. The aft segment was processed normally in the RPSF to prepare for stack.

[33a] NASA, J. Robert Lang, KSC, "STS 51-L SRB Joint Mate Review Team Report," March 18, 1986.

A problem was reported at the 165–168 degree location where a segment case-to-insulation bondline separation 0.109 inch in depth (longitudinally) was found. The OMRS document specifies no separations in excess of 0.050 inch. A Material Review Board (MRB) repair was approved and the separation was filled with an asbestos float-filled, liquid epoxy resin sealant. This repair is standard for this type of separation, and has been performed on numerous segments. Some surface defects were identified on the tang, but none were found to exceed specification.

The right aft booster assembly was transferred from the RPSF directly to the VAB transfer aisle. Positioning of the aft booster assembly on the Mobile Launch Platform (MLP) holddown posts was normal. One iteration of shimming was performed and the subsequent holddown post strain gauge output indicated proper distribution of aft booster assembly loads.

Holddown hardware was installed and stud tensioning began with ultrasonic measurement of stud initial lengths. A problem was reported at holddown post #1 when ultrasonic measurements indicated a stud length twice the actual. The stud and associated hardware at post #1 were removed for offline bench testing. The problem was isolated to a faulty ultrasonic transducer. While awaiting replacement hardware, studs at holddown posts #3 and #4 were tensioned satisfactorily. The stud at holddown post #2 was tensioned but adequate margin was not attained.

This problem in tensioning studs at holddown posts #1 and #2 led to a revision in the schedule. All left hand Solid Rocket Motor segments were stacked while problems on the right hand side were resolved. This procedure had been employed in one-third of the previous stacking operations and was not an uncommon method of stacking.

After installation of the replacement hardware at holddown post #1, studs at posts #1 and #2 were tensioned. The replacement stud at post #1 was brought up to satisfactory tension, but concerns over stud tension at post #2 prompted engineering to request that a problem report be generated.

Engineering determined the tension (approximately 690,000 lbs.) was adequate for SRB stacking, but marginal for launch loads. Therefore, stud removal and replacement was planned after SRB stack but prior to Orbiter mate.

While holddown post stud tensioning proceeded, preparation and inspection of the aft segment clevis was put in work. No problems were identified on the aft segment clevis during this inspection. Since a stacking delay was evident, the clevis was secured and sealed to maintain inspection integrity until stacking could resume.

A Solid Rocket Motor configuration change was released as a result of a handling incident. The SRM–25 left forward center segment was damaged during processing in the RPSF.[34] Deviation Approval Request (DAR) Number RWW–376R1 was approved to replace the damaged segment with a left forward center segment for SRM–26 motor set. In order to prevent flight performance imbalance, the right aft center segment was also reassigned from SRM–

[34] Mishap Report 86–0024, R.G. Smith, "Report of SRM Handling Ring Mishap Investigation Board, Dec. 13, 1985.

26 to the SRM-25 flight motor set. The SRM-26 right aft center segment was transported to the VAB for stacking.

While this segment was outside the VAB, a storm occurred. A Problem Report was generated as rain water was reported leaking from under the segment protective covers. The segment was brought into the VAB transfer aisle and hoisted for pre-stack inspection. At this time, all visible moisture was removed from the aft surfaces of the segment. Inspections were performed and no problems were identified as a result of rain water intrusion. The MRB repair of the separation of insulation from the segment case located at 165/168 degrees was reinspected and final acceptance was verified. A complete inspection of tang surfaces and aft insulation surfaces was performed and no problems were identified.

The use of this segment violated assembly procedure which requires that the segments be protected from direct water entry and it should not have been employed. While there is no evidence of a direct connection to the joint failure, the decision to use this aft center segment was a compromise that need not have been made.

The aft segment clevis diameter was measured at six locations and corresponding measurements were taken of the aft center segment tang. Measurements indicated that a potential for interference existed along the 0/180 and 30/210 degree axes where the tang diameter was larger than that of the clevis. The normal procedure for changing the shape (ovality) of the tang was initiated. The procedure calls for reconfiguration of the segment lifting beam from a four-point to a two-point lift configuration to decrease the tang outside diameter along the axis of interference. The procedure was followed and after stabilization, a decrease in tang diameter of 0.178 inch was measured along the axis of potential interference. Shuttle Processing Contractor (SPC) engineering was called on to evaluate the latest overall characteristics of the joint. At that time the aft center tang was larger in diameter than the aft segment clevis by more than 0.31 inches along the two axes, 0/180 and 30/210.

These measurements still indicated a potential for interference based upon normal KSC experience. SPC engineering determined that additional deflection of the aft center segment case was necessary and prescribed installation of the SRM Circumferential Alignment Tool along the 16/196 degree axis of the tang. The Circumferential Alignment Tool was installed and maximum hydraulic pressure was applied (1200 psig), producing a deflection of 0.196 inch. Later an unspecified torque on the Circumferential Alignment Tool tension rod nut produced an additional deflection of 0.040 inch. This additional torque caused an additional load and exceeded the safe working limit of the tool. Technicians noticed an increase in hydraulic pressure on the pumping unit gauge to 1300 psig at the time torque was applied. This pressure indicates a force of up to 3250 pounds may have been applied to the segment case. Currently, a force of 5000 pounds may be applied to the segment case. The safety limits of the Circumferential Alignment Tool were exceeded (safety factor reduced to 1.2), but the force applied to the segment case was still well below the established maximum. However, the procedure was determined to be inappropriate by the post-accident investigation.

The alignment tool used on the aft center segment of STS 51-L's right hand Solid Rocket Motor is now considered inappropriate by NASA due to the concentrated loads applied at two points. A new alignment tool is now being designed. However, the use of this tool did not appear to have contributed to the STS 51-L's accident.

With the Circumferential Alignment Tool installed, the right aft center segment was hoisted from the transfer aisle and positioned above the aft segment in the VAB High Bay. Installation of primary and secondary O-rings was performed, and no problems were identified. Closeout photographs were then taken showing the O-ring and zinc chromate putty installation.

The joint mating operation proceeded with final inspection of the greased joint surfaces. No problems were identified during engagement of the tang into the clevis, aided by the nearly co-planar relationship of the mating surfaces (within 0.15 inch). The joint mate was completed with installation of all clevis pins and pin retainer clips per the normal procedure. No difficulties were encountered.

After disconnection of the segment lifting beam, the SRM field joint leak test was performed. Following the 200 psig pressurization, the 50 psig decay test was performed and zero pressure decay was recorded, indicating successful assembly of the joint (maximum allowable decay is 1.0 psig over a 10 minute period).

Field joint closeouts were performed in the normal fashion. No problems were reported during pin retainer band and cork insulator installation. Data also indicated normal application of the bead of grease around the seam of the joint. Installation of the systems tunnel floor splice plate across the field joints at the 90 degree location completed the closeout.

Because of its unique design, the clevis of the aft case must always be used as the field joint at the forward end of the aft segment. It was previously flown on the left booster segment on STS 51-C. It was also utilized in qualification test motor QM-4 which was static test fired at Thiokol's Utah plant.

The field joint tang of the STS 51-L aft center segment (serial number L60 had flown previously as forward center segment to aft center segment field joint tang on the left booster on STS 41-D.

In a memo to J. Harrington of NASA's Data and Design Analysis Task Force on February 24, 1986, the Chairman of the SRB Joint Mate Review Team noted the conclusion that the 200 psig O-ring seating operation could produce a blowhole in the putty. Such a blowhole would not be known prior to launch. Since the putty is intended to provide a heat shield to protect the O-rings, the O-rings would be unprotected in cases where blowholes occurred.

6. SUMMARY OF LAUNCH OPERATIONS

Issue 1

How was the decision to launch STS 51-L arrived at and why was it wrong?

Findings

1. The Flight Readiness Review for STS 51-L was conducted in accordance with established procedure.

2. The decision to launch STS 51-L was based on a faulty engineering analysis of the SRM field joint seal behavior.

3. Compounding this erroneous analysis were serious ongoing weaknesses in the Shuttle Safety, Reliability, and Quality Assurance program which had failed to exercise control over the problem tracking systems, had not critiqued the engineering analysis advanced as an explanation of the SRM seal problem, and did not provide the independent perspective required by senior NASA managers at Flight Readiness Reviews.

4. The initial response of Marshall managers to the attempts of Thiokol engineers to raise the issue of temperature effects on the SRM seals caused Thiokol management to discount proper technical concerns and engineering judgement in their recommendation to launch.

5. The Director of Marshall's Shuttle Projects Office may have violated NASA's Flight Readiness Review policy directive by failing to report the results of the January 27 teleconference to the Associate Administrator for Space Flight.

6. The decision of the STS Program Manager to launch despite the uncertainty represented by ice on the Fixed Service Structure was not a prudent effort to mitigate avoidable risks to the Shuttle.

7. The Launch Director failed to place safety paramount in evaluating the launch readiness of STS 51-L.

8. No launch should have been permitted until ice was cleared from the platform leading to the pad escape system.

9. Ice Team personnel and Rockwell contractors properly conveyed their inability to predict the post-ignition behavior of ice.

10. Post-flight analysis indicated that ice did not exhibit the behavior predicted by analysis, and that ice traversed a distance sufficient to strike the Shuttle during lift-off.

11. Failure to enforce a clear requirement for definite readiness statements contributed to failures in communication between NASA and its contractors during launch preparations.

Discussion

Significant in the loss of Challenger was NASA's decision to launch the Shuttle on January 28. The Rogers Commission and the Committee investigation found sufficient evidence to indicate that STS 51-L should not have been allowed to lift off until a number of problems had been corrected. The Committee has examined documentation made available to the Rogers Commission and has reviewed recordings made of conversations among personnel in KSC Firing Rooms on January 27 and 28 in developing its analysis.

What seems evident in the Committee's review of this material is that clear indications existed on the morning of January 28 arguing that a launch of the Shuttle vehicle would not be a prudent decision. Significantly greater risks were present for this launch attempt than were usually found during a launch of the Shuttle. Despite these signals, some of which reached officials with the authority to delay the launch, STS 51-L was allowed to proceed. The Committee is disturbed that expected safeguards in the launch decision process failed to operate.

Specifically, this section examines the inability of the Flight Readiness Review procedure to compensate for poor technical anal-

ysis in preparing the Shuttle system for launch. Also, the efforts initiated by Thiokol engineers to delay the launch until SRM seal temperatures had risen were unsuccessful, nor were their arguments conveyed to the Associate Administrator for Space Flight, as NASA policy apparently requires. Finally, the heavy ice on the pad Fixed Service Structure led NASA's ice team leader to recommend that the launch be scrubbed, but his objections were apparently never conveyed to the STS Program Manager. The Committee concludes that sufficient warning of the risks to STS 51-L was available, and the launch therefore should not have occurred. Readers are directed to Section VIII-A of this report for a complete discussion of each of these areas.

NASA has developed a highly involved procedure to prepare a Shuttle mission for flight. Much of this preparation is discussed in Section VI-A.2.b. In the period immediately proceding a launch, project and program managers participate in a number of meetings that together are known as the Flight Readiness Review. In the case of the Solid Rocket Motor, the apparent cause of the accident, eight levels of review were required to certify the flight readiness of the STS 51-L hardware. (See Table I for date and scope of these reviews)

Flight Readiness Reviews employ the so-called "delta review" concept, meaning that the data presented only represents those elements on the previous flight that fall outside the expected performance of the hardware. The responsible project or program manager must then explain the failure to the satisfaction of the review board and describe the steps that have been taken to assure that the situation will not recur on the upcoming flight. In the case of STS 51-L, however, this concept permitted the SRM seal erosion problem to evade scrutiny. STS 61-C, the mission immediately proceding 51-L, did not fly until halfway through the 51-L FRR cycle. Thus, there was no previous mission to obtain data from Only at the last stage of the cycle, at the L-1 review, did the Associate Administrator learn that the SRM seal erosion problem had been noted again. Mr. Mulloy's presentation characterized the situation as "within the experience base," according to Thiokol's Mr. McDonald.

The history of SRM seal erosion demonstrates the effect that faulty engineering analysis has on the Flight Readiness Review process. Thiokol and Marshall engineering personnel declared the seal erosion problem to be "acceptable," even though the seal design clearly recognized that the elastomeric O-ring seals were not designed to stand up to propellant gases during flight. Relying on a computer model of the situation and a limited battery of tests, Marshall continued to present the situation in Flight Readiness Reviews as "within the experience base;" that is, the deterioration in the seals was no worse than previous cases and thus no concern was warranted.

It is the conclusion of the Committee that the Flight Readiness Review operated as well as its design permitted in the case of STS-51-L. See Section VI-B.1.b.3. It seems clear that the process cannot compensate for faulty engineering judgement among participants. Had the engineering analysis led Marshall to a different conclusion about the severity of the SRM seal erosion problem, the system

would have reacted to these concerns long before the 51-L Flight Readiness Review.

If the Flight Readiness Review process did not fail, however, why was STS 51-L launched? The Committee is seriously concerned by the fact that information indicating that the SRM seals might fail became available in time to delay the launch, and yet these concerns were overridden. Engineers from Marshall and Thiokol argued for hours on the night of January 27 regarding the effect of temperature on the performance of the seals. In the end, Thiokol managers chose to recommend that the launch proceed over the objections of their engineering staff.

In hindsight, it is unfortunate that Thiokol engineers did not present their objections in terms of developing a new launch commit criteria on the SRM joint seal temperature. Doing so would have required that the STS Program Manager would have had to listen to the engineers' presentation. It would also have guaranteed that a more rigorous analysis of the situation would have been forthcoming, simply to explain why the situation had been allowed to continue for so long. Even so, the Committee's investigation indicates that these discussions should have been brought directly to the attention of the Associate Administrator for Space Flight by Marshall's Shuttle Project Office Director. The Committee's investigation also questions whether doing so would have altered the decision made on January 28. (See Section VI-B.1.b.4)

The question remains: Should the engineering concerns, as expressed in the pre-launch teleconference, have been sufficient to stop the launch? The Committee concludes the answer is yes. Thiokol's recognized expert on SRM seals had evidence he believed conclusive and sufficient. His opinion, in the absence of evidence to the contrary, should have been accepted until such time as better information became available.

Finally, the Committee examined the taped conversations among NASA and Rockwell personnel discussing the ice that covered the pad's Fixed Service Structure on January 28. Because the temperature dropped below freezing, NASA had permitted critical water systems on the pad to run during the night. The pad drainage system could not handle the water flow, and allowed water to spill out onto the gantry platforms and freeze.

NASA personnel were sent to the pad to examine the situation and determine whether the situation posed a threat to the Shuttle. What they found was described by Rockwell personnel as "something out of Dr. Zhivago." Icicles hung from platforms and handrails, and could be easily broken off. Sheets of ice covered the gantry platforms, including the platform across which the crew would have to run if it became necessary to use the pad escape system. The ice team leader indicated that he felt the situation was a distinct hazard to the Orbiter thermal protection system, since Main Engine ignition would likely release a great deal of ice debris. Blown by the wind or sucked up by the engines and boosters, the ice could inflict damage on the delicate silica titles that made up the Orbiter heat shield. Asked for his opinion, the ice team leader recommended that the launch be scrubbed until the ice had been removed from the gantry.

Rockwell personnel in Downey, California, expressed similar concerns about the situation after seeing the pad on television. They attempted to determine what would happen to the ice by use of computer modelling, but were not satisfied with the result. Rockwell's chief engineer finally concluded that the situation was little better than "Russian roulette." The company's liaison at KSC noted that the situation was much worse than the threat from ice in the liquid oxygen vent arm, which NASA considered a definite threat to the Orbiter. However, the STS manager, relying on an analysis by engineers at KSC and JSC (using the same model Rockwell found inadequate), decided to launch.

As a whole, the Committee's review of the decision to launch STS 51-L on January 28 indicates a number of questionable practices. It is not clear to the Committee why so many warnings went unheeded by NASA personnel that morning. What is certain, however, is that the Associate Administrator for Space Flight and the Associate Administrator for Safety, Reliability and Quality Assurance should restore a more conservative set of launch rules prior to resuming flights of the Space Transportation System.

Issue 2

Should firing room personnel be allowed to waive launch commit criteria or equipment redlines during a launch countdown without a well-developed technical reason for doing so?

Finding

NASA's management waived its own launch commit criteria on January 28, 1986, without a valid technical reason for doing so.

Discussion

Conversations obtained from the Operational Intercommunication System (OIS), used by the launch team during Shuttle countdowns, indicates that launch commit criteria were waived without sufficient technical justification on January 27 and 28. The Committee reviewed tapes and transcripts which indicate that engineering personnel wrote a waiver for launch commit criteria on the External Tank nose cone temperatures that justified using lower temperatures on the basis of a backup procedure that was invalid.

Should the temperature sensors in the ET nose cone fail, according to Launch Commit Criteria 5.1-4, a secondary procedure correlating data obtained from telemetry channels with a previously derived curve could be substituted. The curve, however, was limited to an ambient temperature range of 40-99 degrees Fahrenheit. Ambient temperatures were outside this range during the countdown, meaning that the backup procedure could not be used. According to the Launch Commit Criteria, exceeding the lower temperature limit could cause "inaccurate ullage pressure readings." Since these pressure readings might be significant in operation of the Shuttle's main engines, inaccuracies might have threatened the safety of the mission. During flight, pressure in the fuel tanks for the main engines is maintained by bleeding off excess gas from the main engine heat exchangers and circulating it back into the External Tank. Misreading the pressure might cause the Orbiter gen-

eral purpose computers to over- or underpressurize the tanks and disrupt fuel flow to the engines.

Also, in discussion with Thiokol personnel during the latter stages of preparing this report, the Committee learned that liquid hydrogen remained in the External Tank throughout the night of January 27. Notwithstanding the effect this had on heat transfer through the aft attachment strut (see Section VII on casing joint design), this indicates that criteria requiring an eight-hour period between tanking cycles may have been violated. This is significant in that, had the tanking cycles been carried out as required, launch of STS 51-L would have taken place in the afternoon of January 28 or the next day. The Committee has not confirmed this possibility.

7. RETRIEVAL, TRANSPORTATION, AND REFURBISHMENT

Issue

Were the motor casings used on STS 51-L damaged as a result of the retrieval, transportation and refurbishment operations following previous launches?

Finding

There was no evidence of damage to the casings or joint due to prior use or preparation for reuse.

Discussion

The aft field joint on Flight 51-L was between two casings that were used previously on STS 51-C. After approximately two minutes of burn-time during the launch operation, the Solid Rocket Boosters are separated from the External Tank, at which time they fall toward the ocean for a considerable distance. Before impact, parachutes are deployed from the Solid Rocket Boosters to slow their decent and minimize impact forces. The Solid Rocket Boosters strike the ocean at a speed of approximately 60 miles per hour (vertical speed component). (There has been no evidence that the casings are distorted by impacting the ocean since the impact loads are low and the cases are still assembled at this point.) The parachutes and boosters are retrieved by divers at sea and both Solid Rocket Boosters are towed back to the Cape by ships. They are towed into a special dock, lifted in slings, conveyed to a wash rack and completely washed down to remove salt water. The casings are made of high carbon steel which is very susceptible to corrosion. The casings are then disassembled, given a visual inspection, and shipped back to Utah for refurbishment. At a plant in Clearfield they are further cleaned and "shot with glass beads" to assure that all foreign contaminants have been removed. The cases are then inspected to determine the dimensions of tang and clevis and for cracking. Inspection for cracks is performed by using a magnetic flux technique. The procedure calls for a test whereby cracks must be of such a minimum size as to be able to withstand four more flight uses without failure. The casings are then subjected to a hydroburst test where they are pressure tested with a mixture of oil and water, to assure sufficient strength to withstand propellant pressure during flight. The hydroburst test is conducted at 1.1 times the maximum expected operating pressure (MEOP). This

gives assurance that the strength can accommodate 10 percent more load than the casings will experience in use. There was no indication that there had been any damage to the casings from Flight 51-C.

C. External Tank

Issue

The External Tank was obviously involved in the accident. Was that involvement a cause or an effect?

Findings

1. The Committee adopts the "Finding" of the Rogers Commission that: "A review of the External Tank's construction records, acceptance testing, pre-launch and flight data and recovered hardware, does not support anything relating to the External Tank which caused or contributed to the cause of the accident."[35]

2. The External Tank ruptured under the forces of a failed Solid Rocket Booster motor. These forces were far outside of any possible design considerations that could have been applied to the External Tank.[36]

Discussion

The 154 foot long, 27 foot diameter external disposable fuel tank contains the liquid oxygen and liquid hydrogen used by the Space Shuttle's three main engines. Structurally, the External Tank (ET) serves as the keel, or backbone, of the Space Transportation System. The two Solid Rocket Boosters are attached to the External Tank near the front and rear of the ET. The Space Shuttle Orbiter is also attached to the ET at one forward, and two aft attachment points.

The Committee was interested in the findings of the Rogers Commission concerning the External Tank because it was obvious to all that the External Tank was directly involved in the accident.

The Rogers Commission investigated five potential faults or failures of the External Tank which could have contributed to the accident.[37]

They are:

Premature detonation of the External Tank Range Safety System;

A structural flaw in the tank;

Damage at lift-off;

Structural overload;

Overheating.

The Committee is satisfied that the Range Safety System on the External Tank did not cause the STS 51-L accident because:[38]

There is no flight data to support premature detonation of the ET range safety package;

---

[35] Rogers Commission Report, Volume I, p. 42.
[36] Ibid., Volume II, p. L-23.
[37] Ibid., Volume I, p. 41.
[38] Ibid., Volume II, p. L-16.

The photographic evidence does not support premature detonation;

Most of the explosive charges included in the ET Range Safety System were recovered with the ET wreckage, undetonated.

The Committee affirms the Rogers Commission finding that there is no data to support structural faults which might have propagated to a size which could have caused catastrophic failure of the External Tank.[39]

The evidence from the Ice Team examination, and photo analysis and television monitoring of the launch give no indication whatsoever that there was any damage to the External Tank during launch and lift-off.

The examination of all flight data indicates that the maximum structural load on the External Tank on STS 51-L was less than 80 percent of the allowable design load from launch until the final explosion.[40]

The Committee also affirms the Rogers Commission finding that the evidence does not support any theory of independent overheating of the External Tank as a cause of the accident.[41]

The failed joint in the Solid Rocket Motor permitted the burning solid propellant gases to escape in the direction of the aft External Tank attachment strut. Temperatures and velocity of these gases caused a rapid erosion and deterioration of the aft strut to the point where it failed structurally under turbulence and maneuvering loads. There were no deficiencies in the design of the external strut. However, the strut was not designed to withstand the "blowtorch" effect of the propellant hot gas stream. During its investigation, the Committee staff visited Martin Marietta's External Tank assembly plant in Louisiana. An issue raised during this visit was whether or not the strut could be relocated such that in the event of another joint failure in that vicinity, the strut would not be damaged. It was learned that relocating the aft strut created more problems that it solved. Furthermore, it was also learned that the gas stream would have almost instantly cut through the insulation on the External Tank and destroyed it anyway. After the failure of the aft strut, the flame continued to bear on the bottom of the External Tank, breaching that tank at the joint of the aft dome. This caused the liquid hydrogen to escape from the tank. Once the flame had penetrated the tank at the weld of the aft dome the failure of that weld spread rapidly and completely around the tank's diameter severing the dome from the rest of the tank. The burning hydrogen ignited by the flame then caused the External Tank to act much like a rocket and created an upward thrust. The right Solid Rocket Booster without the attachment strut to the External Tank, rotated around its long axis.[41a] In so doing, it may have

---

[39] Ibid., Volume I, p. 42.

[40] Ibid.

[41] Ibid.

[41a] It should be noted that the right Solid Rocket Booster did not swing outward at the bottom and cause the nose of the booster to collide with the External Tank as had originally been thought. For this to have happened, the right Solid Rocket Booster would have extended outward at the bottom at a wide angle that is not supported by any of the photographic or telemetric evidence.

jammed the upper Solid Rocket Booster to the External Tank attachment structure and caused it to fail. That structure is located on a large cross beam in the intertank which in turn possibly damaged the cross beam. The other distinct possibility is that the burning hydrogen forced the External Tank upward and caused the cross beam to be damaged and at the same time, caused a rupture of the oxygen tank. With the massive release of energy from the burning of the hydrogen/oxygen, the Shuttle system completely broke apart.

NASA's Accident Analysis Team determined that the Orbiter was destroyed by aerodynamic forces beyond design limits, not by the actual explosion of the External Tank. The report stated,

> All fractures and material failures examined on the Orbiter, with the exception of the SSME's, were the result of overload forces and they exhibited no evidence of internal burn damage or exposure to explosive forces.[41b]

D. Crew Survival

Issue

Was the accident of STS 51-L on January 28, 1986, survivable?

Findings

In the case of the tragic loss of the Space Shuttle Challenger and her crew on January 28, 1986, the Committee is convinced that the accident was not survivable.

Discussion

During the first two minutes and eight seconds of Shuttle flight the two Solid Rocket Boosters provide approximately three million pounds of thrust. That thrust is transferred to the External Tank through the forward Solid Rocket Booster/External Tank attachment structure. The thrust is then transferred to the Orbiter through the External Tank. While NASA has established a "Fast-separation sequence" to allow the Orbiter to separate from the External Tank and Solid Rocket Boosters, the engineering analysis indicates that if separation was attempted while the Solid Rocket Boosters were still firing, the Orbiter would "hang-up" on the forward attachment structure. This would lead to a violent maneuver which would greatly exceed maximum aerodynamic loads on the Orbiter with resulting structural failure and loss of Shuttle and crew.

During the course of hearings before the Committee on the accident the question of survivability was frequently raised. The Committee accepts the view of Captain Robert L. Crippen, who informed the Rogers Commission:

> I've said this before publicly, and I'll say it again, I don't think I know of an escape system that would have saved the crew from the particular incident that we just went

---

[41b] NASA, Leslie Kampschror, "Space Transportation System Mission 51-L Structural Reconstruction and Evaluation Report," Enclosure 8 in "STS 51-L Data and Design Analysis Task Force Search, Recovery and Reconstruction Team Report," Volume IV, Rev. A, May 8, 1986, p. 3.

through. I don't think it is possible to build such a system.[42]

Specifically, the Committee finds that the Space Shuttle System was not designed to survive a failure of the Solid Rocket Boosters during the first 2 minutes of flight; that is, until all the solid rocket propellant fuel has been expended. There were no corrective actions that could have been taken once the boosters ignited. The Challenger was not equipped with any means for separation during the first two minutes of flight. In addition, the crew did not have any means to escape from the Orbiter during this first-stage ascent. Neither the Mission Control Team nor the 51–L crew had any warning of impending disaster. Even if there had been warning, there were no actions that could have been taken to save the crew.[43] (The issue of launch abort and crew escape is discussed in Section VI.A.2.d.)

Joseph P. Kerwin of NASA's Johnson Space Center summarized the circumstances in a memo to Rear Admiral Richard H. Truly, Associate Administrator for Space Flight. The undated memo read as follows:

> DEAR ADMIRAL TRULY: The search for wreckage of the Challenger crew cabin has been completed. A team of engineers and scientists has analyzed the wreckage and all other available evidence in an attempt to determine the cause of death of the Challenger crew. This letter is to report to you the results of this effort.
>
> The findings are inconclusive. The impact of the crew compartment with the ocean surface was so violent that evidence of damage occurring in the seconds which followed the explosion was masked. Our final conclusions are:
>
> The cause of death of the Challenger astronauts cannot be positively determined;
>
> The forces to which the crew were exposed during Orbiter breakup were probably not sufficient to cause death or serious injury; and
>
> The crew possibly, but not certainly, lost consciousness in the seconds following Orbiter breakup due to in-flight loss of crew module pressure.
>
> Our inspection and analyses revealed certain facts which support the above conclusions, and these are related below:
>
> The forces on the Orbiter at breakup were probably too low to cause death or serious injury to the crew but were sufficient to separate the crew compartment from the forward fuselage, cargo bay, nose cone, and forward reaction control compartment. The forces applied to the Orbiter to cause such destruction clearly exceed its design limits.
>
> The data available to estimate the magnitude and direction of these forces included ground photographs and measurements from onboard accelerometers, which were lost two-tenths of a second after vehicle breakup.

---

[42] Rogers Commission Report, Volume V. p. 1431.
[43] Ibid., Volume I, p. 180.

Two independent assessments of these data produced very similar estimates. The largest acceleration pulse occurred as the Orbiter forward fuselage separated and was rapidly pushed away from the External Tank. It then pitched, nose-down and was decelerated rapidly by aerodynamic forces. There are uncertainties in our analysis; the actual breakup is not visible on photographs because the Orbiter was hidden by the gaseous cloud surrounding the External Tank. The range of most probable maximum accelerations is from 12 to 20 G's in the vertical axis. These accelerations were quite brief. In two seconds, they were below four G's; in less than ten seconds, the crew compartment was essentially in free fall. Medical analysis indicates that these accelerations are survivable, and that the probability of major injury to crew members is low.

After vehicle breakup, the crew compartment continued its upward trajectory, peaking at an altitude of 65,000 feet approximately 25 seconds after breakup. It then descended striking the ocean surface about two minutes and forty-five seconds after breakup at a velocity of about 207 miles per hour. The forces imposed by this impact approximated 200 G's, far in excess of the structural limits of the crew compartment or crew survivability levels.

The separation of the crew compartment deprived the crew of Orbiter-supplied oxygen, except for a few seconds supply in the lines. Each crew member's helmet was also connected to a personal egress air pack (PEAP) containing an emergency supply of breathing air (not oxygen) for ground egress emergencies, which must be manually activated to be available. Four PEAP's were recovered, and there is evidence that three had been activated. The nonactivated PEAP was identified as the Commander's, one of the others as the Pilot's, and the remaining ones could not be associated with any crewmember. The evidence indicates that the PEAP's were not activated due to water impact.

It is possible, but not certain, that the crew lost consciousness due to an in-flight loss of crew module pressure. Data to support this is:

The accident happened at 48,000 feet, and the crew cabin was at that altitude or higher for almost a minute. At that altitude, without an oxygen supply, loss of cabin pressure would have caused rapid loss of consciousness and it would not have been regained before water impact.

PEAP activation could have been an instinctive response to unexpected loss of cabin pressure.

If a leak developed in the crew compartment as a result of structural damage during or after breakup (even if the PEAP's had been activated), the breathing air available would not have prevented rapid loss of consciousness.

The crew seats and restraint harnesses showed patterns of failure which demonstrates that all the seats were in place and occupied at water impact with all harnesses locked. This would likely be the case had rapid loss of consciousness occurred, but it does not constitute proof.

Much of our effort was expended attempting to determine whether a loss of cabin pressure occurred. We examined the wreckage

carefully, including the crew module attach points to the fuselage, the crew seats, the pressure shell, the flight deck and middeck floors, and feedthroughs for electrical and plumbing connections. The windows were examined and fragments of glass analyzed chemically and microscopically. Some items of equipment stowed in lockers showed damage that might have occurred due to compression; we experimentally decompressed similar items without conclusive results.

Impact damage to the windows was so extreme that the presence or absence of in-flight breakage could not be determined. The estimated breakup forces would not in themselves have broken the windows. A broken window due to flying debris remains a possibility; there was a piece of debris imbedded in the frame between two of the forward windows. We could not positively identify the origin of the debris or establish whether the event occurred in flight or at water impact. The same statement is true of the other crew compartment structure. Impact damage was so severe that no positive evidence for or against in-flight pressure loss could be found.

Finally, the skilled and dedicated efforts of the team from the Armed Forces Institute of Pathology, and their expert consultants, could not determine whether in-flight lack of oxygen occurred, nor could they determine the cause of death.[44]

E. Sabotage

Issue

Could the accident have been caused by sabotage, terrorism, or foreign covert action?

Finding

The Committee is convinced that there is no evidence to support sabotage, terrorism or foreign covert action in the loss of the Challenger.

Discussion

The Committee carefully reviewed all of the evidence, classified and unclassified, to ensure that there was no sabotage associated with the loss of the Space Shuttle Challenger.

Committee staff met with the Director of Safety, Reliability and Quality Assurance, and the Director of Protective Services and his staff to review the National Resource Protection Plan for the Kennedy Space Center. The Committee is concerned with the vulnerability of the Space Transportation System and endorses the efforts being taken by NASA to provide adequate protection to all elements of the system.

F. Additional Avenues of Investigation

Issue

Could the accident have been caused by some failure other than failure of the joint between the casings?

---

[44] This memo was part of a package release, NASA, 86-100, draft, July 21, 1986.

Finding

As of September 15, 1986, the Committee has not found any credible evidence to support any cause of the Challenger accident, other than the failure of the aft casings joint in the right-hand Solid Rocket Booster. Nor has there been any substantial evidence of a secondary or parallel failure on Flight 51-L.

Discussion

After the accident, the Committee waited until the Rogers Commission completed its work in order not to interfere with the progress being made by the Commission appointed by the President. By the time that work was completed the preponderance of evidence clearly pointed to a failure in the field joint of the right Solid Rocket Booster. However, the Committee was obligated to explore other possibilities which could have led to the same type of failure. Among these possibilities were the following: a failure of the propellant, a structural flaw in the steel casing, and separation of the NBR insulation from the casing. In addition, the Committee was contacted by and sought additional testimony from private citzens who offered their concerns and hypotheses pertaining to the cause of the Challenger accident. These included the following: either a main engine fire or a fire in the main engine compartment of the Orbiter, inadvertent firing of an OMS engine, inadvertent firing of one or more thrusters on the Orbiter, overloading of the aft field joint due to excessive "moment" developed in transit of the Shuttle from the VAB to the launch pad and the use of four separate propellant casings instead of one large casing without field joints.

1. Main Engine Fire

The photographs in Volume I of the Rogers Commission report on pages 26 and 27 indicate a bright spot in the vicinity of the main engine compartment. Photographic evidence is customarily taken to be accurate. In this case, however, it must be realized that the photographs were taken from roughly three miles away and that they were enhanced by computer methods. Computer enhancement has the ability to highlight bright objects and subdue dull ones. In this way, the photographs become distorted, that is, the difference between light and dark becomes unrealistically pronounced. The bright spot in the photographs does, in fact, look like a flame. The second consideration is that the orientation between the Orbiter and the ground where the cameras were is difficult to visualize and leads to erroneous conclusions. During flight the main engines are monitored continuously for changes in pump speed, temperature and pressure. There was no indication whatsoever of a malfunction with the main engines. A fire in the main engine compartment outside of the engines is not credible because of the lack of combustible material to support a fire of any appreciable magnitude. The exception would be a hydrogen leak. But, that was not supported by telemetry data. In addition, NASA has submitted photographs to the Committee from four past successful launches which show the same bright spots. The Committee, therefore, has rejected this as a cause of the accident or as an independent problem.

2. Independent Firing of the OMS Engine or Orbiter Thrusters

The theory that either the OMS engines or the Orbiter thrusters were inadvertently activated and fired is also based on the same photographs stated previously. Those photographs show a bright spot in the same general area where these engines and thrusters are. The Committee has received photographs from Flights 41-G, 61-A, 61-13, and 61-C, all of which show similar "bright spots" in the same location as those seen on Flight 51-L. The Committee is still evaluating the possibility of a second failure in this regard and has requested additional telemetry data from Flight 51-L. Had the thrusters been firing, however, it would have had little impact on the launch of the Challenger. The thrust from these tiny engines is insignificant compared to the thrust from the two Solid Rocket Boosters and the main engines. The inadvertent activation of the OMS engine has been ruled out on the basis of telemetry data received from NASA. NASA has stated that the bright spot seen in the photographs is a reflection from the plume of the Solid Rocket Booster motors. Neither of these possibilities contributed to the Challenger accident.

3. Overloading of the Joint

It is true that in transit from the VAB to the launch pad, the Shuttle system, standing erect on the launch platform and being carried by a crawler, does experience a left-hand turn. At that time, because of the configuration of the Shuttle system, an additional moment, that is force times a distance, is transferred to the field joints including the aft joint on the right-hand Solid Rocket Booster that failed. However, this moment exerts a force which is only 10 percent of the force that the joint receives during other phases of the launch operations. The Committee concluded that this had no impact on the Challenger accident.

4. Insulation Debonding

The Committee investigated the possibility of separation of the insulation from the inside of the motor casings as a potential cause of the Flight 51-L accident. Had the insulation broken lose from the casing, there would have to have been a condition which would have permitted the burning propellant gases to get between the insulation and the casing. Furthermore, there would have to be a continuous gas flow at that point for the propellant gas to transfer a sufficient amount of heat to the casing to cause a failure. This would require an extremely large debonding of the insulation which has never been seen on any Shuttle flight when the Solid Rocket Motors were returned and disassembled for use later. When the Shuttle motors were inspected after usage, what remained of the insulation has always been in place with little damage. A debonding accident would have had to provide tremendous amounts of heat and again would have required a very high flow of the gas into the area where the debonding occurred. That flow of gas would have to be continuous and there is no rationale for envisioning how that could happen. In the case of the Shuttle Solid Rocket Motor design, the pressure acts to maintain the bond between the insulation and the casing, not to remove it. In the absence of these re-

quirements the Committee found that debonding of the insulation was not a cause of the accident.

5. Crack in the Propellant

The Committee investigated whether or not a crack in the propellant could have contributed to loss of the Challenger. A crack in the propellant would have increased the burning surface of the propellant after ignition. This increase in the surface would have resulted in an increase in th thrust from the right Solid Rocket Motor. There was no evidence during the flight of 51-L of a greater thrust in the right Solid Rocket Motor. In additional, a propellant failure would have been more explosive in nature and would not have been observed as one continuous gas flame in a localized area. Consequently, it was concluded that a propellant failure did not contribute to the cause of the accident.

6. Crack in Motor Casing

The Committee was concerned that a crack in the rocket motor casings might have caused the accident if it was located in the same general area where the smoke and flame was observed during launch. All of the casings used on Flight 51-L were hydroproofed at 1.1. maximum expected operating pressure. Had there been a signficant crack in the casing it would have failed the hydroproof test. However, it could be argued that a crack developed between the test and the time the Solid Rocket Motor segments were assembled at the Kennedy Space Center. The failure of cracks under the pressures, such as those contained within the Solid Rocket Motors, would have been a catastrophic failure. The casings would have failed instantly at ignition because cracks in high carbon steel would propagate at a rate near the speed of sound. This is inconsistent with the smoke seen during the early part of the launch, and the lack of smoke or flame up until 58 seconds into the launch. It is also inconsistent with the pieces of the rocket motor casings which were recovered from the ocean which clearly show the abrasion of the hot rocket propellant gases. Consequently, a crack in the casing was ruled out as a contributing cause of the accident.

7. Joint putty temporarily holds and then releases full motor pressure

During the post-accident tests conducted by NASA and Thiokol, it was learned that the performance of the putty used in the joint can be quite variable. In some instances, including temperatures as warm as 75°F, the joint putty can hold back the full operating pressure inside the motor without transferring any of this pressure to the O-rings.[45] In this circumstance, the O-rings will not "seat" and, as the joint "rotates" due to the pressure build-up within the motor, contact can be lost between the O-rings and the metal surfaces they are meant to seal.[46] If the putty were then to release

[45] *Rogers Commission Report*, Volume I, p. 64.
[46] NASA, briefings from staff.

high pressure gases into the joint, these gases could "blow-by" the O-rings, thus causing the joint to fail. This scenario is not a likely failure mode for STS 51-L, because it would produce a leak across a broad area of the joint rather than a small localized leak as observed in the Challenger accident.

VI. DISCUSSION OF CRITICAL ISSUES

A. Technical Issues

1. HARDWARE DEVELOPMENT AND PRODUCTION

a. Problems in Hardware Certification

Issue 1

Having all elements of Space Shuttle flight hardware been adequately certified?

Findings

1. The overall design and certification processes prescribed by NASA for each major element of Space Shuttle flight hardware are very comprehensive.
2. Prior to the STS 51-L accident, in spite of the comprehensive nature of NASA's prescribed design and certification processes, insufficient testing had been conducted to permit an adequate understanding by either Morton-Thiokol or NASA regarding the actual functioning of the Solid Rocket Motor joint. Also, the Solid Rocket Motor had not been adequately certified to meet the natural and induced environmental conditions that are stated in NASA's design standards. The issue of whether or not standards were adequate is discussed in Section VII.
3. The deficiencies in Solid Rocket Motor testing and certification persisted in spite of many reviews of the program by panels of experts: (1) within the manufacturer; (2) within NASA; and (3) from independent, outside groups.
4. These deficiencies in testing and certification of one major element of the Space Shuttle system raise the possibility that other elements of flight hardware (or other subelements of the Solid Rocket Motor) could have similar deficiencies.
5. If NASA is unable to explain why the deficiencies in Solid Rocket Motor testing and certification went undetected by the existing comprehensive set of processes and procedures, the agency will not be able to protect against a similar break-down in its system of checks and balances in the future.

Recommendations

1. NASA should devote more attention to determining why the deficiencies in Solid Rocket Motor testing and certification went undetected, so that appropriate action can be taken to uncover latent problems in existing hardware and to prevent similar problems in future development programs.
2. NASA and its contractors should thoroughly reassess the adequacy of all of the testing and certification that has been conducted

to date on each element of Space Shuttle flight hardware. Where deficiencies are found, they must be corrected.

Discussion

In background briefings for the Committee staff prior to the start of the hearings, NASA described the system of formal reviews that were employed to scrutinize the design and certification of each element of flight hardware.

The review process began with a System Requirements Review in the early 1970's. About 18 months later, each hardware element went through a Preliminary Design Review (PDR). This review was conducted when about 10 percent of the engineering drawings were complete and resulted in approval for the hardware to move into the final design stage. The Critical Design Review (CDR) was held when about 90 percent of the engineering drawings were complete and resulted in an authorization to carry the manufacturing process through to completion. After the end of a detailed test and certification program, NASA conducted a Design Certification Review (DCR) to ensure that all tests and certification results were consistent with specified design requirements and standards.

Because of the extreme complexity of the Orbiter, a series of Configuration Acceptance Reviews (CARs) was also conducted in addition to the standard process of reviews described above. The Phase I CAR was a verification review to ensure that the Orbiter was ready to begin test. Prior to the start of the Orbiter combined system test, several additional incremental test reviews were also conducted. The Phase III CAR was the verification review that resulted in final acceptance of the vehicle for delivery from the manufacturer to NASA.

In Appendix K in Volume II of the Rogers Commission Report, the Development and Production Team discusses in further detail the design and certification processes that were used by each prime contractor. For example, this appendix indicates that Rockwell used a total of 17 design review teams (some divided into as many as 17 subteams) to oversee the design and production work on the Orbiter. The appendix also describes the requirement verification system used by Thiokol for the Solid Rocket Motor as a "closed-loop" system intended to track *each* specification requirement. The system specified the method of verification (analysis, inspection, test, etc.) that was to be used for each program phase (development, acceptance, prelaunch, etc.), along with all applicable requirements of the verification plan. These were tracked through the test plans and reports and then culminated in the issuance of a formal "certificate of qualification". [1]

In addition to this comprehensive system of oversight and review by each prime contractor and each NASA field center responsible for monitoring the work of those contractors, detailed outside reviews of the design and testing programs for each major element of flight hardware also occurred. For example, the Aerospace Safety Advisory Panel regularly reviewed the safety aspects of Space Shuttle flight hardware and annually reported their concerns to

[1] Rogers Commission Report, Volume II, pp. K-12 and K-28.

the NASA Administrator.[2] As another example, NASA Headquarters in 1980 created a Space Shuttle Verification/Certification Committee to thoroughly study the flight worthiness of the entire Shuttle system.[3] This independent committee was chaired by Dr. Walt Williams, NASA's Chief Engineer, and was comprised of recognized experts drawn from the military, private industry, and academia. There was also additional reviews such as a study of the Space Shuttle Main Engine conducted by Professor Gene Covert of MIT.[4]

Finally, before the first flight of the Space Shuttle in 1981, NASA had each contractor and each field center carefully review all of the requirement specifications and certification tests for their flight hardware to ensure that all contract end-item requirements had been adequately certified. Upon determining that all certification requirements had been satisfied, each NASA project and technical manager was requried to sign a Verification Completion Notice (a copy of which is contained in Appendix VI-A of this report). This entire process was then duplicated prior to the first "operational" flight of the Space Shuttle (i.e., STS-5 in 1982). This latter process also culminated in each NASA project and technical manager signing a second Verification Completion Notice (a copy of which is contained in Appendix VI-A of this report.).

Given this comprehensive system of reviews, it is difficult to understand how major inadequacies in design and certification for any element of flight hardware could have gone unnoticed. But it is exactly what happened for the Solid Rocket Motor. Specifically, the Commission concluded that:

> The joint test and certification program was inadequate.

And,

> Prior to the accident, neither NASA nor Thiokol fully understood the mechanism by which the joint sealing action took place.[5]

In addition, the Development and Production Team concluded that:

> Prior to the STS 51-L accident, there was a lack of understanding on the part of MTI [Morton Thiokol Inc.] and NASA of the joint operation as designed.

And,

> JSC 07700, Volume X [the NASA master requirements document for the Space Shuttle program], clearly states the natural and induced environments to which the SRM [Solid Rocket Motor] is to be designed and verified. The field joints . . . were not qualification tested to the full range of the contractually required environments. This led to a lack of complete understanding of the joint design limits.[6]

---

[2] Ibid., Volume I, pp. 160-161.
[3] Ibid., pp. 124-125.
[4] Ibid., Volume II, p. K-24.
[5] Ibid., Volume I, p. 148.
[6] Ibid., Volume II, p. K-30.

Relative to this last point, the NASA requirement documents state that: "The Shuttle Flight Vehicle design shall satisfy the natural environmental design requirements . . .", including air temperature extremes of 20°F to 103°F at "Ferry Sites" and 31°F to 99°F for "Vertical Flight". Also, the requirement documents state that: "Each element of the Shuttle Flight Vehicle shall be capable of withstanding the induced environments imposed during transportation, ground operations, handling and flight operations . . .", including induced Solid Rocket Booster surface temperatures as low as 25°F and induced temperatures as low as 21°F at the point where the aft strut attaches the Solid Rocket Boosters to the External Tank. (Excerpts documenting these temperature requirements are contained in a briefing given by NASA to Joseph Sutter of the Rogers Commission on May 19, 1986, which is reproduced in Appendix VI-A-4 of this report.)

Of principal concern to the Committee is the fact that none of the extensive systems of checks and balances within the Space Shuttle program discovered the lack of adequate testing and certification of the Solid Rocket Motor. This failure of the management and review system indicates to the Committee that other elements of Shuttle flight hardware or other subelements of the Solid Rocket Motor may also be inadequately understood or certified. This will obviously require NASA and its contractors to conduct a careful review of all the testing and certification efforts that have been conducted to date for each element of Space Shuttle flight hardware.

A parallel concern of the Committee is that NASA does not yet know how or why this break-down occurred in this comprehensive system of reviews, checks, and balances. Without such an understanding, the teams that will now be conducting the required reviews of each element of flight hardware will be somewhat disadvantaged because they cannot be certain that they are "asking the right questions" or "looking for the right things." Further, without an understanding of how and why the existing management and control system broke down, NASA will not be able to make the necessary managerial and procedural changes required to be confident that this problem will not reoccur in the future.

Issue 2

Does the Space Shuttle Main Engine have adequate operating margins, and is the "fleet leader" concept adequate to ensure safe operation?

Findings

1. The Space Shuttle Main Engine is an impressive technological achievement. However, it also is one of the higher risk elements of the Space Shuttle system. Anomalous component performance or premature engine shutdown could prove catastrophic to the Space Shuttle and its crew.

2. Some NASA officials familiar with the Space Shuttle Main Engine believe that it should be operated at a throttle setting of 109 percent only in an emergency; others believe the engine could be safely operated at 109 percent on a routine basis.

3. It is widely accepted that the Space Shuttle Main Engine would be safer if its operating margins (for temperature, pressure, operating time, etc.) were increased.

4. The Committee agrees with the sense of Dr. Feynman's concerns with respect to NASA's current, "fleet leader" concept for certifying Space Shuttle Main Engine components, such as high pressure turbopumps, for flight.

5. On a case by case basis, NASA regularly violates its own certification requirements by permitting individual engine components to be used for flight even though they have accumulated an operating time in excess of 50 percent of the two fleet leaders (i.e., in violation of the "2X" rule).

Recommendations

1. NASA should continue its active development program for the Space Shuttle Main Engine. The program should be focused more on increasing operating margins.

2. Because of the safety concerns raised by some knowledgeable officials, NASA should give serious consideration to restricting use of the 109 percent engine throttle setting to emergency situations only. If NASA decides that it needs to use the 109 percent throttle setting for other than emergency situations, the space agency should take whatever actions are required to ensure that adequate margins are present to maintain safety.

3. NASA should closely scrutinize each of the concerns raised by Dr. Feynman regarding the agency's "fleet leader" concept for certifying Space Shuttle Main Engine components. The agency should also closely reassess its practice of selectively violating its "2X" rule for some Main Engine flight hardware elements.

Discussion

The Space Shuttle Main Engine is, very appropriately, described by the Development and Production Team as a "high technology, high power density, state-of-the-art rocket engine."[7] Indeed, the Space Shuttle Main Engine represents a major increase in operating performance over that provided by any other available rocket motor. In his paper Dr. Feynman notes that the Main Engine "is built at the edge of, or outside of, previous engineering experience."[8] The Development and Production Team also observed that the Space Shuttle Main Engine is "a very complex and high risk element of the Space Shuttle system."[9]

It is this last observation that is of more concern here. Specifically, if the Space Shuttle Main Engine were to experience a major problem in flight, the results could be catastrophic. Even premature engine shutdown could prove fatal during certain segments of flight because it would mean that the Orbiter would have to ditch at sea—a maneuver that the Rogers Commission concluded would probably be non-survivable.[10]

---

[7] Rogers Commission Report, Volume II, p. K-3.
[8] Ibid., p. F-2.
[9] Ibid., p. K-3.
[10] Ibid., Volume I, p. 182.

64-420 O - 86 - 4

Therefore, the key question is: how safe is the Space Shuttle Main Engine? An informal review of failures that have occurred during ground testing over the past five years was reported to the Committee staff.[11] It concluded that five of these failures would probably have been catastrophic if they had occurred in flight.[12] It is also of note that each of these failures occurred at an engine thrust setting of 109 percent or greater. However, closer examination of the cause for each failure indicates that most were the result of: poorly installed test instrumentation (engine 2208); an improperly tested "fix" to an engineering problem (engine 2013); the use of "deactivated" components because no others were available (engine 0204); or the existence of a phenomenon that cannot recur because of the adoption of a new safety "red line" in current flight engines (engine 0108). The failure of engine 2308, on the other hand, did uncover a life limit on current engine hardware. That particular engine had accumulated about 20,000 seconds of operation (the equivalent of 40 Space Shuttle missions) in the component which failed (the main combustion chamber). Further, this engine had reportedly logged a significant amount of operating time at a power level of 109 percent.

However, the question of engine safety still remains. The majority of present and past Space Shuttle Main Engine program officials who briefed the Committee staff were personally uneasy at the thought of operating the Main Engine at a thrust setting of 109 percent in anything other than an emergency situation. Specifically, in certain emergency or abort situations, the throttle must be advanced to 109 percent to either reach orbit or to successfully return to the launch site. Under these circumstances, the risks of not using the 109 percent throttle setting (and having to ditch the Orbiter at sea) would obviously be greater than the risks of using that throttle setting. These officials also noted that, at the outset of the Main Engine development program, the 109 percent power setting was referred to as the "Emergency Power Level". As the payload lift performance of the Space Shuttle became increasingly marginal, however, the 109 percent setting was redesignated as the "Full Power Level." Subsequently, several engines were ground tested at the 109 percent power setting for a sufficient duration to certify use of that power setting in normal launch operations.

In addition to Space Shuttle Main Engine program officials at NASA, others have expressed concern with using the 109 percent throttle setting. For example, the Aerospace Safety Advisory Panel voiced concern that "each time NASA flies at 109% we are really pushing the capability of the engine".[13] In his testimony before the Committee, Mr. George Jeffs, President of Rockwell International (manufacturer of the engine) conceded that:

> . . . we don't have a lot of margin at 109 percent. . . . To be comfortable . . . we would recommend that we go to a

[11] Discussions with personnel from Rocketdyne, Pratt & Whitney, and Aerojet General, August, 1986.

[12] These include the destructive failures of engine 0204 in September 1981; engine 2013 in April 1982; engine 2208 in August 1982; engine 0108 in February 1984; and engine 2308 in March 1985.

[13] Discussion with Aerospace Safety Advisory Panel, May, 1986.

> larger throat . . . and that we also add to that the dual manifold gas system. . . ." [14]

All observers agree that the "wear and tear" on an engine operating at 109 percent is substantially greater than when it operates at the more standard 100 to 104 percent throttle settings. However, some NASA officials believe that the successful completion of a traditional certification program involving two engines operating at 109 percent thrust levels is adequate justification for use of the 109 percent power setting for standard missions involving heavy payloads. Other officials, on the other hand, continue to believe that this power setting should be used only in emergencies. If this latter view is adopted, it would mean that the heaviest payloads now planned for launch by the Space Shuttle would have to be moved to expendable launch vehicles. The Committee is not now in a position to accurately predict the programmatic ramifications of such a decision, but the recent national commitment to an enhanced expendable launch vehicle fleet could possibly minimize the negative impacts of a decision to restrict Space Shuttle Main Engine thrust levels to no more than 104 percent.

Though there is substantial disagreement whether or not the Space Shuttle Main Engine should be used routinely at a thrust setting of 109 percent, there is little or no disagreement that the Main Engine would be safer if its operating margins (for temperature, pressure, operating time, etc.) were increased. Indeed, the Development and Production Team noted in its report that one of the formal actions being taken by Rocketdyne in response to the STS 51-L accident is the creation of a "Margin Improvement Board." [15] This board will review and suggest appropriate actions on all recommendations for increased engine operating margins.

Another concern that has been raised regarding the certification of the Space Shuttle Main Engine relates to NASA's use of the "fleet leader" concept. NASA's basic engine certification guidelines require that all components be tested on the ground in two engines for a period of time at least twice as long as the time that those components will accumulate in flight. For example, before turbopumps can be used for four successive flights, the turbopumps in two ground test engines must be tested for the equivalent time that would be required to accomplish eight successive flights. Dr. Feynman cites several problems with this approach. These include:

> The question of what constitutes an "unsuccessful" test? To the Federal Aviation Administration, a cracked turbine blade would constitute a failed test. To NASA, on the other hand, a turbine blade would not be considered to have "failed" until it actually broke in two.[16]

---

[14] Hearings before the House Science and Technology Committee, "Investigation of the Space Shuttle Challenger Accident," 99th Cong., 2nd Sess., Transcript, July 15, 1986, p. 13. (Hereafter referred to as Cmte Hgs).

[15] Rogers Commission Report, Volume II, p. K-27.

[16] After further investigation, the Committee has learned that some qualifying remarks are required for Dr. Feynman's characterization of the FAA engine qualification procedures to be totally accurate. The FAA does not permit cracks in what it calls "critical" engine components. However, cracks located at, or above, the base of a turbine blade are not considered critical by the FAA because: (1) commercial jet engines possess adequate internal shielding to contain any

Continued

The question of whether two "fleet leader" engines represent a better indication of component operating life than a third engine which fails in a lesser time? In other words, should the operating time limits for flight hardware be set at one-half that of the two fleet-leader engines or one-half that of the shortest-lived components?

When a defect is found in a fleet leader engine and a component must be replaced, what engine running time should be used for calculating permissable flight hardware operating times for that component using the "2X" rule: (1) the accumulated operating time up to the start of the final test; (2) the accumulated time as of the end of the final test; or (3) some length of time in between these two extremes? [17]

In the staff review prior to the Committee hearings, NASA officials also noted that the agency frequently violates its "2X" rule for engine flight hardware—permitting components to be used in a particular mission that have not been tested on the ground for twice as long as their intended use in flight. However, these officials noted that this was only done on a "case by case basis" and only for those components that are considered to be highly reliable.

Possibly the most disturbing observation regarding the Space Shuttle Main Engine made by Dr. Feynman in his report is his assertion that: "the Flight Readiness Reviews and certification rules show a deterioration for some of the problems of the Space Shuttle Main Engine that is closely analogous to the deterioration seen in the rules for the Solid Rocket Boosters." If true, this assertion is obviously quite ominous.[18]

b. Recurrent Hardware Problems

Issue 1

What resolutions of inadequacies revealed in the landing gear, tires, wheels, brakes, and nose wheel steering of the landing and deceleration system are required?

Findings

1. The Orbiter landing gear, tires, wheels, brakes, and nose wheel steering, as a system, is experimental, designed to criteria outside any other experience, and uses unique combinations of materials. The original design performance specifications for speed and landing weights are routinely exceeded. The original design did not consider asymmetrical braking for cross wind steering as the normal case although it has become standard practice. Stresses which were not taken into account in the design have surfaced in as yet a very small real world sample.

---

broken blades totally within the engine; and (2) all commercial jet aircraft are designed to fly with one engine inoperative. On the other hand, the FAA does consider as critical any cracks in the "fir tree" region of a turbine blade which is used to attach the blade to the hub of the turbine. (Should a turbine blade break in this region, it may be heavy enough to break through the engine shielding.) This is the region in which cracks are appearing in some Space Shuttle Main Engine turbine blades.

[17] Rogers Commission Report, Volume II, p. F-3.

[18] Ibid.

2. As a consequence, Orbiter landings appear high risk even under ideal conditions, which seldom occur. Exceptional procedural and skill demands are placed upon the pilots to nurse the brakes and tires through every landing. Landing rules have had increasing constraints imposed that hamper operational flexibility and usefulness of the Orbiter.

3. Brake and tire damage have been evident since early on in the program. The Rogers Commission seems very correct in finding the current landing gear system unacceptable. Resolution of landing gear system problems can no longer be put off.

Recommendations

The Committee recommends that NASA:

1. Assemble all of the fragmented studies, analyses, and conclusions on landing gear problems and integrate them into one engineering description of the system as it is now intended to be used. This should include consideration of the basic strength of the struts themselves and their attachments.

2. Write a new *system* specification and match the proposed design improvements to an acceptable reliability and certification specification.

3. Design a test and certification program adequate to meet criteria to fly and to *continue* well into future operations until understanding and confidence in the landing gear system is attained.

4. In anticipation of requirements for a new brake specification, accelerate a program to provide:

Increased brake mass and/or heat sink;

Substantial increase in energy absorption;

Evaluation which weighs the experimental nature of the proposed 65 million foot pound carbon brake and its impact on the system against the penalty of weight of known materials (e.g. steel) for operational confidence.

5. Write updated subsystem specifications to upgrade the landing gear system to acceptable levels of performance to respond to the Rogers Commission's recommendations.

Discussion

The sheer volume of testimony and documentation inevitably gives rise to apparent contradictions. None have surfaced that are assessed as consequential in evaluating the landing gear system problems (NASA's "anomalies") with a view to their solution.

The *main landing gear* (see Figure VI-1) consists of two heavily-loaded, two-wheel struts with two brakes on each and is designed for deceleration only. Roll-out and cross-wind steering correction was originally assigned to the nose wheel steering for the normal case. Each tire, wheel, and brake is supported by redundant fail operational anti-skid brake actuators, control valves, control boxes, and hydraulic power.

The *nose wheel* strut has two wheels on a common axle, no brakes, and is steerable. At the time of the accident nose wheel steering was not permitted except in an emergency because it did not have fail operational or fail safe redundancy. This places additional requirements on the main gear braking system that were a substantive source of main gear problems. Need for correction of

this oversight was most apparent and is the simplest of several difficult landing gear problem solutions. The lack of definitive action in five years of operations is not explained.

The *tires* apparently meet all of their design specifications but are critical for other reasons: (1) If a tire is soft or flat at the time of the nose down load spike (caused by negative lift on the wing, when the nose wheel makes contact) the other tire on that strut will take loads far in excess of its 130,000 pound limit and fail. (2) There is a body of opinion that at almost any time in the landing, one failed tire will assure the failure of its mate. (3) There is no assurance at launch that there is adequate pressure in any of the tires to assure spec performance. (3) Scuffing, cutting, abrading, and wear from spin up, asymetrical braking (cross wind steering), surface roughness, and debris have been more than expected and disallow reuse of most tires (one landing per tire was spec). (4) Anti-skid becomes inactive at 20 knots and damaged brakes will lock up and blow both tires. It would appear that solving a host of other problems will resolve a number of the major tire problems.

The *brakes* by all standards are very large, very light, of conventional configuration, and very experimental because of extensive stretching of materials technolgy by using carbon-beryllium. Brake design is not rigorous, it is very empirical and results are often unpredictable in new designs. The Orbiter brakes incorporate beryllium stators and carbon lined beryllium rotors. Beryllium has low density, high strength, and high heat capacity. Beryllium is very tender and not well behaved at high temperatures. Beryllium has unreliable plastic characteristics at higher temperatures. Use of beryllium in lieu of steel saved perhaps 1000 pounds in the cumulative landing gear weight. The C-5A aircraft uses beryllium rotors and stators.

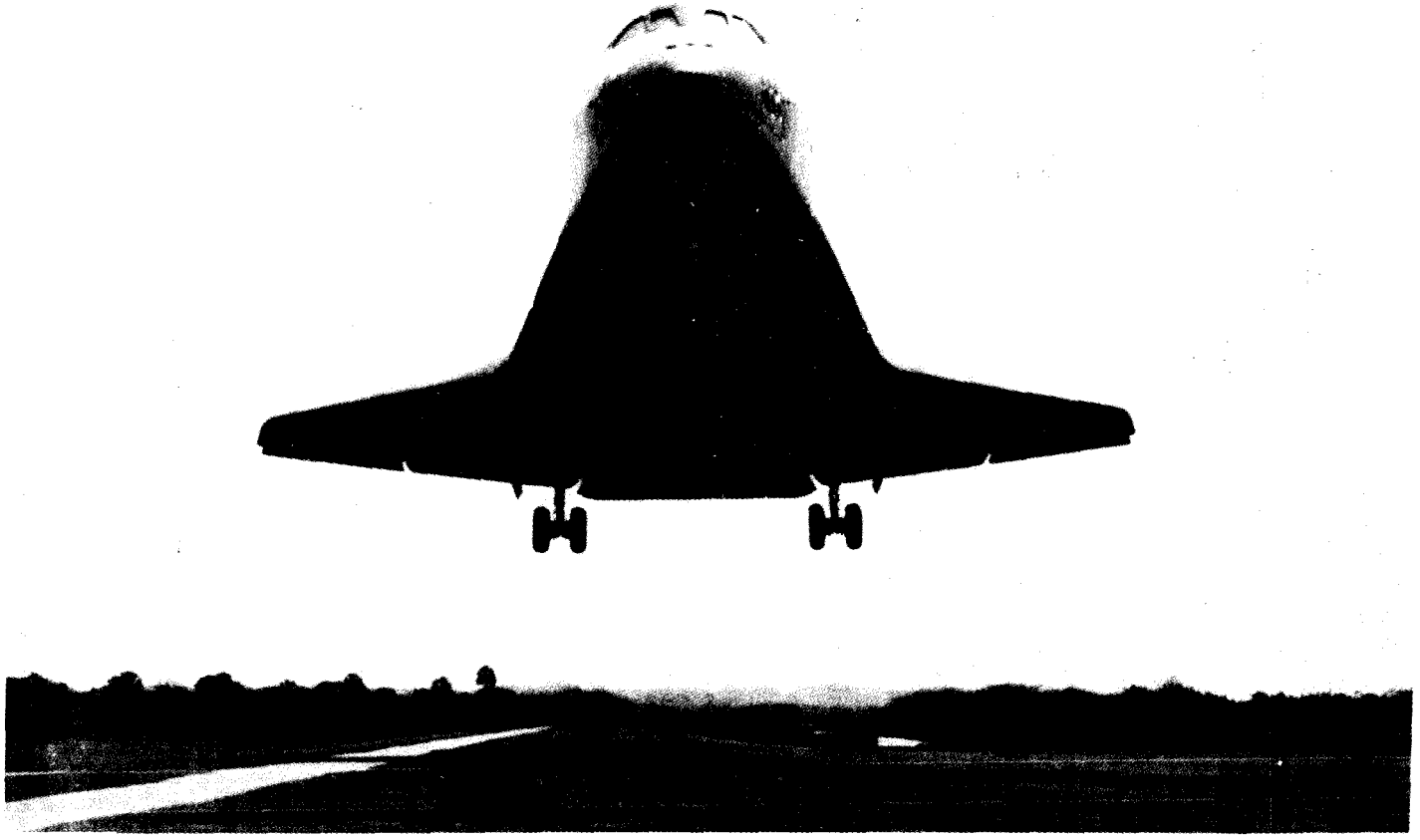


FIGURE VI-1

For the first 23 Orbiter flights, brake damage occurred to varying degrees on 15 flights for a sum of 32 damaged brakes. Failure seemed to favor the right hand gear. A great majority of these damages were not necessarily associated with heavy demands. A very few were caused by approaching the design energy absorption. A case cannot be made against the heavy footed pilot. Regression analysis indicates that there are no significant trends that show brake damage is a function of energy demands within limits, peak demands, or landing weight. It appears that the brakes, relieved of asymmetric steering loads, will approach design energy absorption specifications for one landing but those specs are inadequate to current requirements for repetitive operations.

The primary source of failure is cracking and fragmentation damage arising from hot spots and chatter or dynamic loads from vibrations. It was conjectured that these can be worse if the brake is lightly used ("not tightly clamped"). The pilots were given the astounding instructions to never release a brake once applied because if the brake is reapplied the loose fragments will destroy it and cause seizure.[19]

From the history unfolded to the Commission and Committee investigation, it is not difficult to understand why the landing gear system is marginal in today's operation. The original Orbiter design criteria were quite different and are in part:

System	Original Orbiter	Current Orbiter
Drogue chute	Primary deceleration	Deleted.
Nose wheel steering	Asymmetric loads	Delete-emergency only.
Tires	One landing per tire	Same.
Landing weight, abort (worst case)	225,000 lb	240,000 lb.
Lightweight wheel brakes	Emergency, drogue back up and final deceleration.	Primary deceleration and stop.
Wheel brake life	5 landings dynamometer certified	Same.
	(Typical:	
	Bombers	40 landings.
	Airliners	100 landings
		5 max landings.
		1 emergency.
		(all without thrust reversal).

The whole operational load to decelerate, stop, and steer fell on what was originally the emergency backup brake system. The five landing design is impossible to fine tune to that degree and may yield only a design of imminent failure. It follows that every landing with the now increased normal and abort landing weights is an engineered emergency. How much the increased demands and weight have intruded into the landing gear strut design factor of safety and margin are unknown, but certainly a concern.

In summary, weight savings and inability to retrofit (i.e. larger tires and/or brakes) collided with the then state of the art resulting in the normal high risk of landing being compounded. It is a tribute to the pilots that they were able to carry such a tender system this far. Redundancy of tires and wheels has never been

[19] Rogers Commission, "Meeting of Challenger Commission; NASA, JSC", Halloway, T. et al., March 9, 1986, p. 313; and Bobko, K. et al., March 24, 1986, p. 152.

design practice. A successful landing could be made with both tires and brakes gone from one side with a steerable nose wheel and laterial stability from the opposite truck, if on a hard surface (concrete). The lake bed and stabilized overrun zones would probably be another matter.

Testimony implies that full-time nose wheel steering and higher capacity brakes are a top priority requirement for return to operations. Reference is made several times to a replacement 55 million foot pound all carbon brake in lieu of the 42 million foot pound carbon-beryllium brake. Rockwell International gave testimony that a 65 million foot lb. brake was in work.[20] To maintain the equivalent BTU's per pound, the carbon brakes must peak at much higher temperatures. This poses a new stress on the temperature environment of the tires and wheels. Carbon brakes used on the Concorde, B747, 757, 767 and C-5B are said to be experiencing dynamic failure modes. Carbon brake design has a better data base to work from than the beryllium but such a new brake will continue to be experimental and developmental in nature in the Orbiter application.

Issue 2

What actions should be taken relative to other recurrent problems with flight hardware?

Finding

There have been many instances of in-flight anomalies and failures of other elements of Space Shuttle hardware, some involving mission critical pieces of equipment. Some of these past problems have been corrected while others have not.

Recommendation

NASA should ensure that before reinstituting Space Shuttle flight operations, it fully understands and has corrected all instances of serious in-flight anomalous behavior or failures involving mission critical pieces of flight hardware.

Discussion

Throughout the Space Shuttle program, there have existed a number of recurrent hardware problems in addition to those discussed elsewhere in this report. Some have already been solved. For others, new hardware has been ordered that hopefully will resolve the problem, while still other problems remain unsolved. Listed below are some examples of recurrent problems that have occurred with elements of Space Shuttle flight hardware:

Anomalous behavior of Space Shuttle Main Engine hydraulic actuators—on two occasions (STS-41D and STS-51F). This resulted in engine shutdown just prior to liftoff.

Numerous instances of failure of temperature and pressure sensors within Space Shuttle Main Engines—in one case (STS-51F) this resulted in the premature shutdown of a good engine

[20] Cmte Hgs, Transcript, July 15, 1986, p. 9.

during flight.[21] (During some periods of the launch phase, this could result in ditching of the Orbiter at sea, with probable catastrophic results.)

Frequent occurrence of cracks in turbine blades and sheet metal parts of Space Shuttle Main Engines, requiring that engine components be replaced often.

Nonconsistent erosion performance of Solid Rocket booster nozzles—with one instance (STS–8) nearly resulting in a potentionally disastrous "burn through".

Evidence of damage to Solid Rocket Booster nozzle O-rings in 13 of the 23 missions for which the booster sets were recovered.[22]

Malfunctions in the Solid Rocket Booster recovery system (e.g., parachutes)—with one occurrence (STS–4) resulting in the loss of a flight set of Solid Rocket Boosters.

At least 48 instances of anomalous inflight behavior of the Auxiliary Power Units that drive the Orbiter's flight controls during launch and landing.[23] In one case (STS–9), two auxiliary power units failed during landing, shutdown, and then exploded several minutes after the Orbiter had come to a stop on the runway.

Anomalous behavior or total failure of the General Purpose Computers on the Orbiter.

Ejection of thermal insulation from the "intertank" region of the External Tank (i.e., the region between the liquid oxygen and liquid hydrogen tanks), causing tile damage on the Orbiter.

Failures in the Orbiter's Thermal Protection System, including the loss of tiles, the disconnection of thermal blankets, and chemical decomposition of the "screed" layer beneath many tiles on the Orbiter Challenger.

At least 63 instances of anomalous inflight behavior of the Reaction Control System that controls the flight orientation of the Shuttle while in orbit and during the initial stages of reentry.[24]

At least 78 cases of anomalous inflight behavior of the communications and tracking equipment on board the Orbiter.[25]

Clearly, some of these problems are more serious than others, and as noted earlier, some have been solved. However, the Committee is mindful of the conclusion of the Rogers Commission regarding the existence of similar situations for the Solid Rocket Motor prior to STS 51–L: "a careful analysis of the flight history of O-ring performance would have revealed the correlation of O-ring damage and low temperature." [26] Also referring to Criticality I flight hard-

---

[21] The two events mentioned in this and the preceding paragraph involving Main Engines on STS–51F were separated by 17 days, with the pad abort due to an actuator failure occurring on July 12, 1985, and the inflight abort due to a double thermocouple failure occurring on July 29, 1985.

[22] Rogers Commission Report, Volume I, pp. 129–31.

[23] NASA, "Shuttle Flight Data and Inflight Anomaly List, Revision H," Johnson Space Center, Houston, Texas, January, 1986, pp. 2–8 and 2–9.

[24] Ibid., pp. 2–6 and 2–7.

[25] Ibid., pp. 2–16 and 2–17.

[26] Rogers Commission Report, Volume I, p. 148.

ware elements, the Commission recommended that, "NASA should establish a system of analyzing and reporting performance trends for such items." [27]

In a similar vein, Dr. Feynman observed:

> The argument that the same risk was flown before without failure is often accepted as an argument for the safety of accepting it again. Because of this, obvious weaknesses are accepted again and again, sometimes without a serious attempt to remedy them, or to delay a flight because of their continued presence.

And,

> The acceptance and success of these (previous) flights is taken as evidence of safety. But erosion and blow-by are not what the design expected. They are warnings that something is wrong. The equipment is not operating as expected, and therefore there is a danger that it can operate with even wider deviations in this unexpected and not thoroughly understood way.[28]

In the spirit of these observations, it would seem clear that NASA should make sure that it fully understands all past instances of inflight anomalies and failures involving critical elements of hardware. Then, when appropriate, NASA should correct the underlying causes of these anomalies and failures.

c. Other Engineering Concerns

Issue

What action should be taken relative to other engineering concerns regarding critical elements of Space Shuttle flight hardware?

Finding

In recent years, serious engineering concerns have been raised regarding the safety of some elements of Space Shuttle flight hardware, such as the 17 inch flapper value and the heat exchanger feeding the liquid oxygen tank.

Recommendation

1. NASA should ensure that, as a part of its current review of Space Shuttle safety, it identifies, thoroughly evaluates, and then takes appropriate action on all serious engineering concerns raised regarding mission critical elements of Space Shuttle flight hardware.

2. NASA should give special attention to both the cost and risks of using Filament Wound Case Solid Rocket Boosters for very heavy Space Shuttle payloads versus the cost and programmatic impacts of simply transferring those payloads to expendable launch vehicles.

---

[27] Ibid., p. 201.
[28] Ibid., Volume II, p. F-1.

Discussion

In the months since the Challenger accident, there has been renewed interest in scrutinizing engineering concerns that have been raised in recent years regarding the safety of some elements of Space Shuttle flight hardware. Typical examples of some of these concerns involve the following pieces of equipment:

The 17 inch "flapper valves" on the fuel lines between the External Tank and the Orbiter. The inadvertent closing of one of these valves before Main Engine shutdown could be catastrophic, causing a rupture of a fuel line and/or the External Tank. Failure to close after engine shutdown, on the other hand, could cause the External Tank to crash into the Orbiter after being jettisoned.

The heat exchanger used to produce gaseous oxygen to pressurize the liquid oxygen tank in the External Tank. This heat exchanger is located inside one of the turbopump preburners of the Space Shuttle Main Engine. Should a rupture occur in the wall of the heat exchanger, high temperature hydrogen gas could be driven into the liquid oxygen tank or additional oxygen could be driven into the preburner—either situation could be catastrophic. A solution to this problem could be to move the heat exchanger outside of the Main Engine, possibly using the engine's hydrogen cooling jacket as a source of heat to produce the required gaseous oxygen.

The Filament Wound Case version of the Solid Rocket Booster now under development for use in launches involving very heavy Space Shuttle payloads. The Aerospace Safety Advisory Panel argues that this system may have questionable structural strength safety margins in the transition areas between individual case segments.[29] Safety concerns such as these have been raised regarding the Filament Wound Case Soild Rocket Boosters by the Aerospace Safety Advisory Panel for several years. In testimony before the Committee on May 15, 1986, Mr. John Brizendine, Chairman of the panel, repeated a conclusion from the panel's most recent report: "Until the issue can be resolved with a high level of confidence, . . . the Filament Wound Case Solid Rocket Boosters should not be used for STS launch. . . ."[30]

Regarding the last concern in the above listing, the Committee notes that the recent decisions to substantially delay the availability of the Space Shuttle launch facilities at Vandenberg Air Force Base and to increase the availability of expendable launch vehicles could potentially eliminate the need for Filament Wound Case Soild Rocket Boosters. Specifically, the Filament Wound Case Solid Rocket Boosters were originally intended only for use at Vandenberg; and the increased availability of large expendable launch vehicles may provide a viable option to heavy-lift launches using the Space Shuttle.

---

[29] Hearings before the Space Science and Applications Subcommittee of the House Science and Technology Committee, "Strategy for Safely Returning Space Shuttle to Flight Status," 99th Cong., 2nd Sess., Transcript, May 15, 1986, p. 102.

[30] Ibid.

The current requirement for preeminent emphasis on Space Shuttle flight safety obviously necessitates that all major engineering concerns such as those listed above should be identified, thoroughly scrutinized, and appropriately acted upon.

d. Desirable Tests Not Yet Approved

Issue 1

Is the current ground test program for the SSME adequate to provide a complete understanding of the engine's operating characteristics and safety margins?

Findings

1. The Committee supports the Findings and Conclusions of the Development and Production Team concerning the SSME, particularly the concern that "Hardware availability and the potential of damage to hardware and facilities resulting from tests malfunctions have constrained . . . [full margin] . . . testing during the ground test program." [31]

2. The Committee shares Dr. Feynman's concern that there has been a slow shift toward decreasing safety in the SSME program.

3. There is not a sufficient understanding of SSME blade cracks and fractures.

Recommendations

1. The Committee concurs with the Development and Production Team conclusion that over testing, limits testing, and malfunction testing in the SSME program should be re-emphasized to demonstrate full engine capability.[32]

2. NASA should prepare and submit to the Committee a cost-benefit analysis of testing a SSME to destruction including: (a) utilizing additional SSME test stands; (b) utilizing additional hardware for the ground test program; and (c) the value of such a test.

3. A vigorous study of fracture behavior should be conducted to minimize the hazard of cracked SSME blades and to increase the reliability and safety margin of blades. New blades and/or new policies for duration of blade use should be incorporated prior to the next Shuttle flight.

Discussion

The development and operation of the SSME is a remarkable achievement and represents the leading edge of technology in large liquid hydrogen/liquid oxygen rocket engines. Great attention to detail was emphasized by engineers at both Rocketdyne and Marshall as well as timely recognition and resolution of technical problems.[33] Despite intense oversight, individuals privately speculated, prior to the 51-L accident, that if an accident were to occur it would probably be the result of an SSME failure simply because of the uncertainties innate to a technology pushing the "edge." This awareness of the uncertainties promoted high quality engineering and contributed to the success of the SSME program unlike the ap-

[31] Rogers Commission Report, Volume II, p. K-32.
[32] Ibid.
[33] Ibid.

parent complacent attitude toward the mature solid rocket technology.

Volume II of the Rogers Commission Report analyzes in great detail the development, production and operation of the SSME.[34] The Commission's Findings and Conclusions regarding the SSME are appropriate. However, the Committee feels that even more rigorous testing of the main engine is necessary to ensure that safety margins and hardware reliability are not compromised.

For example, Commission member Dr. Feynman notes that the "top-down" [35] approach used to design the SSME has made it difficult and expensive to discover the causes of component and subsystem problems.[36] Specifically, Dr. Feynman writes that NASA and Rocketdyne (the SSME prime contractor and a division of Rockwell International), do not have a relatively precise knowledge of when a turbine blade is likely to crack, how quickly a crack will grow to fracture, and under what various rated power levels these phenomena will occur.[37]

Mr. Jeffs, President, North American Space Operations, Rockwell International, described the blade problem and explained Rocketdyne's testing efforts to improve blade life and minimize blade cracks and fractures.

> . . . we are working the blades and bearing problems and have been for some time.
>
> We have given ourselves confidence in flying the engine with those kinds of blades through off-limit testing. We've taken the worst cracked blades we could possibly find and run them in engines to see if we could make those cracks grow. We have not been able to do so. At the same time, it's not satisfactory for us to continue in the long-term flying cracked blades, and that's why we're putting so much effort on fixing those blades. I believe that we should have fixes for those blades before the next flight.
>
> . . . the blades on the engines . . . I hedge a little bit on exactly when we can incorporate those into the vehicle. I believe we can do it by the 1988 period, but it's going to take a lot of certification testing . . .[38]

During staff discussions with NASA personnel, the issue of SSME destruction testing arose. Some NASA personnel expressed the desire to test an SSME to destruction, but noted the lack of test stands and hardware.[39] Curently, NASA and Rocketdyne have three test stands. Some individuals privately noted that the SSME program should have four or five test stands to run and engine to destruction, to test product improvements, for flight support and anomaly resolution, and for acceptance tests of hardware. Others have explained that it is not necessarily the number of test stands that is the key to a successful SSME program, but rather the

---

[34] Ibid., pp. K-23 through K-27; K-31.

[35] Most military and civilian aircraft engines are designed from the bottom-up approach, in which each component, starting with the material used all the way through engineering testing of subsystems and subcomponents, is evaluated prior to the final design of the entire engine.

[36] Rogers Commission Report, Volume II, p. F-2.

[37] Ibid.

[38] Cmte Hgs, Transcript, July 15, 1986, pp. 10-12, 61.

[39] Discussion with NASA personnel, Washington, D.C., May 30, 1986.

amount of hardware available to feed the test stands. The Rogers Commission found that:

> The number of engine test firings per month had decreased over the past two years. Yet this test program has not yet demonstrated the limits of engine operation parameters or included tests over the full operating envelope to show full engine capability. In addition, tests have not yet been deliberately conducted to the point of failure to determine actual engine operating margins.[40]

In addition, Dr. Feynman said:

> Using the completed engine as a test bed to resolve such questions is extremely expensive. One does not wish to lose entire engines in order to find out where and how failure occurs. Yet, and accurate knowledge of this information is essential to acquire a confidence in the engine reliability in use. Without detailed understanding, confidence can not be attained.[41]

There has been some concern raised about the value of testing an SSME to destruction. It is important that engine testing simulate flight as closely as possible so that information learned in testing can readily be applied to actual flight engines. For example, running an engine longer than an actual flight may not be useful in understanding what effects starting and stopping have on lifetimes of engine components, such as turbine blades. However, damaging or destroying an engine while testing components under flight conditons will yield valuable information. Consequently, it is important for NASA and Rocketdyne to aggressively test components to their design life even at the expense of a ground failure. It is understandable that the 51-L accident may have resulted in a more conservative SSME ground test program in terms of a fear of failure. However, if safety is to be the prime consideration in the STS program, then there has to be the freedom to fail in order to learn. It is far better to lose an engine on the ground than in flight.

Issue 2

Is the leak/combustion threat of the External Tank's hydrogen pressure valve a hazard warranting testing?

Findings

1. The Committee supports the Rogers Commission concern regarding the hazard posed by the liquid hydrogen vent and relief valve.[42]

2. The Committee suports the intent of the ET prime contractor, Martin Marietta, to pursue outdoor wind tunnel testing to eliminate the liquid hydrogen vent/relief valve hazard.[43]

---

[40] Rogers Commission Report, Volume I, p. 192.
[41] Ibid., Volume II, p. F-2.
[42] Ibid., Volume I, pp. 192-93.
[43] Ibid., Volume II, p. K-23.

Recommendation

NASA, in conjunction with the appropriate contractor, should consider designing and conducting an ET liquid hydrogen leak/burn test to determine if corrective actions should be taken prior to the next Shuttle flight.

Discussion

The Rogers Commission identified the hazard posed by the partially open vent/relief valve on the ET's liquid hydrogen tank. This valve can indicate it is closed when, in fact, it might be partially open. A liquid hydrogen leak and subsequent combustion could result in the loss of vehicle and crew.[44] There are two ways of determining if the valve is closed. While both are highly accurate, neither can adequately assure closure. To date,

> . . . no test has been permitted to leak and burn hydrogen in a wind tunnel and analytical methods of determining the heating rates associated with leaking hydrogen gas into the 1.5-foot thick boundary layer of External Tank are recognized by the analyst to be inadequate and inconclusive.[45]

During the Commission investigation representatives of Martin Marietta stated a concern for the vent/relief valve leak hazard and indicated an intent to pursue outdoor wind tunnel testing.[46]

Issue 3

Does the present Range Safety System (RSS) on the External Tank present an unreasonable risk?

Finding

There is substantial controversy over the relative benefits and risks of the present RSS on the External Tank.

Recommendation

The Committee believes the Administrator should prepare and submit to the Committee a comprehensive review of RSS requirements.

Discussion

There has been considerable discussion through the years about the advantages and disadvantages of having a Range Safety System (RSS) radio controlled destruction device on the External Tank.

There have been recorded instances of spacecraft being struck by lightening during the launch.[47] At least some of the astronaut corps feel strongly that the ET RSS creates an unnecessary risk to the crew.

The Committee has been informed that the ET RSS was included during the design phase because of a range safety requirement.

---

[44] Ibid., Volume I, pp. 192-93.
[45] Ibid,. Volume II, p. K-23.
[46] Ibid.
[47] NASA, *The Apollo Spacecraft, A Chronology,* 1978, Volume IV, p. 319, Appollo 12.

The question that should be asked is: "Do the relative risks and advantages of an ET RSS justify its inclusion as a part of the STS?"

Therefore, the Committee believes that as part of an overall review of safety requirements, the Administrator should ensure that NASA and the appropriate Air Force officials responsible for range safety requirements review RSS requirements as they apply to the ET.

e. Production/Refurbishment Issues

Issue 1

Should 100 percent X-ray inspection of the propellant and insulation for the Solid Rocket Motors (SRM) be resumed?

Findings

1. Previous X-ray inspections led to only one SRM being rejected for Shuttle use.
2. There is no non-destructive inspection method which can guarantee a defect-free SRM. X-ray inspection cannot detect "kissing" voids in which the SRM insulation is touching the SRM steel casing but is not bonded to it. Debonded insulation at the end of an SRM segment could provide burning propellant gases with a path to the SRM steel casing and could result in loss of vehicle and crew. X-ray inspection can detect propellant cracks and large voids which if undetected could also result in a catastrophic situation.
3. Although there is no guarantee that X-ray inspection has been a particularly effective method of detecting propellant and insulation SRM flaws, it remains one of the best available methods to monitor the SRM manufacturing process.

Recommendations

1. NASA should consider reinstating full X-ray inspection of the propellant and insulation for all motors used on succeeding flights until new, more accurate inspection methods can be developed and implemented and there is unquestionable confidence in the SRM production process.
2. NASA, in conjunction with the appropriate contractors, should investigate the development of new, more accurate inspection techniques which can detect "kissing" voids and other potential defects that cannot be detected by X-ray inspection.

Discussion

While the 51-L accident has focused attention on the design of the SRM joint, the explosion of a Titan 34D rocket on April 18, 1986, has focused attention on the production and inspection processes of the SRM. Evidence has shown the Titan failure was caused by a "thermal insulation coating that pulled away from inside one of the two Solid Rocket Booster motors, allowing hot propellants to burn through the rocket's outer casing" nine seconds after being fired from Vanderberg Air Force Base.[48]

---

[48] Michael Isikoff, "Major Flaws Ruled Out in Two Rockets," *Washington Post*, July 3, 1986, p. A-1.

The Titan Solid Rocket Motors receive only visual inspection of bond lines and local ultrasonic inspection as required.

While there are some significant differences between the Shuttle SRB and the Titan motor, the design of the Shuttle SRB was primarily based on the Air Force's Titan III solid rocket.[49] Also, the design of the insulation on the Shuttle booster is virtually identical with the Titan design.[50] Brig. Gen. Nathan Lindsay, Chairman of the Air Force board investigating the accident said, "This was a failure we would have assigned a very low probability to. We've flown 70 flights with the [Titan] Solid Rocket Motor and this was the first failure." [51]

The Shuttle SRB has flown 25 flights with one unrelated failure. In testimony before the Rogers Commission, NASA officials "made it clear that the kind of separation of insulation that apparently led to the destruction of the Air Force Titan 34D was commonplace on the Shuttle." [52] According to NASA officials, it was also common practice to visually inspect and repair unbonded insulation of the SRM end segments.[53]

Full X-ray inspection was conducted on all SRM segments used in the demonstration and qualification programs and the first five Shuttle flights. Full X-ray inspection of these early motors was required as part of the development and verification plan, and was scheduled for reassessment after the flight of STS-5. During this period 24 motors were fully X-rayed. Three demonstration center motor segments exhibited excessive voids in their propellant, but only one segment was rejected. Studies established the voids were due to low casting rate and the method of dispersing propellant into the segments. As a result controls were implemented and verified by X-ray inspection. It was also discovered during this time that an SRM segment of the size required for the Shuttle could contain 12,000 voids and be fired successfully without threat to the mission, vehicle or crew.

After evaluation of data from the SRM segments used up through STS-5 and data from military Solid Rocket Motors, NASA's confidence in the SRM production process was such that the SRM X-ray policy was changed. A cost-benefit analysis also contributed to this decision. Beginning with STS-6, X-ray inspection was only conducted on all aft segments in the propellant hand-trimmed area and the segment produced following the identification of a process anomaly, process change, or design change. X-ray inspection of the aft segments in the propellant hand-trimmed area was continued because data indicated that only 3 percent of a segment's insulation had to be bonded, particularly the ends, in order of the segment to burn properly and safely. In October of 1985 NASA implemented a recommendation from the Aerospace Advisory Panel to change its X-ray policy to include random inspection of one SRM segment per month. Because of a SRM production lead

---

[49] Rogers Commission Report, Volume I, p. 121.
[50] David E. Sanger, "Flaw in Titan's Boosters is Identified," the *New York Times*, June 4, 1986, p. A-23.
[51] Isikoff, pp. A-1, A-20.
[52] Sanger, p. A-23.
[53] Ibid.

time of approximately eight months, no SRM segment inspected under this new policy was flown before the 51-L accident.[54]

X-ray inspection, while the best available method to detect propellant voids and cracks, cannot detect so-called "kissing voids" in which the insulation is touching but not bonded to the SRM steel casing. Staff discussions with NASA personnel revealed that other inspection methods are being analyzed. Thermography and accousticolography techniques could both be used to detect voids and unbonded insulation. These techniques may not be refined enough to use on the boosters flown on the next Shuttle flight. However, a mechanical pull test should be available to test the new motors to ensure that insulation is bonded to the SRM steel casings prior to pouring the propellant. In addition, NASA will reinstate its initial 100 percent full X-ray policy which applied to the earlier demonstration motors and first five flights. According to NASA officials the continued use of X-ray inspection once Shuttle flights have resumed will depend upon success of the new SRM design and development of new inspection techniques.

Issue 2

Are all production and other activities involving Criticality 1 and 1R hardware at prime and secondary contractor facilities labeled as "critical" processes?

Findings

1. Critical processes are formally identified and controlled by NASA. All processes are classified and controlled by the contractor's Process Change Control Board.[55]

2. The O-ring used in the case joint is critical to the sealing integrity of the joint, yet it is not designated as a "critical process by either the Parker Seal Co. or Hydrapack, the manufacturer and supplier respectively.[56] This raises the possibility that other Criticality 1 and 1R hardware components are also not appropriately designated by their manufacturer as "critical" processes.

Recommendations

1. NASA should require the manufacture of critical items, such as the O-rings, to be designated "critical" processes. Contractors should formally notify their employees involved in critical manufacturing processes of the serious nature of particular production processes.

2. NASA should conduct a thorough review to ensure that all manufacturing processes involving Criticality 1 and 1R hardware components of prime and secondary contractors are appropriately designated "critical" processes.

Discussion

"Critical processes are formally identified and controlled by NASA. All processes are classified and controlled by the contractor's Process Change Control Board." [57] Failure of Criticality 1 and

---

[54] Discussion with NASA personnel, Washington, D.C., Sept. 12, 1986.
[55] Rogers Commission Report, Volume II, p. K-14.
[56] Ibid., p. K-31.
[57] Ibid., p. K-14.

1R components or systems will result in loss of vehicle and/or crew. The Commission's investigation revealed that the O-ring used in the case joint, whose failure led to the destruction of 51-L, was not designated a critical process.[58] NASA personnel explained that O-ring production was not so classified because final O-ring inspection occurred at KSC.[59]

NASA's safety, reliability and quality assurance (SR&QA) philosophy is that you cannot inspect quality into products, rather you must build it in. It is questionable, then, why NASA would choose to rely solely upon O-ring inspection for quality control and not emphasize the criticality of the O-ring production process to the O-ring manufacturers and their employees.

While the O-ring manufacturing process did not contribute to the 51-L accident, the fact that such a critical item was not designated a critical process raises the possibility that other critical items may not be so designated. Identifying critical processes and educating contractors and subcontractors about critical items and manufacturing processes would be in line with NASA's policy of building in quality and safety.

Issue 3

Do O-ring repairs compromise safety?

Finding

The Committee supports the Development and Production Team Finding and conclusion that the "limit of five repair joints per O-ring is an arbitrary number" and that "repair of inclusions and voids in the rubber . . . appears to be an area of potential problem." [60]

Recommendation

NASA should review its O-ring repair policy and contractor repair practices in terms of their effects on O-ring performance and safety. Such review should be completed prior to the resumption of Shuttle flights if, as anticipated, the new SRB joint design uses O-rings.

Discussion

The NASA/Commission Development and Production Team questioned the adequacy of the SRM joint O-ring process and quality control. According to the D&P Team, "the O-ring is allowed to include five scarf joints, a quantity which is arbitrarily established, and repairs of inclusions and voids are routinely made by the vendor after receipt of the material supplies".[61]

Issue 4

What impact does growth of SRM case size have upon booster and Shuttle performance and safety?

---

[58] Ibid., p. K-31.
[59] Staff meeting with NASA personnel, Headquarters, Washington, D.C., May 13, 1986.
[60] Rogers Commission Report, Volume II, p. K-31.
[61] Ibid., p. K-3.

Finding

The Committee concurs with the Development and Production Team Finding that "Remeasurement of two used SRM case segments indicated both tang and clevis sealing surfaces have increased in diameter beyond the anticipated design limits." [62]

Recommendation

NASA and the appropriate contractor should resolve through analysis and testing prior to the next Shuttle flight the cause of SRM case size growth and its impact upon booster and Shuttle performance, reliability of refurbished SRM case segments, and safety.

Discussion

During the investigation of the 51-L accident, the NASA/Commission D&P Team "determined by measurement of two flown case segments that both the SRM tang and clevis sealing surfaces have increased in diameter beyond the anticipated design limits. The growth is believed to be material related and related to the hydrostatic proof test pressure level." [63]

f. Review of NASA's Redesign/Recertification Plan

Issue

Is NASA's SRM redesign and hardware recertification plan a viable and realistic one which will result in a safer, more reliable Space Transportation System?

Findings

1. NASA's SRM redesign plan is a step in the right direction. Moving the proposed launch date beyond June 1987 is a responsible and realistic decision. The membership of the SRM Redesign Team is representative of qualified individuals in and outside of NASA. With the expert assistance of the specially appointed National Research Council (NRC) Independent Oversight Group, the new SRM design should be a significantly safer and more reliable Shuttle element.

2. NASA's current hardware recertification plan is also a step in the right direction. The use of independent review contractors distinguishes this recertification plan from earlier reviews. However, given the failure of previous reviews to discover the deficient SRB joint certification, the Committee is concerned there is still the possibility that the recertification effort may not reveal other certification deficiencies, if indeed they exist. The plan also raises concern about the qualifications of independent reviewers to evaluate certain elements given the uniqueness of particular Shuttle components.

3. The joint was never fully tested as a separate element of the SRM. The various forces that act on the joint during stacking, launch, and flight are difficult, if not impossible, to duplicate in a test of the joint under all conditions that could be experienced during launch and flight.

---

[62] Ibid., p. K-31.
[63] Ibid., p. K-3.

4. It is unclear what function the new Safety Office will perform in the redesign of the SRB field joint and other critical elements of the Shuttle, as well as NASA's recertification plan.

Recommendations

1. The Committee recognizes the national need to return the Shuttle to flight status as soon as reasonably possible. As noted in NASA's July 14, 1986, report to the President, safety will determine the launch schedule. However, NASA should consider the proposed launch date of early 1988 as a flexible one which should be slipped further if necessary. The Shuttle should not be launched again until NASA can assure that safety criteria have been met.

2. In establishing a test program to certify the new Solid Rocket Motor design, NASA should consider the feasibility of including in combination and in the proper sequence all of the thermal and structural loads expected to be experienced by the Solid Rocket Motor during ignition, lift-off, and flight.

3. The independent review contractors participating in the hardware recertification plan should utilize sufficient specific technical expertise to insure adequate recertification of all elements of the STS.

4. The Committee requests that the new Office of Safety, Reliability and Quality Assurance conduct an independent assessment of the SRB field joint redesign efforts. In addition, the new office should also be integrally involved in reviewing all other critical component redesign efforts and NASA's recertification plan.

Discussion

The "Strategy for Safely Returning the Space Shuttle to Flight Status" includes the plans to redesign the SRM joint, reverify hardware design requirements, and to completely review all "critical" items. This strategy was proposed March 24, 1986, by Admiral Truly, the Associate Administrator for Space Flight and supplements the Rogers Commission's recommendations.[64]

The redesign of the SRM joint is being conducted and supervised by a cross section of competent and qualified individuals from NASA centers, including Marshall Space Flight Center, the Astronaut Office, and individuals from outside NASA. An expert advisory panel of 12 people, six from outside NASA, has also been appointed. Further, at the request of the NASA Administrator, the National Research Council has established an Independent Oversight Group which reports directly to the Administrator. (See Appendix VI-B.)

To date, "many design alternatives have been evaluated, analyses and tests have been conducted, initial verification plans have been established, and overall schedules have been developed."[65] In addition to designing a new joint that will use existing hardware, an alternate design that does not use current hardware is also underway. Study contracts have been let to five companies to independently develop new designs and review current baseline ideas and tests already conducted by NASA.

---

[64] NASA Response to Rogers Commission, July 14, 1986, pp. 37-40.
[65] Ibid., p. 12.

In 1985, as joint problems continued with the Solid Rocket Motors, NASA recognized the need for a design that would limit the rotational movement between the joint tang and clevis that occurs at motor ignition. NASA was embarking on the design of a Filament Wound Casing which would have the advantage of allowing for an increase in payload. Since a new design was called for, it was to NASA's advantage to correct some of the joint problems at this time. In the new design, to reduce the rotational movement in the joint, a hook was added to the inner leg of the clevis. This would significantly limit any change in the spacing gap between the tang and the clevis in the area where the O-rings are installed. Shortly thereafter, Thiokol, on its own initiative, ordered new forgings which were thicker in the tang area so that the capture feature could be machined into the casing. The hook has become know as the "capture feature". In August of 1985, however, the capture feature then under consideration was significantly different than current joint designs with the capture feature. Some of these differences include the presence of a third O-ring, the addition of a second pressure test port, the adoption of an interlocking design for the case insulation in the vicinity of the joint, and the removal of all putty from the joint.

The new design, however, with the capture feature, appears to complicate the stacking operations and could increase the potential for damage, particularly leading to the creating of metal silvers during mating operations.

The hardware recertification plan appears to be a thorough approach to verifying that components meet design requirements. The recertification plan involves three levels of review to be conducted by: (1) NASA personnel; (2) current Shuttle contractors; and (3) independent contractors.

A major concern about the thoroughness of the hardware recertification plan was expressed by Mr. Nelson:

> . . . looking back on how the whole system functioned we found out that there were a whole bunch of people involved in the SRB design and the certification process. There were the internal groups at Thiokol; there was the oversight by Marshall; there was a through review by an outside group, headed by Dr. Williams; there was the Aerospace Safety Advisory Panel; there was the certification process and signing off, [not only for the test flights] but for the first Shuttle flight, STS-1; and then there was that same certification process and signing off again that occurred before STS-5.
>
> Now, still all of those problems went undetected by so many groups.[66]

A more detailed discussion of previous hardware certification is in Section VI-A.1.a. of this report.

Mr. Nelson further asked how this plan could provide confidence that it is relatively safe to fly again.

---

[66] Cmte Hgs, Transcript, July 15, 1986, p. 95.

The use of independent contractors distinguishes this recertification plan from previous ones. Mr. Davis, President, Martin Marietta Michoud Aerospace, said:

> . . . some of the things that are being done differently . . . I think will help . . . Marshall Space Flight Center has contracted with other companies for independent FMEA/CIL assessments of their hardware. In particular, Rockwell is doing a total independent assessment of my External Tank hardware, and I think that's well looked-to. I look to their expertise to question everything we did and maybe give us some advicè on how to make it better.[67]

Mr. Murphy, Executive Vice President and General Manager, United Technology Booster Production Company, commented:

> I think one of the things that is going to prevent a recurrence of what happened in the past, as far as oversight committees are concerned, is that we have come a long ways since the initial certification of the program. We now have advanced analytical tools which were not available before. We also have the flight environments for the 24 sucessful flights; it gives us a true indication of what the environment is that we're going to be facing. Plus, again, the environment of the whole aerospace industry has changed dramatically since 51-L. And all of these, I think, will be taken into consideration and will provide the oversight and the proper review of items that never occurred before.[68]

Mr. Murphy explained how his company will recertify the new SRB design:

> The recertification has three primary elements which follow a logical progression of evaluation. First, we will reestablish the basic design requirements from Level II and Level III. Second, we will establish a verification program based upon those requirements. And third, we will reestablish that the design and the hardware are in compliance with the first two elements.
>
> Key activities to be performed as we recertify the SRB will include the traceability of all the requirements into all levels of SRB design and system environments, verification of the SRB design data base and analyses, establishment of tools such as the failure modes and effects analysis, and validation that our paper systems have properly incorporated requirements, constraints, and criteria.[69]

Another distinction of this plan is that hardware will be completely recertified through actual testing and analysis as if it were being done for the first time. Some earlier certification reviews were abbreviated paper checks many of which focused on only certain components.

---

[67] Ibid., p. 96.
[68] Ibid., p. 98.
[69] Ibid., p. 50.

> Mr. ROE. . . . we're talking, where at all possible, actual field testing. Do you concur with that approach? . . .
>
> Mr. DAVIS. Yes, I'd say I agree with that. As a matter of fact, I believe that's what all the program contractors, are out doing at this point.[70]

The use of independent review contractors is a necessary and critical component of the recertification plan. However, a legitimate concern has been raised by the current contractors and Committee staff regarding the ability of independent contractors to review technologies and components for which they may have limited expertise. For example, solid and liquid rocket propulsion has often been referred to as a "black art" for which there are few experts. NASA has contracted with Martin Marietta to independently review the SRB and SRM certification. Understanding the uniqueness of rocket propulsion, Martin Marietta has supplemented their in-house talent with outside experts to assist in the certification review.

The complete review of the Critical Items List (CIL), Hazard Analyses (HA), and Failure Modes and Effects Analyses (FMEA) is in response to the Rogers Commission's third recommendation and is intended to identify those items that must be improved prior to flight, and to affirm the completeness and accuracy of each FMEA/CIL for the current NSTS design. This is the first such review since the system was originally instituted at the beginning of the NSTS program and involves, according to NASA, many man-hours of effort and a very large staff. Supporting this effort are independent contractor reviews of the various FMEA/CIL activities associated with each major component and system of the National Space Transportation System. There are six such activities. These include, in addition to the four major subsystems that comprise the Shuttle (Orbiter, Solid Rocket Booster, Space Shuttle Main Engine, and the External Tank), the Vandenberg Launch Site and the Kennedy Space Center operations. The re-evaluation is scheduled to be completed by March 1987. It will involve all levels of NASA management, with auditing and oversight functions to be provided by outside personnel from the Aerospace Safety Advisory Panel and the National Research Council in accordance with the recommendations of the Rogers Commission.

A reconsideration of the level of design center involvement (i.e., the field centers that are responsible for designing various components of the Shuttle) in equipment processing or systems processing is required. The establishment of an Office of Safety, Reliability and Quality Assurance under a separate Associate Administrator should lead to improvements or an increase in the audit activities associated with the overall development and production process activities within the program. A systems design review that is presently underway within NASA has led to some 70 or 80 items over and above the CIL review that have been brought to the attention of Level II as potential problem areas.[71]

---

[70] Ibid., pp. 59-60.

[71] Discussion with NASA officials, Washington, D.C., July 10, 1986.

Program management at Level II has requested a complete audit of the problem reporting system in order to assure that only priority issues are elevated to the Level II status for review. NASA suggests the problem has been that too many items of lower categorization than Criticality 1 or 1R have been brought up to Level II and have swamped the ability of this management level to adequately analyze Criticality 1 items.

The new office of Safety Reliability and Quality Assurance is now operational. It is the view of the new Associate Administrator that the role of the SR&QA office will be to assure that modifications to the SRB field joint design, are extensively reviewed during the processes of development, fabrication and testing.[72] It is the plan of this office to establish a position at headquarters to review the configuration management system that presently operates across the National Space Transportation System. It is also a goal of the Associate Administrator to establish and improve lines of communication among the various NSTS elements in order to improve component integration and information interfacing among the various elements of the Shuttle.

The Committee is fully aware that faulty designs, improper fabrication techniques and component certification efforts can only be detected and identified through the implementation of proper quality control methods and procedures. The task of the NSTS program managers and the contractors is to assure that the quality is built into the design and production of Shuttle hardward. Nevertheless, the Committee also recognizes that the highest level of quality control methods and reliability engineering must be applied to all phases of the Shuttle production process, utilizing the latest state-of-the-art techniques of testing and analyses.

2. OPERATIONS

a. Shuttle Processing Issues (including Spare Parts)

Issue

In 1983, NASA consolidated fifteen separate contracts and awarded a single Shuttle Processing Contract (SPC) encompassing all ground processing related to launch and landing of the Space Shuttle. There are two issues associated with this contract: (1) How sound is the concept of a unified SPC; and (2) How well has the SPC contractor actually performed? A related issue is the quality of essential logistical support, especially spare parts, provided to the contractor by NASA.

Findings

1. Performance under the SPC has improved since the inception of the contract. However, up to the time of the Challenger accident, contractor performance continued to be plagued by excessive overtime, persistent failures to follow prescribed work procedures, and inadequate logistical support from NASA.

2. High overtime rates have hampered SPC performance. Overtime rates had increased significantly during the six months prior

[72] Discussion with NASA officials, Washington, D.C., August 13, 1986.

to the Challenger launch, to the point that critical personnel were working weeks of consecutive workdays *and* multiple strings of 11- and 12-hour days. Fatigue resulting from work patterns of this sort can constitute a threat to safety. In fact, worker fatigue was a contributing factor in a mission-threatening incident on Flight 61-C, the mission immediately prior to the January 28 Challenger launch.

3. There are numerous documented cases when contractor employees failed to comply with guidelines for carrying out assigned duties, including specific "Operations and Maintenance Instructions" (OMI's). Such failures contributed to both of the major mishaps in 1985 involving Shuttle processing—namely, the November 8, 1985, "handling ring" episode which led to significant damage to a Solid Rocket Motor segment slated for use on STS51-L, and the March 8, 1985, "payload bay access platform" episode which led to significant damage to a payload bay door. Failure to follow an OMI also led to improper (and mission-threatening) handling of the hydrogen disconnect valve during the 51-L launch operations. All of these incidents show a lack of discipline, both with respect to following prescribed procedures and with respect to reporting violations of these procedures.

4. At the time of the Challenger accident, the lack of spare parts caused a degree of cannibalization (i.e., the removal of a part from one Orbiter to satisfy a need for a spare part on another Orbiter), which was the highest in the history of the Shuttle program and which was a threat to flight schedule and flight safety. Excessive cannibalization leads to multiple installations, retesting, added documentation, delayed access to parts, and increased damage potential. As a result, cannibalization contributes directly to excessive overtime.

5. There is no clear evidence whether or not greater involvement of the development contractors would improve Shuttle operations.

Recommendations

1. Because of the serious quality and safety concerns surrounding the contract, NASA should conduct a careful review of Shuttle processing, the SPC contract, and the relationship of flight hardware contractors and report its findings, recommendations, and proposed contract modifications to the Committee. NASA's reexamination should include a comparison of efficiency and safety under the SPC versus efficiency and safety during pre-1983 Shuttle processing operations, which heavily involved the development contractors.

2. NASA should examine the issues of spares availability and cannibalization and provide the Congress with a management and budgetary plan for correcting previous logistical problems.

3. NASA should stop routine cannibalization and develop guidelines (including appropriate control and review procedures and roles for the SR&QA office) governing permissible cannibalization.

4. The Committee recommends that NASA provide its re-invigorated safety office with the authority to enforce scheduling that leads to safe overtime rates.

Discussion

In a press release dated September 5, 1986, NASA announced that it has extended the SPC with Lockheed for three additional years, beginning October 1, 1986. Admiral Truly also announced his intent to conduct a thorough review of the SPC, a process which might lead to contract amendments.

Lockheed's award fees at the Kennedy Space Center have not been at the highest possible levels due to mishaps and management problems. The contractor has received the following award fees for Shuttle processing at KSC:

LOCKHEED SHUTTLE PROCESSING CONTRACT—AWARD FEE HISTORY

Period	From	Award fee available	Rating adjective	Rating score	Percent of maximum award fee earned	Award fee earned
First	Oct. 1, 1983	$6,618,880	Excellent	90.0	80	$5,295,104
Second	Apr. 1, 1984	1,299,404	Good	78.5	32	415,809
Third	Oct. 1, 1984	1,308,554	Good	76.0	24	314,053
Fourth	Apr. 1, 1985	1,308,554	Excellent	91.0	84	1,099,185
Fifth	Oct. 1, 1985	1,308,554	Very good	89.0	76	994,501
Sixth	Apr. 1, 1986	1,296,664	(1)	(1)	(1)	(1)

[1] To be determined.

The rating scale runs from unacceptable to marginal, good, very good, excellent and superior. Two of the five ratings to date have been at the lower end of the scale.

At the time of the Challenger accident, Shuttle processing had suffered from inadequate spare parts for well over a year, and the problem was getting worse. The inventory of spare parts had run close to projections until the second quarter of fiscal year 1985. At that time, inventory requirements for spares began to increase faster than deliveries. A year later, the inventory should have been complete, but only 65 percent had been delivered.[73]

The number of cannibalized parts was increasing at an alarming rate. Forty-five out of almost 300 required parts were cannibalized for Challenger before Mission 51-L.[74] Eighty-five parts were cannibalized on 61-C, the mission preceding 51-L.[75] In fact, the number of cannibalized parts on each of these 1986 missions far exceeded the number of cannibalized parts on any previous mission. In 14 missions flown in 1984-1985, the average number of cannibalized parts was 14; in the 1986 mission, the number had increased nearly five-fold to 65.

The cause of the spare parts crisis was budgetary decisionmaking by NASA management. In October, 1985, the logistics funding requirements for the Orbiter program, as determined by Level III management at Johnson, were $285.3 million, but that funding was reduced by $83.3 million, necessitating major deferral of purchases of spares.[76] By the spring of 1986, the Shuttle logistics program

[73] Rogers Commission Report, Volume II, p. I-16.
[74] Ibid.
[75] Cmte Hgs, Transcript, July 16, 1986, p. 42.
[76] Rogers Commission Report, Volume I, p. 173.

was about one year behind; and under the proposed flight schedule, no Orbiters would have been available as spare parts bins.

NASA is well aware of the spare parts problem. In fact, during the Committee's hearings, Admiral Truly testified:

> I can assure you that during our downtime we're going to take a hard look at it and make sure that the flight rates that we build up to after this accident are supportable by the logistics system that we have in place.[77]

The Committee received mixed reactions on whether development contractors need to be more involved in the SPC. Proponents of this approach argue that the current separation of responsibilities between the design organizations and the processing organization has created additional interfaces which make coordination, communication, and responsiveness more complex. Further, the processing contractor may not possess the necessary technical background to recognize either system degradation resulting from multiple missions or the criticality of the hardware being tested and processed.[78] The Rogers Commission report stated that the likelihood of improper Shuttle processing would probably be decreased if Rockwell, as overall development contractor, and Martin Marietta, who has a consulting role on the pre-launch processing of the External Tank, were subcontractors to Lockheed, as the other Shuttle development contractors are.[79]

At Committee hearings, contractors reacted predictably to proposed changes in the SPC. Development contractors, such as Rockwell and Martin Marietta, told the Committee that their organizations should be vested with beginning-to-end responsibility—design, development, manufacturing, operation, and refurbishment of their respective Shuttle element.[80] Lockheed, on the other hand, argued that there is already a very close relationship between itself and the development contractors. For example, there is one development engineer for every four Lockheed engineers. Development contractors participate in all meetings and are required to authorize and approve anything that is anomalous to the regular documented procedure.[81]

Ultimately, SPC performance will determine the proper balance of development contractors in the processing contract. NASA, in close consultation with the Congress, will need to make an impartial and ongoing assessment of comparative safety and performance under a consolidated versus unconsolidated SPC. Preliminary figures from NASA seem to indicate that Shuttle processing incidents have actually declined during the more recent consolidated-contract phase.[82] Further, it is likely that the fundamental problem to date with the SPC—overtime—would be exacerbated by the additional contractor coordination that would be required by greater inclusion of development contractors.

---

[77] Cmte Hgs, Transcript, July 11, 1986, p. 39.
[78] Rogers Commission Report, Volume II, p. K-32.
[79] Rogers Commission Report, Volume I, p. 195.
[80] Cmte Hgs, Transcript, July 15, 1986, p. 62.
[81] Cmte Hgs, Transcript, July 16, 1986, pp. 13-14.
[82] NASA, documents on the SPC contract, supplied to the Committee in July, 1986; Cmte Hgs, Transcript, July 16, 1986, p. 67, and Attachment C.

The responsibility for high overtime rates in the SPC must be shared by both NASA and the contractor. Mr. E.D. Sargent, President of Lockheed Space Operations Company, testified:

> One of the problems that bothers us and drives us to overtime is either unplanned work or another form of unplanned work which is a hold or abort on the pad where we have critical skills that are required to perform functions.[83]

There is no doubt that late mission changes initiated by NASA are in large part responsible for Lockheed exceeding the five percent overtime target in the SPC contract.

But in fact overtime levels had grown from an initial SPC rate of 5.3 percent in April, 1984, to 13.9 percent in January, 1986—levels far in excess of what could be attributed solely to late mission changes. The peak monthly overtime level of 15.2 percent occurred in November, 1985. Although NASA managers at Kennedy attribute the November rate to the Thanksgiving holiday, the overall trend in overtime is undeniable—for each of the six months prior to the launch of STS 51-L, overtime exceeded 10 percent.[84]

More important that the average overtime rates was the overtime for certain employees with critical skills. Records show that there was a frequent pattern at Kennedy of combining weeks of consecutive workdays with multiple strings of 11- or 12-hour days. For example, one Lockheed mechanical technician team leader worked 60, 96.5, 94, and 80.8 hours per week in succession during the four weeks ending January 31, 1986.[85] While shiftwork is commonplace in many industrial settings, few can equal a Shuttle launch's potential for inducing pressure to work beyond reasonable overtime limits.

Research has shown that when overtime becomes excessive, worker efficiency decreases and the potential for human error rises. Noteworthy in this regard is Lockheed's review of 264 incidents that caused property damage in 1984 and 1985. More than 50 percent of these incidents were attributable to human error, including procedural deviations, miscommunications and safety violations.[86] On one occasion a potentially catastrophic error occurred just minutes before a scrubbed launch of Shuttle flight 61-C on January 6, 1986, when 18,000 pounds of liquid oxygen were inadvertently drained from the Shuttle's External Tank. The investigation which followed cited operator fatigue as one of the major factors contributing to this incident. The operators had been on duty at the console for eleven hours during the third day of working 12-hour night shifts. If the launch had not been held 31 seconds before lift off, the mission might not have achieved orbit.[87]

The adequacy of and adherence to Operations and Maintenance Instructions (OMI's) have been raised as areas of concern leading to quality and safety problems. Review of various SPC mishap reports and of the procedures leading to the launch of 51-L and earlier

---

[83] Cmte Hgs, Transcript, July 16, 1986, p. 24.
[84] NASA, documents on the SPC contract, supplied to the Committee in July, 1986.
[85] Rogers Commission Report, Volume II, pp. G-2-3.
[86] Ibid., p. G-1.
[87] Ibid.

Shuttle flights highlight both the need for review and update of inadequate OMI's and the need for improved contractor performance in implementing adequate OMI's.[88] NASA's own review of flight 51-L showed several examples of improperly implemented procedures. The most serious error occurred when a console operator improperly closed the liquid hydrogen disconnect valve to the External Tank liquid hydrogen manifold. Although the valve appeared to function during 51-L, improper valve operation could have doomed 51-L just as surely as the failed rocket booster. As important as the failure to follow the OMI was the fact that the valve closure problem was never documented. Without proper documentation a full assessment of the problem was not made prior to launch of 51-L.[89] This lack of documentation is reminiscent of what occurred during "de-stacking" of Solid Rocket Motor segments from STS-9. Although destacking revealed water in the joints, this incident was never documented—an oversight which ultimately may have prevented an appreciation of the dangers of ice formation in booster joints during a cold-weather launch.[90]

b. Pressures on Shuttle Operations

Issue

Was NASA under pressure to fly more flights? How did this pressure originate? Will it recur?

Findings

1. The Congress and the Executive Branch jointly developed the policy that the Space Shuttle should, in a reliable fashion and at an internationally competitive cost, provide for most of the Free World's space launch needs. By and large, both Branches failed to appreciate the impact that this policy was having on the operational safety of the system.

2. NASA was under internal and external pressure to build its Shuttle flight rate to 24 per year, primarily to reduce costs per flight, but also to demonstrate and achieve routine access to space. NASA has never achieved its planned flight rate.

Recommendations

1. NASA must not attempt to achieve a flight rate beyond that which (i) can be supported by the budget and staff resources available; and (ii) is consistent with the technical maturity of the Shuttle and the flexibility desired and needed in scheduling payloads. Management should ensure efficient use of resources but should not impose a flight rate on the system.

2. Once operation of the Space Shuttle resumes, the Committee should maintain a close and continuous oversight of Shuttle flight rate, planning, and operations. The Committee should ensure both that flight rate flows logically from the resources provided and that flight safety is not compromised beyond acceptable limits.

---

[88] Kennedy Space Center Mishap Reports, No. 85-0070, April 5, 1985, and No. 86-0024, Dec. 13, 1985.

[89] Rogers Commission Report, Volume II, p. I-15.

[90] Ibid. p. I-14.

Discussion

Flight Rate. The goal of the Shuttle program has been to become the Nation's primary space transportation system launching virtually all U.S. payloads and many foreign payloads, all at a reasonable price. Thus, there has been an explicit promise to deliver launch services.

Being a very complicated vehicle, the Shuttle demands a large trained workforce which must be retained between launches. In addition, there are the costs of maintaining large and complex launch facilities. Therefore, there is a large fixed cost in the Shuttle program of approximately $1.2 billion per year. By comparison the additive or marginal cost for a single flight is around $60 million (depending on how the accounting is done). Therefore, it is clear that (within limits) the cost-per-flight can be reduced by flying more flights, that is, by spreading the large fixed cost over more flights. However, it is also clear that the total cost—that is, the total amount of money that has to be appropriated—will increase as the number of flights increases because fixed costs are fixed and marginal costs must be added for each additional flight.

Therefore, to focus on cost-per-flight can be misleading. A lower cost-per-flight, achieved by flying more often, would allow a lower *price* to be charged to users, but does not lower the cost of the program. Because NASA had committed to lower the price to customers of Shuttle flights, there was a pressure to do this by increasing the flight rate. Nevertheless, NASA never achieved its planned flight rate.

For example, in 1976 NASA predicted 49 flights in fiscal 1984 and 58 in 1985. As late as August 15, 1983, 45 days before the start of fiscal year 1984, NASA planned 9 flights for fiscal 1984 and 12 for 1985. NASA actually flew four Shuttle flights in fiscal 1984 and 8 in 1985.[91] Of course management worked hard to reduce this gap between plans and performance.

The emphasis on reducing costs per flight and delivering launch services has caused a very basic and pervasive pressure to increase the flight rate in the Shuttle program. This is well documented in Chapter VIII of the Rogers Commission report.[92]

Presumably, the Challenger accident has changed this situation. Recommendation VIII of the Rogers Commission states in part that "NASA must establish a flight rate that is consistent with its resources."[93] NASA's response to this recommendation hints that this may not be the case. NASA speaks of determining "the maximum achievable safe flight rate."[94] Such a flight rate would again leave no "margin in the system to accommodate unforeseen hardware problems" as the Commission found was the case before the accident.[95] The NASA response makes it clear that the flight rate

---

[91] Hearings before the House Committee on Science and Technology, FY 1978 NASA Authorization, September 14, 1976, Volume I, Part 1, p. 394; NASA, "Space Shuttle Payload Flight Assignments," August 15, 1983. NASA; "Space Shuttle Payload Flight Assignments," November, 1985.

[92] Rogers Commission Report, Volume I, pp. 164-77.

[93] Ibid., p. 201.

[94] NASA Response to Rogers Commission, July 14, 1986, pp. 30-31.

[95] Rogers Commission Report, Volume I, p. 177.

will be determine based on studies and that "program enhancements . . . *required to achieve the flight rate*" will be implemented [emphasis added].[96] This is reinforced in the NASA response to Recommendation IX where NASA says "NASA has initiated an assessment of spare parts requirements *to adequately support the flight rate planning.*" [emphasis added] [97] Thus, it seems that once again a planned flight rate could become a controlling factor.

A finding of the Pre-Launch Activities Team is that during the preparation of 51-L for launch "Manpower limitations due to high workload created scheduling difficulties and contributed to operational problems."[98] This is perhaps one of the clearest examples of the inappropriate logic at work in the system before the accident, because "manpower limitations" are *not* due to "high workload" in the system. Manpower and other resources are limited before the workload is planned. Problems are created when the workload assigned is inappropriate to the manpower available.

In a March 24, 1986, memorandum on "Strategy for Safely Returning the Space Shuttle to Flight Status," Admiral Truly reveals a better attitude toward flight rate in speaking of a "realistic and . . . achievable launch rate that will be safely sustainable." Admiral Truly also states that "the ultimate safe sustainable flight rate and the build up to that rate will be developed utilizing a 'bottoms up' approach in which all required work for the standard flow . . . is identified and that work is optimized in relation to the available work force."[99]

NASA prepared several reports for the Rogers Commission, and the Mission Planning and Operations Team (MPOT) Report indicates a good awareness of the general problem of over-ambitious flight rate planning. For example, that report states that compared to the need to devote resources to making the transition to an operational system the "increasing flight rate had the highest priority." The MPOT report continues, "In other words, it appears that the flight rate was not tied to the ability of the system to support it, but rather the system was reacting to the established flight rate." A major conclusion of the MPOT report is that "The NSTS Program should develop a bottoms-up strategy for expanding flight rate."[100] In other words, flight rate cannot be imposed from above, but must be determined by available resources.

The disturbing fact is the trend in the NASA statements. The earlier statements (i.e., the Truly memo and the MPOT report) indicated an awareness of the danger of trying to achieve an imposed flight rate. However, as mentioned above, the most recent statement, the NASA response to the Commission, once again speaks of achieving the planned flight rate.

The Rogers Commission has documented the fact that before the Challenger accident the Shuttle system was approaching a state of saturation in which no more flights could be accommodated. If the accident had not occurred flight rate saturation may have eventually been reached due to bottlenecks in crew training on the mis-

[96] NASA Response to Rogers Commission, July 14, 1986, p. 31.
[97] Ibid., p. 33.
[98] Rogers Commission Report, Volume II, p. I-14.
[99] NASA Response to Rogers commission, July 14, 1986, p. 40.
[100] Rogers Commission Report, Volume II, p. J-31.

64-420 O - 86 - 5

sion simulators [101] or because of inadequate spare parts for the Orbiter.[102]

Availability of training time on simulators and availability of spare parts can both be improved by the application of more resources. Nevertheless, if the achievement of a planned flight rate is the overriding concern, removal of one bottleneck may only reveal another one. Eventually, pressures will be brought to bear on safety. The pressure on NASA to increase Shuttle flight rate has been complicated by the need to maintain program flexibility (which means to accommodate changes in the payloads on the manifest) and by the "developmental" nature of the Shuttle system. Manifest changes and the developmental nature of the system create problems in the planning of Shuttle missions.

In addition, it is interesting to note that until the training, spares, and mission planning problems are resolved, achievable flight rate may not depend on whether or not Challenger is replaced.

c. Impact of Pressures on Shuttle Operations

Issue

Did operating pressures adversely affect the safety of the Shuttle program?

Findings

1. The pressure on NASA to achieve planned flight rates was so pervasive that it undoubtedly adversely affected attitudes regarding safety.

2. The pressure to achieve planned flight rates was compressing mission preparation as earlier missions were delayed due to unforeseen problems. Had the accident not occurred there would soon have been a collision between planned launch dates and mission preparation needs which could not have been met by overtime, cannibalization, or other undesirable practices. Operating pressures were causing an increase in unsafe practices.

3. The schedule of payloads planned to fly on the Shuttle (the manifest) was frequently changed. Each change rippled through the NASA Shuttle organization and through the manifest and, especially if made shortly before launch, would increase the demands on personnel and resources in order to achieve the planned flight rate.

4. The Space Shuttle has not yet reached a level of maturity which could be called operational as that term is used in either the airline industry or the military. Each Shuttle flight is fundamentally unique, and requires unique preparations. Therefore, small changes in a mission can cause significant perturbations of mission planning and crew training.

Recommendations

1. The new Associate Administrator for Safety, Reliability and Quality Assurance must assure that any pressures to increase the

[101] Ibid., Volume I, p. 170.
[102] Ibid., p. 174.

Shuttle flight rate do not adversely influence mission preparation. The Associate Administrator must have the authority not only to stop a particular flight, e.g., at a Flight Readiness Review, but to stop the whole mission planning process if necessary.

2. Where appropriate, NASA should take steps to make the mission planning process standard and routine to reduce the time and resources needed to plan a mission. Before requesting more resources for the existing mission planning process (manpower, facilities, equipment) NASA should identify ways to improve the process.

Discussion

There is no doubt that operating pressures created an atmosphere which allowed the accident on 51-L to happen. Without operating pressures the program might have been stopped months before the accident to redesign or at least understand the SRB joint. Without operating pressure the flight could have been stopped the night of January 27. This is documented in the Rogers Commission report in Chapters V and VI.[103] Specific manifestations of launch pressure and the resultant atmosphere in the agency are described in detail in Section VIII of this report.

Nevertheless, it has become clear that the Shuttle launch system was not functioning well and was becoming increasingly unsafe as flight rate was increased. This is documented in Chapter VIII of the Rogers Commission report.[104]

Mission Planning.—Mission planning refers to the process of defining and preparing each Space Shuttle mission. It is important to understand the mission planning process in order to understand why pressure to achieve a given flight rate could have adverse impacts. The process is lengthy, complex, and tightly interrelated. That is, many steps must be done in sequence, and many different flights have to use limited resources and facilities.

Mission planning begins at NASA headquarters with the customer services manager in the Office of Space Flight. Both financial and policy agreements between NASA and the customer are negotiated and signed. Technical documentation begins at this time although the level of mission-specific work is low.

After flight assignments are made by NASA Headquarters and the mission is defined, a process of continual review begins. Payloads are assigned to a particular flight 33 months prior to launch. At this time a Payload Integration Plan (PIP) is developed which includes a preliminary analysis of the mission.

Payload safety is the responsibility of the payload developer. He must be throughly familiar with NASA safety requirements and must certify that his payload meets them. NASA audits the certification process but performs no visual inspection of the payload for conformance to safety standards.

Once the cargo of a particular mission has been defined, or "baselined", the significant engineering work of mission processing actually begins. NASA refers to this as the "production process". The product of the process is the launch of a particular mission,

---

[103] Ibid., pp. 82-151.
[104] Ibid., pp. 164-77.

but there are many other intermediate products such as flight and training software, crew activity plans, and handbooks and checklists for the crew to take on the flight.

The launch production process template is displayed schematically on Figure VI-2. The template begins 15 months before the scheduled launch date (L-15), at which time a Flight Definition and Requirements Directive (FDRD) is issued. This marks one of seven defined "freeze points" of the 15 month mission-specific pre-launch activity. A freeze point simply means that a particular activity is norminally defined so that no time changes can occur without a formal process to authorize and document the change. In theory, non-mandatory changes are not made after a freeze point. As noted below, significant changes do indeed occur after the various freeze points in the schedule.

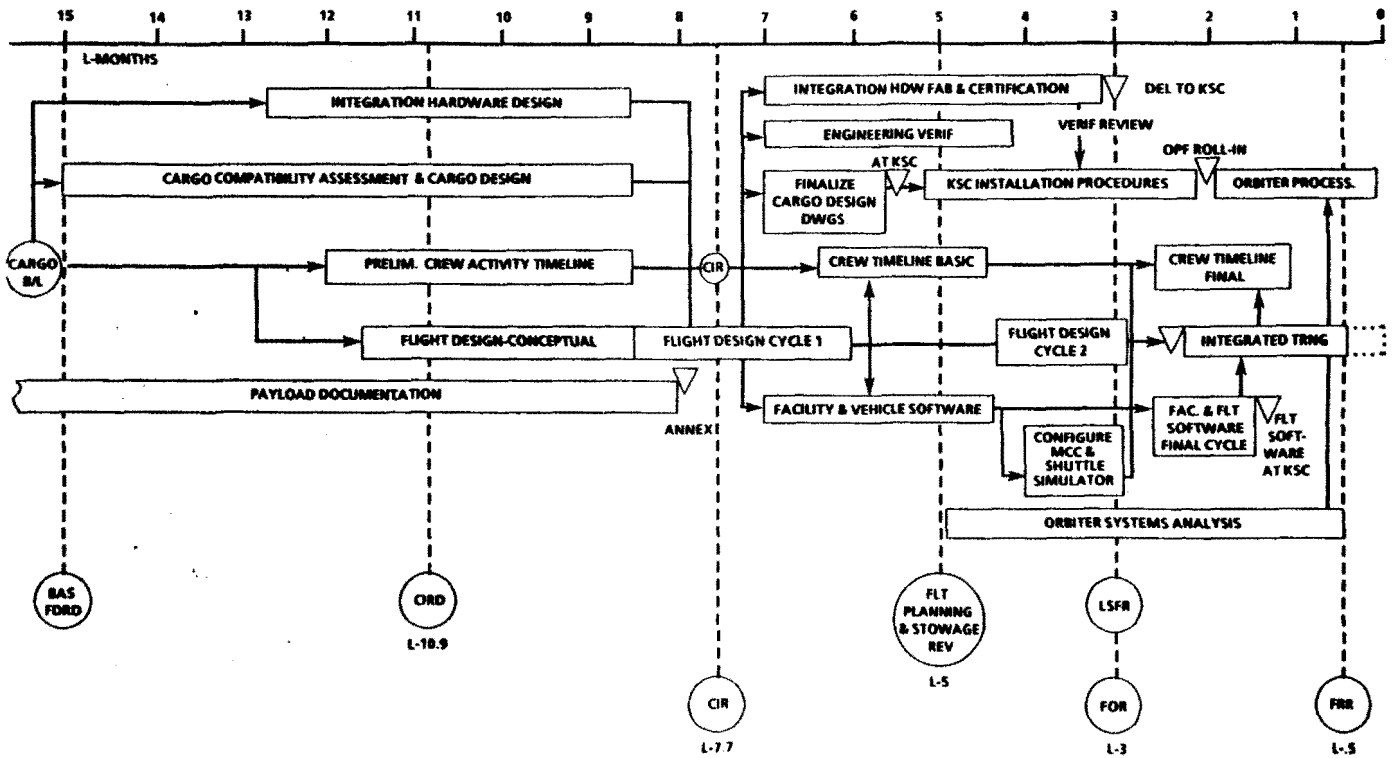

GENERIC TEMPLATE (CIR 7.7 MO)
15
14
13
12
11
10
9
8
7
6
5
4
3
2
1
0
L-MONTHS
INTEGRATION HARDWARE DESIGN
CARGO COMPATIBILITY ASSESSMENT & CARGO DESIGN
CARGO B/L
PRELIM. CREW ACTIVITY TIMELINE
FLIGHT DESIGN-CONCEPTUAL
PAYLOAD DOCUMENTATION
CIR
ANNEX I
INTEGRATION HDW FAB & CERTIFICATION
DEL TO KSC
ENGINEERING VERIF
VERIF REVIEW
OPF ROLL-IN
AT KSC
FINALIZE CARGO DESIGN DWGS
KSC INSTALLATION PROCEDURES
ORBITER PROCESS.
CREW TIMELINE BASIC
CREW TIMELINE FINAL
FLIGHT DESIGN CYCLE 1
FLIGHT DESIGN CYCLE 2
INTEGRATED TRNG
FACILITY & VEHICLE SOFTWARE
FAC. & FLT SOFTWARE FINAL CYCLE
FLT SOFT-WARE AT KSC
CONFIGURE MCC & SHUTTLE SIMULATOR
ORBITER SYSTEMS ANALYSIS
BAS FDRD
CRD
L-10.9
FLT PLANNING & STOWAGE REV
L-5
LSFR
CIR
L-7.7
FOR
L-3
FRR
L-.5

FIGURE VI-2

At 7.7 months before launch, the Cargo Integration Review (CIR) occurs. This is a critical point in the mission definition and launch process. The customer participates in this review. All baseline requirements for flight design, flight and ground operations, and crew size are defined. The engineering requirements for a particular mission are approved. The CIR essentially separates the design process and concept documentation from the actual Orbiter processing, installation and certification of the hardware and software, and final crew training and engineering verification of the various systems on the Orbiter. Typically 10 to 20 percent of the mission-specific preparation work is accomplished by the time of the CIR, and after the CIR the process is driven by the Shuttle mission preparation milestones.

At L–3 months the flight operations review (FOR) takes place. This freeze point allows the customer to review all final flight operations plans. At approximately the same time the launch site flow review (LSFR) takes place, at which the timing and flow of the Shuttle and its cargo through the Orbiter processing facility (OPF), the vehicle assembly building (VAB), and to the launch pad are all reviewed and baselined. No changes should occur after this point in time, but, of course, some do.

At L–2 weeks the Flight Readiness Review (FRR) takes place. Its purpose is to verify the fact that for this mission the hardware and software are ready for flight. It is here that the final commitment to a specific launch date and time is made.

At the FOR typically 40 to 50 percent of the work of the production process has been accomplished. At the time of the Flight Readiness Review, almost all of the work must have been accomplished because the Flight Readiness Review is not for the purpose of working out problems but merely to certify that problems have been resolved.

As the targeted flight date approaches, conditions are continually reviewed and last minute changes are made as necessary through a series of meetngs and teleconferences.

This simplified description does not begin to reveal the details of the launch production process but study of the template should give some indication of its complexity. With over 50 percent of the work typically needing to be accomplished in the last three months before launch, and with typically 20 or more flights in work at any given time, it should be clear that last minute changes can be very disruptive and costly.

The developmental or non-operational status of the Shuttle also contributed to problems as the flight rate increased. Less time between flights meant that results from one flight could not be incorporated into the early planning for the next one. In other words, any change resulting from feedback from the previous flight was necessarily a last-minute change. Because of the developmental nature of the Shuttle system, such changes were to be expected. At 24 flights a year there would be about two weeks between flights. Allowing some time for flight data analysis, this would mean that results of the previous flight typically would not be available at the Flight Readiness Review. Indeed, the O-ring erosion results of flight 61–C were available only immediately before the 51–L launch.

The complex and lengthy mission planning process was under increasing pressure and was being strained to achieve the planned launch rate. Two activities that were compressed were training of the flight crew and training of the ground launch crew.

Training.—When training and other preparations is compressed, progam quality is likely to suffer, and errors become more likely. Given the situation with NASA's safety program—which the Rogers Commission described as "silent"—errors were less likely to be detected before harm could occur. Errors can be caused by personnel taking shortcuts with respect to established procedures. Two examples are given in the Pre-Launch Activities Team report:

> The most significant error encountered was during the launch countdown. While preparing for propellant loading, the LH2 Orbiter to ET disconnect Valve was opened by the console operator. He had erroneously failed to follow the required steps in the OMI. A follow-on error was made in that this occurrence was not properly documented. Since proper documentation was not present, a full assessment of the problem was not made prior to the launch of STS 51-L. Flight data from STS 51-L indicated the valve did perform satisfactorily.
>
> Another major error occurred when the integrity seals on the ET aft restraints were broken and not reported. It is believed that the seals were broken in error, but the break of integrity was not reported in accordance with established procedures.
>
> The underlying factors contributing to these errors were not determined during the processing reviews.[105]

These errors apparently had no adverse impact on the mission, but indicate a breakdown of the discipline so necessary for a process as complex as launching a Shuttle.

Shuttle crew training is an important part of mission preparation. The crew of 51-L had training loads as high as 70, 63, 65, 59 and 58 hours in the several weeks before their launch. This was due to the fact that their training started some 3 weeks later than scheduled.

It must be noted that the crew also had 3 easy weeks during this period. During the weeks which included Thanksgiving, Christmas and New Year's they only trained 31, 27 and 49 hours, respectively. No harmful effects of compressed Shuttle crew training have been documented but common sense indicates that the situation must have been less than optimal.

It will be recalled that the launch of flight 61-C, which immediately preceded 51-L, was delayed several times. It was originally scheduled to launch on December 18th and eventually launched on January 6th. The Commission report describes how the launch date slips for 61-C became a scheduling factor for the training through integrated simulations for 51-L.[106] Delay of 61-C launch pushed a bow wave of tests at the Kennedy Space Center which required 51-L prime crew and/or mission control center resources and thereby

---

[105] Rogers Commission Report, Volume II, p. I-15.
[106] Ibid., p. J-13.

constrained the time at which integrated simulation training could be conducted. The 51-L training schedule was changed several times during the last weeks prior to launch due to launch slips of 61-C and the desire to suspend work between the Christmas and the New Year holidays. Eventually all 51-L training was accomplished with some change of spacing between the simulations. If the originally planned spacing of simulation training was optimum, then the changed spacing probably was not.

It is not clear exactly why the 51-L crew was late in starting its training, because it should have started training before the delays of the 61-C launch began. What is clear is that the crew training is a serial effort which cannot occur until software is available to drive the simulation computers.[107] The necessary software cannot be written until the specific flight configuration of the mission has been designed. This is a situation in which each event must wait on the completion of the previous one. In the case of 51-L there were delays and development of some software elements. But it is not clear that the development of these elements was in fact started on time. It is clear that there was considerable remanifesting of 51-L, for example during most of 1984 the Cargo Integration Review was scheduled to occur on September 4 but due to remanifesting this slipped and the CIR eventually occurred on June 18, 1985. In April 1985 a major change was made when the Orbiter assigned to the mission was changed (from OV-104 to OV-099) and major payload changes were made. This caused a slip of launch date from November, 1985, to January, 1986. There were small middeck payload changes in October, November and December of 1985.[108]

It is clear that these changes must have delayed the delivery of software which, in turn would delay the start of crew training. Crew training was not related to the accident, but it does seem clear that the system was breaking down (i.e., data presented in the Commission report shows that in January of 1986 the delays in the projected start of crew training were growing).[109]

Examination of the record shows that pressure to achieve the planned flight rate was forcing the crew to train later and later, which meant higher weekly training loads. This was very likely compromising the effectiveness of the crew training and thus the safety of the missions, although no harm had been documented at the time of the accident.

Manifest Changes—As described in detail above, the planning of a Shuttle mission requires more than a year of significant work, with the first major "freeze point" occurring 15 months before planned launch. A freeze point is a place in the mission planning schedule where decisions are made about the mission and its implementations. In theory, these decisions are made in a cumulative fashion so that earlier decisions do not have to be changed as the mission is refined through the planning process. Indeed, if there are no changes, this is in fact the way the system works; however, there are changes.

---

[107] Ibid., p. J-38.
[108] Ibid., pp. J-7-12.
[109] Ibid., p. J-42.

The first freeze point occurs when the mission is officially defined and payloads are assigned to a specific Orbiter. Another major freeze point occurs approximately seven months before launch at the Cargo Intergration Review (CIR). Typically, more than 80 percent of the work necessary to prepare a mission occurs after the Cargo Integration Review. Changes in the mission after the CIR tend to be much more expensive than changes made earlier in the process.[110] The Rogers Commission has adequately documented the fact that changes to the Shuttle manifest were common and major.[111] As of April, 1986, the six missions planned to follow flight 51-L which were not dedicated missions, i.e., not missions having only one customer, had a total of 30 changes or an average of five each after the start of the production process. Eleven of these changes were major that is, they involved the exchange of different types of major payloads.[112]

Manifest changes can be divided into four basic categories depending on the origin of the change. Some changes are caused by hardware problems such as when the Tracking and Data Relay Satellite was found to have a problem and was deleted from flight 51-E. As there is no reason to launch a faulty satellite, NASA virtually is obligated to allow such faulty satellites to be changed out.

The second category of manifest change results from what could be called "customer request." For example, many communication satellites have been rescheduled at the customer's request for business reasons. Again, NASA is in an awkward position because if the satellite is not needed, NASA would not want to be in the position of insisting that it be launched. (Although there have been cases when customers launched satellites and stored them on-orbit.)

A third category is caused by the belated recognition of operational constraints in the Shuttle system. For example, it has been found that a payload combination would exceed the landing weights for the transatlantic abort sites.

In another example of this type of change, it was found that there was no acceptable launch window for a planned combination of payloads which needed to be put in different orbits. It would seem that NASA could improve its mission planning production process to minimize this kind of manifest change by doing a better job of assessing the impact of operational constraints on payload combinations earlier in the planning process. Of course, one must allow for the late emergence of subtle operational constraints which would only be discovered as a result of deep analysis relatively late in the process. Nevertheless the MPOT report suggests that NASA sometimes carries unworkable flights on the manifest.[113]

The fourth category of manifest change is due to external factors, many of which are totally within NASA's power to deny. It appears that many of the Headquarters requests for changes are made in order to put on the manifest science experiments which

---

[110] NASA briefing on STS Production Process by Elaine Hofstetler—Presented to Committee Staff on May 19, 1986.

[111] Rogers Commission Report, Volume I, pp. 166-73; Volume II, pp. J-26-29, J-33-51.

[112] NASA briefing, May 19, 1986.

[113] Rogers Commission Report, Volume II, p. J-46.

are essentially payloads of opportunity. This would include the Get Away Specials (or GAS-cans). It has been considered highly desirable to give this kind of standby status to scientific experiments because they have had low priority on the manifest. That is, it is a way for such experiments to get a relatively early flight.

When changes are made in the manifest they tend to ripple through the system and affect not only the mission in work but also all the other missions in work. For example, changes mean rework—things need to be done over. Software for the mission may have to be rewritten. Inevitably, this causes some delay and compresses the time available for other work scheduled downstream in the process if the launch date is to be maintained. Of course, if the flight rate is to be achieved, launch dates must be kept.

Other missions are affected because the reworks necessary as a result of changes will pull engineers and technicians away from other projects. For example, in January, 1986, there were 21 flights in process. Given the fact that that resources available were finite, more work on one mission means that other missions have to wait. The result is that the mission preparations for the other missions also are compressed as they wait for the proceding mission to clear the process. The world system becomes less and less resilent, there is more and more overtime, and there is temptation to take shortcuts in the process.

It is important to note that "manifest changes" can also be viewed as "payload flexibility" as in the case of the "GAS-cans" mentioned above. Therefore, there may be a need to decide more specifically what we intend the Shuttle system to accomplish. If maximizing flight rate is to be the overriding consideration, then flexibility will have to suffer. However, if NASA adopts to rigid a posture with regard to payload changes, customers or users may object. For example, as pointed out above, there is no point in launching a faulty satellite. Most space operations are simply not mature enough for NASA to enforce a rigid manifest.

It would seem that a better way to minimize the adverse impacts of manifest changes would be to simplify the mission planning process so that freeze points could be later, that is nearer to the launch date, so that consequently changes would occur relatively earlier in the process, therefore with less impact.

Given the history of the program, it is known that there will be changes in the manifest and that the impact of these changes will be serious. It does not seem, therefore, that it would be particularly fruitful to try to develop analytical management tools to predict the impact of changes in the existing system (an effort NASA has suggested). Rather, effort should be directed toward developing a new, improved mission planning system. Also, the MPOT report claims that the impact of changes is already predictable, and can be budgeted.[114]

Operational Status of the System.—In addition to reconsidering the priority which should be attached to maximizing flight rate, there is also a need to consider the degree to which the Shuttle itself can be made more "operational."

---

[114] Ibid., p. J-38.

The Rogers Commission Report makes much of the fact that the Shuttle is not operational.[115] The same point was made strongly to Committee staff in interviews with personnel at Kennedy Space Center involved in launch processing. The Roger Commission made no recommendation on this matter and NASA in its reponse to the Commission has not direcly commented on it.

As early as 1981, senior NASA officials agreed that the Shuttle should be brought "to a cost-effective operational status" and that to that end Shuttle design should be "frozen".[116]

The Shuttle was declared operational after its fourth flight, but that the program clearly was not capable of functioning in a manner that would be called operational in any other milieu. Each Shuttle flight is, indeed, unique. Large amounts of software must be written *de novo* for each flight. This is appropriate for a developmental program but clearly but clearly will not work as NASA tries to move into a truly operational phase.

Prior to the Challenger accident NASA had realized that the mission planning process had to be drastically improved, probably through standardization. Unfortunately, pressure to increase the flight rate was driving all available resources into speeding up the existing system. There simply were not resources available to analyze the mission planning system and see where it could be simplified.[117]

If the Shuttle is to fly routinely, the mission planning system must be reworked to that end. For example, the Commission report makes the point that the two flight simulators were a bottleneck in the astronaut training process.[118] Undoubtedly this was true. What is not clear is whether there is another way. For example, would it be possible to develop specialized crews, say a group of astronauts trained to deploy communication satellites, who would need much less training to repeat identical or similar missions, thus reducing the demands on the simulators? The point is not that such savings must be found or can be found, but that they must be sought and resources must be dedicated to the search for such savings. If, indeed, no such standardization of mission planning is possible, NASA must face up to this fact and operate the Shuttle accordingly. As mentioned above, there are disturbing signs that NASA is moving once again toward achieving the highest possible flight rate without fundamentally changing its approach to Shuttle operations.

Pressure to Reduce Cost and Turn-around Time.—NASA was under pressure to reduce flight costs and to reduce turn-around time between flights. In some cases they could achieve both objectives at once by eliminating work done between flights (e.g., testing and refurbishment). A NASA memo shows that such actions were being pursued as early as August, 1981, after only one Shuttle

---

[115] Ibid., Volume I, p. 170–71.

[116] Memo from W. R. Lucas, Director, Marshall Space Flight Center, to James M. Beggs, Administrator, dated August 21, 1981; subject: "ET/SRB Productibility/Cost Reduction"; the relevant sentence reads: "I wholeheartedly agree with your statements that Shuttle performance requirements and design should be frozen so that we can concentrate all efforts on bringing the system to a cost-effective operational status."

[117] Rogers Commission Report, Volume II, p. J-31

[118] Ibid., Volume I, p. 170.

flight.[119] Attached to the memo are lists of activities to improve the producibility and reduce the cost of the SRB and ET. These include reduction of "mandatory government inspection requirements" for SRM processing by Thiokol and reduction of SRM propellant verification testing.

The point is not that these particular actions were unsafe, but that even very early in the flight program there were pressures on testing and inspection activities in the program.

Shuttle Process Issues.—Section VI.A.2.a. of this report, on "Shuttle Processing Issues" discusses several matters such as the availability of spares, overtine, and the adequacy of OMIs. It is clear that operating pressure aggravated and issues discussed there. For example, had there been no operating pressure there would have been less pressure on spares, less overtimd, and more time either to revise OMIs or to execute them.

Change Control Process.—Section VI.B.1.d. on "Change Control Process" discusses how the pressure to increase flight rate compromised the hardware change control process. An important factor is the developmental (i.e., not-yet-operational) nature of the Shuttle System which means that large numbers of significant hardware changes can be expected.

d. Other Safety Issues

Issue 1

What is the criticality of landing safety associated with programmed and abort landing sites and their local characteristics?

Findings

1. The Committee finds that many of the normal and abort landing safety problems will be alleviated when the Rogers Commission's and the Committee's (section V.A.1.b., this report) recommendations to upgrade the landing gear system are implemented. When the landing gear system is understood, straightforward calculations and operational rules will determine acceptable runway dimensions and conditions.

2. The Committee found no reason to fault NASA's current procedure on launch constraints based upon operational judgement and conservative rules on local conditions at planned abort and landing sites. However, since an obvious finding is that the Orbiter is a developmental system, it is axiomatic that unanticipated "dicey" circumstances will arise.

3. It was found that for the least landing gear system stress, runway preference is Edwards Air Force Base (EAFB) (concrete), KSC, and Rogers Dry Lake (EAFB "lake bed") in that order. No reason was found to invalidate the KSC runway design. The reasons for the "dry" course surface still prevail over concern about wear on tires designed for one landing. Additional constraints at KSC because of lesser lateral stabilized overrun area may be needed to bring its safety to the level of the EAFB runway.

4. The NASA Landing Safety Team's proposal to provide standard landing aids and arresting barriers at all sites and their em-

[119] NASA Memo from W.R. Lucas, to James M. Beggs, Administrator, dated August 21, 1981.

phasis on runway surface characteristics for repetitive tire use takes on a new dimension that is in addition to the Rogers Commission's recommendations.

5. Weather, by far, is the most significant factor governing operational decisions, Orbiter damage, and landing safety. The constraint is simply that acceptable weather must be forecast with confidence within the time frame needed. Ultra-conservative rules prevail because of the predictable unpredictability of Cape weather. New and innovative local weather analysis and forecasting research is a high priority. The African Coast and southwestern United States sites enjoy more stable and predictable weather.

Recommendations

The first priority to achieve an acceptable degree of landing safety and to have a sensible base to work from for improvement is to implement the recommendations of the Rogers Commission and the Committee on the landing gear system improvement to attain an operational capability. Then:

Instrument the system, and schedule all landings at Edwards runway for systematic concurrent testing until the landing gear system is understood.

Write a clean sheet set of rules based on results.

Determine the risk of accident with the B-747 Shuttle Carrier Aircraft (SCA) and its impact upon the Shuttle program.

Extend every reasonable effort to assure a mission planning process to minimize the need for abort site landings.

Reevaluate and determine the degree of risk acceptable at abort site landings and bring abort site capability up to meet that risk level.

Expand astronaut matched team flight landing practice to cover all known exigencies. Propose additional training craft if necessary.

Join in a venture with NOAA to invent new technology and techniques to learn new ways to understand the dynamics of Cape Kennedy weather phenomena to supplant current inadequacy to forecast two hours ahead.

Discussion

This discussion assumes that landing gear system improvements are to be implemented. The substance of the testimony and results of the Committee investigation are fairly clear.

The EAFB runway will remain the primary programmed landing site for the duration of the Shuttle program simply because of the capricious nature of the Cape weather. All landing parameters favor Edwards runway as the best for safety and it approaches 100 percent predictable availability.

The safety of Rogers dry lake is permanently compromised because of the lake bed surface. Its firmness and surface strength are variable and the surface has considerable debris scattered on it. Should the tires blow on one strut, it would dig in and the Orbiter would not be controllable as it would be on a concrete runway with nose wheel steering and brakes. This is also true of stabilized lateral and longitudinal overrun areas of the concrete runway.

From the body of testimony, it can be deduced that given a landing gear system that meets operational requirements, acceptable weather, and an adequately trained pilot, the Orbiter can consistently achieve the acceptable level of low risk landings that was originally intended at Edwards and KSC. The worst KSC case is the heavy weight abort Return To Launch Site (RTLS) landing. Night landings at these sites add an element of risk that cannot be evaluted until day landing confidence is restored. The only astronaut testimony on night landings was not favorable.[120]

Landing safety at remote abort sites presents, by far, the worst case including all facets of navigation, weather, energy management, depth of pilot training, other air traffic intrusion, alignment, approach, heavy weight high speed landing, narrow and short runways, and fire and rescue support, and perhaps even terrorism or sabotage. In short, the classical emergency landing is just that—an emergency landing. It will surely test the skill of the pilot. The only sure cure for abort landing exposure is a successful launch.

Testimony gave reference to one RTLS site (KSC), five TAL (Trans Atlantic or Trans Abort Site) sites (Casablanca, Dakar, Moron, Rota, Zaragoza), and three AOA (Abort Once Around) sites (EAFB, White Sands Northrop, KSC). At least one each of these must be available within the rules of visibility, wind, dew point, precipitation, ceiling, cloud cover, turbulence, and gusts, and provide TACAN, MLS, PAPI (Precision Approach Path Indicators), and Ball Bar lights as deemed necessary for the mission; the RTLS within 25 minutes of launch, the TAL at about 35 minutes, and the AOA in an hour and 45 minutes.

The Orbiter is not a good handling airplane to fly. The Orbiter landing is the most demanding task of airmanship expected of an aviator today. It is a complex and sophisticated blend of automation, systems management, and manual skills:

The Orbiter re-enters with a 1100 mile cross track capability to begin the Terminal Area Management phase, 52 miles out at Mach 2.5 and 82,000 feet.

Computer energy management delivers the Orbiter to the alignment circle on TACAN where the pilot takes over at three minutes out on a 19 degree glide scope aligning on PAPI lights.

At 13,000 feet, 6 miles and two minutes out, he initiates flare to intercept the 1.5 degree glide slope at 275 knots.

Guiding on the Ball Bar lights, he approaches and lands around 200 knots depending on his weight.

At 140 to 120 knots, he begins to brake and decelerates to a stop.

If MLS terminal navigation is not available, the pilot can rely upon onboard radar for precision altitude and use his heads up display to assist what is nominally a visual approach and landing. There is no room for computer, navigation or pilot error. Training aircraft training and practice is an element of major importance to successful Orbiter landings under the variety of conditions facing the pilots. Unrationed crew team flight training is deemed essen-

[120] Rogers Commission Report, Volume V, p. 1455.

tial to landing safety. Conversely, suggested autoland systems for this application did not find much support because they would pose a whole new development and certification hazard.

Landing safety will make a lot more sense if and when the cloud of imminent landing gear system failure is dissipated. That has been a pervasive note through the entire testimony and investigation.

Issue 2

Has adequate provision been made for crew safety in case of in-flight emergencies? That is, has adequate provision been given to launch abort options and crew escape options?

Findings

1. Crew escape options were considered when the Shuttle was originally designed and the basic situation has not changed. Many initially attractive options do not significantly reduce risk to the crew either because they may not reduce exposure to the principal hazards or because they add risks of their own.
2. A crew escape system for use in controlled gliding flight might be feasible and worthwhile.
3. Crew escape during the ascent phase appears infeasible.
4. Launch abort during SRB burn appears impossible but it may be possible to decrease risk to the crew after SRB separation, primarily through mission design.

Recommendation

NASA should continue to respond to the recommendations of the Rogers Commission regarding (i) crew escape during controlled gliding flight and (ii) increasing the possibility of successful emergency runway landings. NASA should re-examine all crew survival options and report to the Committee on its findings.

Discussion

Before addressing the particulars of the findings and recommendations regarding launch abort and crew escape a few general comments on safety and risk will establish a useful framework.

Any new safety equipment installed on the Orbiter will bring with it its own new risks. It will also add weight to the Orbiter and will have associated capital and operating costs. Each of these must be addressed.

New Risks.—Consider for example the possibility of adding ejection seats to the Orbiter. The United States Air Force experience with ejection seats has been that they are only about 80 percent effective.[121] The point is that ejection seats are not a panacea. Any safety equipment has a chance of failing; ejection seats in particular always have a potential of premature activation which would result in the crew being ejected when there is no need.

Additional Weight.—In order to accomplish its purpose, the Shuttle must put payloads, i.e., weight, in orbit. Adding weight to

[121] Briefing to Committee Staff, May 28, 1986, "Report of the First Stage Abort Options History Task Group Chartered by the Mission Planning and Operations Team"—Barney Roberts, Advanced Programs Office, Johnson Space Center.

the Orbiter reduces the payload weight that can be orbited and therefore reduces the justification for the program. This is perhaps made clearer by considering a *reductio ad absurdum.* Suppose one could develop a new escape system—perhaps an ejection pod which could reduce risks to the crew by 90 percent but weighed approximately 65,000 pounds. Since the Shuttle payload capability is only about 65,000 pounds there would be no remaining payload capacity in the Shuttle, and some risk would still remain. There would be no point to installing such a system because it would be a very bad trade. Evidently, one must do an engineering cost-benefit calculation and decide if the benefit is worth the penalty for each proposed change.

New Costs.—The same type of cost-benefit calculation must be done in the financial dimension. It is important to emphasize that the question is not "how much is a life worth?," but rather "where can an extra amount of funding best be spent to reduce total risk to the crew, the mission, and the Orbiter?"

Risks will never be zero—what NASA must do is to better understand the risks and minimize the most dangerous exposures.

The risk, cost and weight penalties of crew escape systems that could hope to operate effectively while the SRB are thrusting are very large. This dictates that it is much more efficient to put program resources into reducing risks by improving the reliability of the SRB's and the whole Shuttle system during the period of time that the SRB's are thrusting. For example, if one of the SRB's should develop a problem so that there was a need to separate the Orbiter from the SRB's and External Tank, it is essentially impossible to do this successfully while the SRB's are still thrusting. There are potential means of terminating SRB thrust which amount to explosively opening holes in the rocket casing. The holes allow the burning gases to exit the casing at several places so that there is no net thrust. Such a mechanism has the potential of premature activation which could lead to loss of the crew and the mission. In addition, the resulting deceleration loads on the Orbiter would require significant redesign, if the Orbiter were to survive.[122]

A large part of the problem is that the launch situation is very dynamic. Decisions and implementation of decisions must be made very rapidly. The decisions are binary; that is, either "go" or "no-go," and the implementation must be largely automated for speed of execution. Thus, if a premature activation begins it will almost certainly go to completion.

In the case of 51-L accident, the first ambiguous indication of a problem came at about 65 seconds into the mission. At 72 seconds the system was coming apart and by 74 seconds the Orbiter was destroyed. The first signs of trouble were ambiguous because indications that the Orbiter was adjusting to aerodynamic forces due to the leak in the SRB joint appear very similar to signals generated when the Orbiter responded to upper atmosphere winds. It would be very risky to initiate any kind of crew escape action based on

[122] Cmte Hgs, Transcript, June 25, 1986, pp. 132-35, 139-41. Former astronaut, General Thomas Stafford, testified strongly in favor of crew escape systems but seemed to represent a minority view.

this sort of signal. The Solid Rocket Boosters began coming off the system at 72 seconds, after which an escape system might well have been inoperable due to mechanical deformations of the Orbiter structure under the aerodynamic loads that resulted. Thus, there was a period of time of something less than 9 seconds during which some kind of escape system might have been able to help the crew. It seems clear that attempting to develop a system to respond effectively to a situation such as this would be unproductive and that it would be wiser to improve the safety and realiability of the system during this ascent phase.

After the termination of SRB thrust, immediate crew escape is difficult because the Orbiter has achieved a very high altitude. However, under a range of circumstances it is possible to fly the Orbiter back to a controlled gliding landing at a runway. Under other circumstances, for example if the main engines fail shortly after SRB termination, the Orbiter may be forced to ditch into the ocean and such a ditching is not survivable.

Therefore the Rogers Commission recommended, and the Committee agrees, that NASA should attempt to minimize these risks. That is, NASA should take steps to increase the probability of the Orbiter being able to fly to a landing site and NASA should attempt to develop a way for the crew to escape from controlled gliding flight, for example, if the Orbiter is approaching a ditching or a crash landing.

After SRB termination a principal risk is that the Orbiter could lose one, two or three main engines. Depending on when and how this occurred it might be possible to fly the Orbiter to a landing site. It may be possible and perhaps practical to increase the probability of the Orbiter successfully accomplishing this maneuver through flight design. That is, it might be possible to accept somewhat reduced payloads and achieve more conservative trajectories which would minimize the exposure of the Orbiter to ditching or crash landing if main engine failure were to occur during the accent phase.

If the Orbiter finds itself in a situation (due to Main Engine failure or other failure) where it cannot fly to a runway but is otherwise under control, the crew might be able to escape during the controlled gliding descent. This would apply not only during the ascent phase but also during the landing phase. For example, if the reentry trajectory were miscalculated and the Orbiter could not reach the planned landing site the crew might have adequate time to bail out. There is a change that such a bailout system could be achievable with acceptable performance penalties. Certainly this last option—crew bail-out during gliding flight—must be very carefully studied.

The trade offs and calculations that have to be made in the area of crew escape and launch abort are activities in which astronaut involvement would be most useful.

Astronauts clearly represent the principal source of flight experience and therefore can make major inputs to decisions regarding what is practical to accomplish during flight. It is pointless to add risks, weight, and cost for a system that cannot be operated by the astronauts during flight conditions. Involvement of astronauts in management is discussed in section VI. B. 2. a. of this report.

In summary, space flight will always be a bold and dangerous venture. NASA must work to better understand the risks of space flight and in particular the risks of each Shuttle launch and to reduce these to an acceptable level.

B. Management Issue

1. TECHNICAL MANAGEMENT

a. Risk Management Issues

Issue

There a coordinated and effective risk management program in the NSTS?

Findings

1. NASA does not explicitly use a centralized program that coordinates all the factors that encompass an adequate risk management program.
2. As a result of the accident, NASA is reexamining the Failure Modes and Effects Analyses (FMEA) and Hazard Analyses (HA) to reassess risks associated with the designs of Shuttle subsystems.
3. NASA's lack of statistical data on the performance of certain components will limit the usefulness of sound engineering judgment in much the same way as it limits the usefulness of probabilistic risk assessment.

Recommendations

1. NASA should develop and provide to the Committee a description of an overall risk management program as it relates to the Space Shuttle. This effort should include a determination of whether or not a more centralized coordination of a risk management program and issuance of direct risk management guidance directives are needed.
2. NASA should review analytical methods utilized in the performance of risk assessment, including statistical analyses, trend analyses and probabilistic risk assessment methodologies to determine their applicability to the NSTS program. Assistance from the National Academy of Sciences, or other appropriate organizations with expertise in these matters, may be required to adequately perform this review.
3. NASA should review its certification testing to ensure that all critical items are adequately tested. Data obtained from these tests should be used when appropriate in conducting a formal risk assessment.

Discussion

NASA does not have a specifically labeled risk management program. The process is accomplished by the agency through its configuration management program and the FMEA performed on each component of the Space Shuttle. The identification of critical items is the principal product of these analyses. The ability to make the programmatic or engineering changes necessary to enhance the

safety and performance of flight systems while controlling costs and schedule is the task of the risk management activity.

The process of risk management as applied to systems such as the Shuttle can be described schematically as shown in Figure VI-3,[123] which shows the various steps that might be imposed upon flight systems such as the Shuttle through a risk management program.

[123] Douglas B. Feaver, "Success Relaxed NASA's Vigilence", *Washington Post*, May 26, 1986, pp. A-1, A-10.

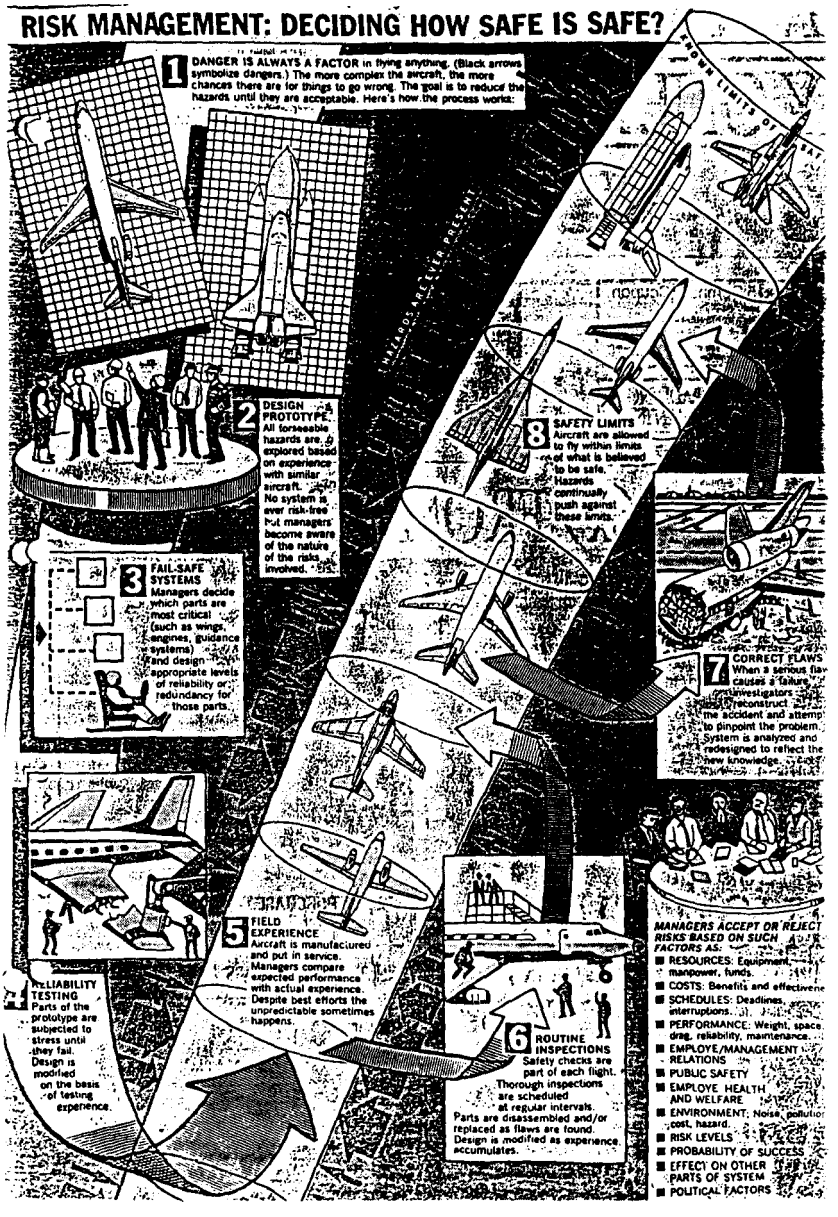

RISK MANAGEMENT: DECIDING HOW SAFE IS SAFE?
1 DANGER IS ALWAYS A FACTOR in flying anything. (Black arrows symbolize dangers.) The more complex the aircraft, the more chances there are for things to go wrong. The goal is to reduce the hazards until they are acceptable. Here's how the process works:
HAZARDS ARE ALWAYS PRESENT
2 DESIGN PROTOTYPE. All foreseeable hazards are explored based on experience with similar aircraft. No system is ever risk-free, but managers become aware of the nature of the risks involved.
3 FAIL-SAFE SYSTEMS Managers decide which parts are most critical (such as wings, engines, guidance systems) and design appropriate levels of reliability or redundancy for those parts.
4 RELIABILITY TESTING Parts of the prototype are subjected to stress until they fail. Design is modified on the basis of testing experience.
5 FIELD EXPERIENCE Aircraft is manufactured and put in service. Managers compare expected performance with actual experience. Despite best efforts the unpredictable sometimes happens.
6 ROUTINE INSPECTIONS Safety checks are part of each flight. Thorough inspections are scheduled at regular intervals. Parts are disassembled and/or replaced as flaws are found. Design is modified as experience accumulates.
7 CORRECT FLAWS When a serious flaw causes a failure, investigators reconstruct the accident and attempt to pinpoint the problem. System is analyzed and redesigned to reflect the new knowledge.
8 SAFETY LIMITS Aircraft are allowed to fly within limits of what is believed to be safe. Hazards continually push against these limits.
KNOWN LIMITS OF SAFE
MANAGERS ACCEPT OR REJECT RISKS BASED ON SUCH FACTORS AS:
RESOURCES: Equipment, manpower, funds.
COSTS: Benefits and effectiveness
SCHEDULES: Deadlines, interruptions.
PERFORMANCE: Weight, space, drag, reliability, maintenance.
EMPLOYE/MANAGEMENT RELATIONS
PUBLIC SAFETY
EMPLOYE HEALTH AND WELFARE
ENVIRONMENT: Noise, pollution, cost, hazard.
RISK LEVELS
PROBABILITY OF SUCCESS
EFFECT ON OTHER PARTS OF SYSTEM
POLITICAL FACTORS

FIGURE VI-3

Top NASA managers lack a clear understanding of risk management. Dr. Fletcher, NASA's Administrator, made the following statement, when asked by Mrs. Lloyd to "describe the elements of NASA's risk management activities . . .":

> Well, risk management is a pretty generic term. Risk management is decided in Headquarters in terms of what are the chances of an overall failure of a system under a given set of circumstances. When you get down to the flight team, the launch crew in those last several hours or couple of days, risk management is an entirely different thing. They have to look at the factors that have come up just before launch and assess whether this is a risk we want to take. This is a judgement question; you can't make calculations at this point.[124]

Dr. Silveira, NASA's Chief Engineer, testified on the same day that,

> As we had mentioned in the testimony that we gave previously . . ., the only time that we had gone into trying to assess a probability, if you will, or a risk, was as a result of a request that was made by DOE for their analysis that they were performing at that time, to assess the probability of failure of the vehicle, to assess the danger when we are flying the RTG's, the radioactive material.
>
> As far as in our program, and any major decisions that we would make, we have a number of reasons why our past history had indicated that that was not a good way of doing it. As a result, we don't use it generally in our risk management, we prefer using things like the failure effects and analysis that we do; the technical engineering judgement, using things to control our failures rather than depending upon a probability analysis to assess it.[125]

However, Mr. Robert Thompson, who was Shuttle Program Manager from 1970 to 1972, testified on July 24th before the Committee in a much less ambiguous fashion regarding his view on the importance of risk management:

> I would first like to make an observation on the decisionmaking process. Evidence, in retrospect, points to a long period of time, especially based on post-flight inspections when the joint design weakness was 'sending a message' and the true potential of this message was not perceived and reacted to.
>
> This, combined with perlaunch discussions between Marshall and Thiokol, points out the need that must pervade the Shuttle management team in the future. A very strong risk management . . . I have parentheses around risk management. I will be happy to expand on that. It has a certain meaning to me. A very strong risk management organization must be kept in place and a continuing search for potential failures must be maintained. . . .

---

[124] Cmt Hgs, Transcript, June 12, 1986, p. 186.
[125] Ibid. p. 187.

> The role of the program manager in this risk management organization must be very strong and clear. The entire program organization from top to bottom must be clearly chartered and as people come and go these organizational relationships must be carefully maintained.[126]

Based upon the divergences of these testimonies, the Committee concluded that although NASA's Space Transportation System program contains the elements of a risk management program, there needs to be a new and heightened coordination of the separate activities by NASA in order to minimize the risks inherent in Shuttle flights.

The FMEAs determine the worst case "What if" scenarios for all possible failure modes and their potential worst case or intended effects.[127] As a result of performing the FMEA, a list of critical items is identified. NASA's FMEA assure that all Criticality 1 and 1R systems are properly identified and classified. The failure of these items would produce loss of life and/or loss of vehicle. The FMEA applies strictly to the hardware associated with the NSTS and is "bottoms-up" analysis, in which a single component failure is traced and its effect on a particular subsystem, subsystem interfaces, and the overall flight systems is determined. Accompanying the FMEA is the Hazard Analyses (HA) which is, according to NASA, a "top-down" approach that takes into account human factors in evaluating the consequences of particular accidents or accident scenarios. Hazard Analysis is the basic tool of the safety evaluation.

The FMEA as used by NASA assigns no probability numbers to event sequences along a given failure path. Although NASA regards the methodology of FMEA as rigorous, within the agency there was a wide variation in the engineering judgments among the design engineers and senior management in the NSTS program on the probability of failure of the Shuttle.[128] The Committee, in hearings held earlier this year related to the safety aspects of the Shuttle Centaur in its utilization of Radioisotope Thermoelectric Generators on board the Shuttle spacecraft, also found wide discrepancies in the estimate of the failure probability for the Solid Rocket Booster among the experts.[129]

NASA has rejected the use of probability on the basis that such techniques are insufficient to assure that adequate safety margins can be applied to protect the lives of the crew. They also argue that their problem correction procedures preclude the establishment of a sufficient statistical database, because once a single point failure has been identified through the FMEA, steps are taken to design

---

[126] Ibid., July 24, 1986, p. 106.

[127] It is the prime responsibility of the design engineers working with reliability analysts to perform the FMEA in accordance with guidelines established in NASA documents (Appendix VI-C). These documents are provided as part of each statement of work submitted to the contractor. From such FMEAs, a Critical Items List is established in which particular components under the responsibility of the contractor are categorized in accordance with their criticality to the mission, crew, and/or spacecraft. Included as Appendix VI-D is NASA's document 100-2G entitled *Reliability Desk Instruction, Flight Hardware Failure Mode and Effects Analyses (FMEA) and Critical Items List (CIL).*

[128] Rogers Commission Report, Volume II, p. F-4.

[129] Hearing before Subcommittee on Energy Research and Production and Subcommittee on Space Science and Applications of the Committee on Science and Technology, 99th Cong., 2nd Sess., March 4, 1986 (No. 97). "Review of RTG Utilization in Space Missions."

the safety features into the component, thereby eliminating the failure mode or establishing sufficient redundancy to preclude catastrophic failures associated with the particular component. This change of the component means that earlier data no longer apply.

On the other hand, with respect to certification testing of the Space Shuttle Main Engine, NASA seems to argue that a useful statistical data base can be generated even though the configuration of the engine is changed as data is accumulated. That is, as running time is accumulated in SSME certification testing, major components—e.g., the high pressure turbopumps—are replaced, and yet NASA seems to believe that the total accumulated running time has some meaning for determining engine life time.[130]

All subsystems of the NSTS are intended to meet design requirements that incorporate the fail-safe features as a minimum with fail-operational/fail-safe criteria placed on all Orbiter avionics systems.[131] Fail-safe requirements are defined as designs which can withstand a single failure and permit return of the crew to the ground safely. Fail-operational/fail-safe is defined as permitting two sequential failures while enabling crew return. There are some parts of the NSTS which must be exempted from meeting these criteria. The reason is that it is not possible to improve the safety features of these systems through redundancy or other means. Such systems are the primary structure, the thermal protection system, pressure vessels and the premature firing mode of the pyrotechnics. For example, the pressure vessel cannot be provided with redundancy in a safe manner because addition of another pressure vessel would only enhance the failure probability or the criticality of this component.

The FMEA is a very conservative analysis according to NASA since it provides information on worst case situations of all possible failure modes and the potential worst case effects. Even so, the Committee was unable to determine the degree to which flight anomalies and trend analyses in historical performance data are utilized to insure that the appropriate measures are taken in the design and testing of various critical components to assure ultimate safety and minimization of risk.

NASA is presently reviewing the 748 Criticality 1 items and the 1,621 Criticality 1R items. Based upon a series of tests and analyses and the availability of methods and instrumentation to detect problems associated with various Criticality 1 and 1R items, waivers are given to permit flight of critical items. Before a waiver is granted, according to NASA, extensive documentation and review of each item on the Critical Items List (CIL) for which a waiver has been applied must be undertaken and approved all the way through Level 1 management. There is a difference between the number of waivers granted and the total number of items on the Critical Items List. For Criticality-1 items this difference reflects the number of systems exempted from the criteria of fail-safe or fail-operational/fail-safe. NASA, however, does not distinguish in its quality control procedure between exempted items and those items which are not exempt from the waiver process. According to

[130] Rogers Commission Report, Volume II, pp. K25-26.
[131] NASA Briefing on July 10, 1986.

NASA, this categorization of exempt versus waiver is strictly a management technique for identifying components and systems on the Space Shuttle in terms of their safety compatibility.

The Committee finds the FMEA to be an appropriate method for identifying the Critical 1 and 1R elements of the NSTS; however, not all the elements so identified pose an equal threat. Without some means of estimating the probability of failure of the various elements it is not clear how NASA can focus its attention and resources as effectively as possible on the most critical systems. Moreover, waivers can be granted without assurance that an adequate level of safety has been achieved.

b. Launch Decision Process

Issue 1

Is the process for establishing launch constraints and dealing with them effective?

Findings

1. There is no clear understanding or agreement among the various levels of NASA management as to what constitutes a launch constraint or the process for imposing and waiving constraints.
2. Launch constraints were often waived after developing a rationale for accepting the problem rather than correcting the problem; moreover, this rationale was not always based on sound engineering or scientific principles.

Recommendations

1. NASA should establish rigorous procedures for identifying and documenting launch constraints. The individual(s) responsible for implementing this procedure should be clearly identified, and well defined and understood criteria for waiving the constraints should be established.
2. NASA should exercise extreme caution in waiving launch constraints before correcting the problem that led to the launch constraint. The rationale should be based on rigorous scientific/engineering analyses or tests and should be understood and accepted by the Program Manager.

Discussion

No single system exists for establishing and dealing with launch constraints within the Shuttle Program; for example, Marshall maintains their own system through their Problem Assessment Center (PAC) to deal with problems affecting the propulsion system. In testimony before the Rogers Commission, Mr. Mulloy explained that the system was established to provide visability for problems relating to the propulsion system and a "launch constraint" was in effect a flag to alert the Project Office to address the problem at the Flight Readiness Review.

> A launch constraint means that we have to address the observations, see if we have seen anything on the previous

> flight that changes our previous rationale, and address that at the Flight Readiness Review.[132]

The NSTS Program Manager stated that he was unaware that a launch constraint had been imposed as a result of the O-ring erosion. Unawareness of this launch constraint was also claimed by the Level I Program Office and key Thiokol personnel: Mssrs. Ebeling, Kilminster, Russell, McDonald, and Boisjoly.[133]

In staff briefings, it was suggested by NASA personnel that perhaps "launch constraint" was a poor choice of words to describe this process for flagging problems. Those individuals who claimed no knowledge of a launch constraint had certainly been made aware of the O-ring erosion problem. This problem and the resolution had been discussed throughout the system including the FRRs. Therefore, although it is difficult to understand why the Program Manager and others weren't more familiar with the Marshall PAS, as a practical matter it probably had little effect on the final decisions. These "launch constraints" were potential problems that had to be resolved prior to flight and the Level III Project Managers were responsible for resolving any problems dealing with their systems. During the Rogers Commission hearings, Mr. Mulloy acknowledged that he had ultimate responsibility for waiving the launch constraints and ultimate responsibility for the launch readiness of the Solid Rocket Boosters.

Although the O-ring erosion continued to occur, and with no apparent pattern, the SRB Project Manager repeatedly waived the launch constraint. Throughout the Rogers Commission hearings and the hearings of the Committee on Science and Technology, NASA witnesses continually justified their decision to continue flying the Shuttle based on their previous successful flights. This reliance on their "experience base" was a major factor in the repeated waivers of the Marshall imposed launch constraint on the SRBs. Chairman Rogers asked Mr. Mulloy what was meant by "addressing" the problem, and Mr. Mulloy responded:

> I mean present the data as to whether or not what we have seen in our most recent observation, which may not be the last flight, it may be the flight before that, is within our experience base and whether or not the previous analyses and tests that previously concluded that was an acceptable situation is still valid, based upon later observations.[134]

Mr. Mulloy also explained his reliance on the experience base in testimony before the Science and Technology Committee:

> That was presented to me as a rationale to continue flying, one we had seen it on STS-2, what we saw on the last flight wasn't as bad, therefore it was an acceptable risk.[135]

---

[132] Rogers Commission Report, Volume V, p. 1513.

[133] Ibid., p. 1590; note: Yet it was Mr. McDonald who wrote a letter to the SRB Project Office recommending that the O-ring problem be dropped from the Problem Assessment System (PAS), which was in fact equivalent to removing the launch constraint.

[134] Rogers Commission Report, Volume V, p. 1513.

[135] Cmte Hgs, Transcript, June 17, 1986, p. 151.

The Committee concurs with Dr. Feynman's analysis that NASA had no understanding of the O-ring erosion phenomenon, and their rationale for accepting it was not based on sound engineering principles.

> . . . The acceptance and success of these flights is taken as evidence of safety. But erosion and blow-by are not what the design expected. They are warnings that something is wrong. . . . The fact that this danger did not lead to a catastrophe before is no guarantee that it will not the next time, unless it is completely understood. . . . The origin and consequences of the erosion and blow-by were not understood . . . officials behaved as if they understood it, giving apparently logical arguments to each other often depending on the "success" of previous flights.[136]

Issue 2

Are the launch commit criteria procedures adequate to ensure the safety of the mission?

Findings

1. The procedure used for developing launch commit criteria is systematic and thorough; however, violations of the criteria do not necessarily mean "no go". Therefore, NASA sometimes relied on engineering judgments made during the terminal countdown in determining whether to launch.
2. Launch commit criteria were sometimes waived without adequate engineering analysis or understanding of the technical reasons for establishing the criteria.

Recommendations

1. NASA should review the launch commit criteria procedures, especially those for dealing with violations, to lessen the reliance on engineering judgments under stress.
2. When situations arise where "real time" engineering judgments are unavoidable, NASA should adopt a more conservative approach to waiving previously established criteria. In no case should a criterion be waived without a thorough understanding of the rationale for the establishment of the criterion.

Discussion

Launch commit criteria define limits on specific system parameters which are required to be monitored during the terminal countdown. When these limits are exceeded the launch is held until the condition is corrected or an acceptable alternate capability or procedure is instituted.

Proposed criteria are developed by NASA and contractor personnel and are submitted to the NSTS Program Office for review and disposition. All changes are controlled by the Level II PRCB (Program Requirements Change Board) and all launch commit criteria are reviewed prior to each flight at the launch site flow review (8 weeks prior to launch), the Flight Readiness Review and the L–1

[136] Rogers Commission Report, Volume II, p. F–1.

review. Where practical Launch Commit Criteria include preplanned decisions on courses of action to be taken when violations occur.

The process described for developing and controlling the launch commit criteria is systematic and thorough; however, in briefings by NASA personnel it was learned that it is not uncommon to experience violations of the specified limits. These can often be resolved in a straight forward manner based on a prior plan of action; however, the Committee is concerned that in those situations where no preplanned course of action is available, real time engineering decisions are being made under the stress that is inherent in a pre-launch environment. This is particularly undesirable when it is perceived that there are pressures to launch.

For example, it was learned that on the morning of the scheduled launch of STS 51-L the Mission Evaluation Room (MER) Manager requested a waiver of the Launch Commit Criteria lower limit of 31 degrees F.[137] The Flight Director can not unilaterally waive launch commit criteria and since the temperature at launch was above 31 degrees it became unnecessary to pursue the matter further. Had it been necessary to waive the criterion, the Flight Director would have advised the Program Manager who then would have orally polled the Project Managers before making the final decision. One can only conjecture at this point what the decision would have been; however, the Committee is concerned that at least two key managers in the decision making chain (i.e. the MER Manager and the Flight Director) were prepared to waive the criterion without thoroughly understanding it.

Issue 3

Are the launch readiness review procedures and communications adequate?

Finding

The Committee finds that the review procedures and communications used to assure flight readiness were systematic, thorough, and comprehensive and provided ample opportunity for surfacing hardware problems prior to flight. Level I FRRs are usually recorded (audio); however, there is often no record made of other key prelaunch meetings.

Recommendation

NASA should make every reasonable effort to record meetings where key decisions might be made; in particular, all formal Flight Readiness Reviews, including the L-1 and the Mission Management Team meeting should be recorded, where feasible by video.

Discussion

The Flight Readiness Review process encompasses a series of reviews beginning with contractor reviews of their systems, and going through the Project Management review (Level III), and NSTS Program Management review (the "Pre-FRR"), and culmi-

[137] Rogers Commission Report, Volume II, pp. J22-23.

nating in the Level I (Headquarters) review which is referred to as "the" FRR. One additional formal review takes place 24 hours before launch and is called the "L-1" review. This is conducted by the Mission Management Team (MMT) which is appointed by the Associate Administrator for Space Flight at the time he calls for the FRR. All open work and action items identified at the FRR are closed out at the L-1. In addition to conducting the L-1 review, the MMT functions as a technical advisory body for the Program Manager and is on call beginning 48 hours before the launch until after the mission is completed and the Orbiter is safed.

The Committee concurs with the Rogers Commission that NASA should record key pre-launch meetings; however, the Committee finds no basis for concluding that the Flight Readiness Review procedure is flawed; on the contrary, the procedure appears to be exceptionally thorough and the scope of the issues that are addressed at the FRRs is sufficient to surface any problems that the contractors or NASA management deem appropriate to surface. However, the Flight Readiness Reviews are not intended to replace engineering analysis, and therefore, they cannot be expected to prevent a flight because of a design flaw that management had already determined represented an acceptable risk. In addition all the appropriate offices, including the Chief Engineer representing SR&QA, are represented at the FRRs. Specifically, from the first evidence of O-ring erosion to the final decision to launch 51-L, the process provided ample opportunity to review and assess the severity of the problems; moreover, all levels of NASA management were made aware of the erosion.[138] However, a process is only as effective as the responsible individuals make it. For example, see section VI B.2.c. on the weakness in the SR&QA organization.

Issue 4

Was the failure to inform the Level I or Level II Program Managers of the Teleconference involving NASA and Morton Thiokol on the eve of the launch a factor in the decision to launch?

Findings

1. The Committee finds that Marshall management used poor judgment in not informing the NSTS Program Manager or the Level I Manager of the events that took place the night before the launch, specifically the stated concerns of the Thiokol engineers. However, the Committee finds no evidence to support a suggestion that the outcome would have been any different had they been told.

2. The Committee finds the efforts of Thiokol engineers to postpone the launch commendable; however, Thiokol had numerous opportunities throughout the normal flight readiness process following flight 51-C in January, 1985 to have the new minimum temperature criteria established.

---

[138] Ibid., pp. H1-97.

Discussion

The management of the Shuttle Program has given the responsibility for the Solid Rocket Boosters to the Marshall Space Flight Center. It is the Marshall Center that contracts with Thiokol for the hardware and related services pertaining to the SRBs. The NSTS Program Manager relies on the Marshall management and technical expertise for issues relating to the SRB and it is unreasonable to expect him to take technical advice from the contractor's engineers. This position is supported by the actions taken by Mr. Aldrich and Mr. Moore with regard to the Rockwell concerns over ice.[139] Unlike the SRB situation where the Thiokol managers gave a written positive recommendation for launch, the Rockwell managers refused to give an unqualified go for launch; yet Mr. Aldrich asked for and accepted the recommendations of the Orbiter Project Manager and the Directors of Engineering at JSC and KSC. The Committee finds no evidence to suggest that in the instance of the Thiokol engineers' concerns, either Mr. Aldrich or Mr. Moore would have disregarded the recommendation of the technical managers with the expertise in solid rockets (i.e. Marshall and Thiokol) and relied instead on their own assessment of the engineers' concerns.

Launch commit criteria and launch constraints should be established well in advance of a scheduled mission and should be based on rational, scientific and engineering arguments, including previous flight experience. Thiokol engineers based their arguments for a 53 degree temperature criteria on the fact that this was the coldest temperature experienced to date and they had experienced severe (but not necessarily the worst) erosion on that flight. However, a test firing had been conducted at 40 degrees joint temperature which resulted in no joint problems (technicians had "tamped" the joint putty before the test, however, a procedure not used on flight hardware). Moreover, it was pointed out in the hearing that this flight had occurred a year earlier and no mention had been made of changing the temperature criteria for launch.

> Mr. VOLKMER. But in all of the memorandums, et cetera, that had occurred before—in-between the time, January 1985 and January 1986, you don't specifically say that. . . .
>
> Mr. BOISJOLY. That is right, . . . It was nobody's expectation we would ever experience any cold weather to that degree before we had a chance to fix it again, so that basically is why it wasn't pursued any further than that from my personal standpoint.[140]

That was later questioned by Mr. Nelson in remembering that flight 61-C (the flight just prior to 51-L) had been scrubbed four times for reasons unrelated to temperature when the temperatures were less than 53 degrees during several of those scrubs, reaching down into the low 40s during the first scheduled launch.

---

[139] Ibid., Volume I, pp. 114-17.
[140] Cmte Hgs, Transcript, June 18, 1986, pp. 83-84.

> Mr. NELSON. . . . and so my question is, did any of these same concerns with the temperature come up in discussions during the final checks before those attempted launches?
>
> Mr. MCDONALD. I am not aware that they had, Congressman. I don't know. I wasn't at that launch, but I don't recall that that came up.[141]

Mr. Nelson later asked the Commander of 61-C, Cdr. Robert L. Gibson, whether he recalled any discussion among management or any of the contractors regarding the desirability of launching in 41 degree weather; Commander Gibson also recalled no special concerns regarding temperature.[142]

Mr. Packard also questioned Mr. McDonald about the temperature during earlier attempts to launch 51-L and asked whether in fact it had been below 53 degrees during some of those attempts. Mr. McDonald replied, "That is correct", and when asked whether temperature had been discussed at those times, Mr. McDonald said, "No, it was not . . . Nowhere was it, no." Mr. Packard also asked why, in Mr. McDonald's judgment, temperature had not been discussed in as much as the temperature was below what they believed to be safe, and Mr. McDonald answered, "I don't—I can't answer that." [143]

Mr. Packard also noted the delay in evaluating the effects of temperature, quoting from Mr. Kilminster's testimony, "As launch was scheduled for early the next day, our engineers immediately commenced evaluating the available data." He asked why they waited until the night before the launch to begin even considering the whole question of O-ring resiliency and O-ring problems under cold weather conditions. Mr. Kilminster replied that this was in response to a specific request by NASA.[144]

This indicated that the concerns and recommendations of the Thiokol engineers were solicited by NASA, and in as much as they had not come forth with the recommendation for a higher minimum temperature criterion on earlier occasions when it was planned to launch at temperatures below 53 degrees, it is unlikely that this recommendation would have been made on this occasion without the specific inquiry by NASA.

The Committee finds no evidence that new data were presented during the January 27th teleconference that were not available to Thiokol at the time of the Flight Readiness Review. Moreover, the information presented was substantially the same as that presented at the August 19th briefing (see Section VIII) at which time they had recommended that it was safe to fly as long as the joints were leaked checked to 200 psi, were free from contamination in the seal area and met O-ring squeeze requirements. No mention was made of a temperature constraint at that time or anytime between then and the January 27th teleconference.

The Committee finds that Thiokol's advice and recommendations to NASA were inconsistent, and therefore, the arguments present-

---

[141] Ibid., p. 98.
[142] Ibid., June 25, 1986, p. 78.
[143] Ibid., June 18, 1986, p. 100.
[144] Ibid., p. 101.

ed during the January 27th teleconference might not have been as persuasive at the time as they now appear to be in hindsight.

Issue 5

Do the principal contractors have an appropriate role in the launch decision making process?

Finding

The principal contractors have an active role throughout the decision making process right up to the launch; however, the look of a firm requirement for their concurrence at the time of launch does partially relieve them of responsibility for mission success.

Recommendation

Principal contractors should be required to make a clear, unambiguous statement concerning launch readiness just prior to launch.

Discussion

Participating contractors are required to sign off prior to launch that their flight system or facility is ready to support the flight. This is generally a one-time requirement for a given mission and although they are orally polled prior to the flight, they are not generally required to make any additional written positive commitment for a "go" prior to launch. Mr. Richard Davis, President, Martin Marietta Michoud Aerospace, explained:

> Up to and including the L-minus-one-day review, there's no doubt that every company has a very strong voice; and, as a matter of fact, at the L-minus-one-day review, they are required to stand up and commit their hardware as go or no-go. And those are very unequivocal commitments, also. After that time, then the reviews are more mission management meetings that are held, and as you get down into the countdown, it turns into more of a real time polling of the people that are actually controlling the launch.
>
> In those latter meetings we are not, I would say, formally involved in those unless there is some problem with the hardware itself . . . We are polled by the Director of Engineering prior to the launch actually proceeding, so we are sort of polled in an informal manner. We are not asked at any time after the L-minus-one-day for a formal go or no-go.[145]

Contractors can stop the launch if they have serious reservations about the safety of the mission, and presumably they would.

> Mr. DAVIS. . . . I have never felt that if I needed to stop a launch, I could not stop it. While I have not been asked for a positive go or no-go, the ability is always there if I decide no, to stop the launch.[146]

---

[145] Ibid., July 15, 1986, pp. 71-72.
[146] Ibid., p. 73.

However, the present system permits them to "express concern" without actually saying, "stop the flight, it is unsafe". If the odds favor a successful flight they do not have to be responsible for cancelling, yet if the mission fails they are on record as having warned about potential dangers. (see Section V, discussion over Rockwell concerns over ice)

Issue 6

Are astronauts adequately represented in the decision making process?

Finding

The astronauts believe they currently have the opportunity to make inputs into the process and are reluctant to assume a greater responsibility for the decision to launch.

Discussion

Considerable discussion at the hearing focused on the astronaut's interest in being more involved in the decision making, for example by attending management meetings. Capt. Young made the point that astronauts really didn't have the time to attend a lot of meetings, or the technical expertise to influence the decision.

> We could certainly put people in those kinds of meetings. I am not sure they have the technical expertise to really be able to say go or not go.[147]

With regard to the SRB seals, he pointed out that he and Captain Crippen had attended a briefing at Thiokol where it was stated that the seals weren't even necessary, and some people were complaining about having to put two seals in. And he suggested that if others in the agency had understood the problem they would have stopped the flights.

> The rest of the agency, if they had been aware of this problem, we wouldn't have flown. We would have fixed it. If other people responsible in the management structure had the feeling this was a serious problem, we wouldn't have gone. We have to believe that, because there, on the Orbiter, there are 1500 criticality 1 items on the Orbiter alone, on STS-1, those items are still there, and if the management system can't make sure those things are ready to fly, we can never fly again.
>
> If you have an astronaut saying every step of the way, don't fly because of this, that or this, where they have no expertise, it would be troublesome.[148]

Mr. Lujan asked whether NASA should consider a new class of astronauts with specific technical expertise who would fly occasionally. Capt. Young suggested that this was not a good use of an astronaut's talents.

> You can get real good engineers to do the same thing, a heck of a lot cheaper, and make just as good inputs. . . .

---

[147] Ibid., June 25, 1986, p. 42.
[148] Ibid., p. 44.

> In the main, you like to keep astronauts around to fly spaceships because that is their talent, and that is what they want to do. . . .[149]

General McDivitt concurred:

> There should be a caution about putting too much responsibility on astronauts, when they don't have the time to do it. Like the flight crew commander is very busy prior to flight and does not have time to spend a lot of his time involved in reviewing engineering decisions that have already been made by very professional people. . . .[150]

In response to suggestions that the astronauts might have stopped the launch of 51-L had they been aware of the problems with the seals, Capt. Young provided an excellent analogy to illustrate his skepticism that they would have altered the decision to launch:

> If an engine man comes up and says that engine is ready to fly and the turbine blades are a little cracked but we have run tests and we can show with a cracked turbine blade the engine pumps are not going to come apart and we have got to fly, would an astronaut say no, you are not going to fly until you change the turbines, for example?[151]

There was complete agreement among the astronauts who testified that the crew should be able to make inputs to the decision making process, but they all felt they now have this opportunity; they can and do attend FRRs and other meetings. However, there was a strong feeling among the astronauts that they had to rely on the expertise of the engineers and the technical competence of the managers and could not be expected to intervene in that process. They believed it was unrealistic to expect the crew to make the go or no-go decision; astronauts should not be expected to represent the principal concern for safety.

Major Slayton made the point that astronauts in general were willing to take more risk than management, not less.

> One philosophical point that needs to be brought out here . . . is that the crew commanders and astronauts in general view things a little bit different than everybody else does to begin with and you have to recognize that and be a little bit cautious.
>
> In general a crew commander, if given a choice, is willing to take more risk than his management. That has been the case in the past and he is more likely to give you a 'go' and you need somebody at a higher level that is willing to, on his behalf, willing to take the bull by the horns and have the guts to say 'no go' on behalf of the crew.[152]

Col. Hartsfield concurred:

---

[149] Ibid., p. 49.
[150] Ibid., p. 53.
[151] Ibid., p. 64.
[152] Ibid., p. 54-55.

64-420 O - 86 - 6

> I wanted to say that I feel that it is just like in our own government, the buck stops at the White House or the Congress perhaps, but somewhere, but certainly above the level of the rest of us.
>
> I think that the decison to go or "no go" rightfully belongs with the upper management, and not, my personal opinion, not with the crew. The crew input should be felt very strong.[153]

c. Technical Expertise of Personnel

Issue

Does NASA have an adequate level of in-house technical expertise to manage the Shuttle Program properly?

Findings

1. During the last decade NASA has had significant decreases in manpower. A disproportionate reduction may have occurred in the safety, reliability and quality assurance staff at NASA headquarters and at the Marshall Space Flight Center. Additionally during the period preceding the Challenger accident, the Office of Space Flight also suffered a decline in staff. The decreases may have limited the ability of those offices to perform their review functions.

2. The information presented to NASA headquarters on August 19, 1985 was sufficient to require immediate and concentrated efforts to remedy the joint design flaws. The fact that NASA did not take stronger action to solve this problem indicates that its top technical staff did not fully accept or understand the seriousness of the joint problem.

Recommendations

1. NASA should review the numbers and qualifications of key staff in technical and management positions and should consider additional training and recruitment of individuals to further the quality and safety of NASA's missions.

2. The Committee should maintain on-going oversight of this analysis and conduct an in-depth examination upon the conclusion of NASA's review.

Discussion

In the wake of the Challenger accident, serious questions arose over whether NASA had sufficient technical capability to identify and solve problems like the SRB seal problem. It is argued that through reductions in staffing levels and departures to the private sector by experienced technical employees, NASA lacked in-house problem assessment capability. This is an issue that is not subject to ready answers, and an in-depth examination of NASA technical capacity was generally beyond the scope of the Committee's hearing.

However, it is clear that over the last 15 years NASA has had significant staffing reductions and that a disproportionate number of these reductions may have occurred in the areas of quality assurance and safety.[154] While NASA argues that its personnel levels for these functions "were adequate,"[155] the Rogers Commission found:

---

[153] Ibid., p. 55.
[154] Ibid., June 11, 1986, pp. 59–70.
[155] Ibid., p. 60; material supplied for the record.

> Reductions in safety, reliablility and quality assurance work force at Marshall and NASA Headquarters have seriously limited capability in those vital functions.[156]

Reductions were not limited to the safety and quality assurance program. The former Associate Administrator for Space Flight, Jesse Moore, testified that his office also experienced a decline in the number of staff. As Mr. Moore observed, "we need to . . . get as much technical expertise into the Office of Space Flight as we possibly can" in order to "work on a plane with the real experts—the contractors, the engineers, the safety people at the contractors and at the NASA centers. . ." [157]

Similar views were voiced by former Shuttle program manager Robert Thompson:

> I think we have to look pretty deep in our organization to make sure we are keeping enough technical muscle in the organization to continually search for these pending problems that are somtines pretty subtle. Sometimes they just don't, as I say, announce themselves. So you have to be willing to expend the resources and keep that technical muscle in place and you have to put that technical muscle close to the heart of the issue so that they can perceive a problem if it is just beginning to occur.[158]

It does not necessarily follow however, that reductions in the numbers of technical personnel automatically limit the ability of headquarters to identify and correct emerging problems. The adverse impact flows from those reductions that cut into crucial areas. Accordingly, the Committee is pleased that Admiral Truly has undertaken an examination "throughout the agency and particularly in . . . the Space Shuttle program" to make sure that "we have not only the right numbers but the right kind of trained people . . ." [159] It is hoped that this analysis will identify appropriate technical staffing levels and positions that must be maintained if the agency is to properly perform its function.

NASA technical expertise is further reduced by the departure of highly skilled employees. During fiscal year 1985, approximately 1500 employees left the agency, over one-half of these (784) were engineers, technicians and scientists.[160] If present trends continue,

---

[156] Rogers Commission Report, Volume I, p. 161.

[157] Cmte Hgs, Transcript, July 24, 1986, pp. 80-81.

[158] Ibid., pp. 117-18.

[159] Ibid., June 11, 1986, pp. 61-2. Former Shuttle Program Manager, Robert Thompson, indicated the need for this analysis, stating: ". . . and I think the matter of being sure you have the proper people selected and that those people are properly indoctrinated and trained for their position and they clearly understand the responsibility and reporting channels of their positions, I think those are all areas for improvement."—Cmte Hgs, Transcript, July 24, 1986, p. 121.

[160] Material submitted for the record in response to written questions from Chairman Roe (letter dated 9/18/86). That submission includes the following table:

NASA LOSSES FISCAL YEAR 1981-86

[In fiscal years]

	1981	1982	1983	1984	1985	1986
Total	1,983	1,556	1,176	1,530	1,494	1,169
Non-AST* Engineers	32	26	13	19	9	5
AST ngineers	665	503	386	576	569	457
Life scientists	6	6	0	3	2	4
Technicians	261	207	161	156	204	180

*Non-Aerospace Technologist.

NASA can expect to lose between 7500 and 9000 technical and scientific employees over the next ten years.[161] While 50 percent of these personnel losses are formally attributed to retirement, NASA officials "know. . .that many retires leave NASA for higher paying jobs in industry."[162] Additionally, 17 percent of the departing employees acknowledge that they are leaving NASA for more financially rewarding jobs.[163]

NASA is concerned that the difficulty it will experience in replacing these employees is essentially the same that led to the departures; the agency's "salary structure is not sufficiently flexible and competitive to attract the very best talent our nation has to offer."[164] Therefore, despite liberal hire authority for engineering positions, NASA is experiencing difficulty in recruiting entry-level engineers, largely due to salary. As noted by the Agency:

> Currently the Government pays GS–7 recent college graduates in all engineering disciplines a special salary rate of $23,170. This is the statutory maximum under the current special salary rate provisions. At the same time, our private sector competitors are offering these graduates an average salary of $27,000 to $29,000 depending on the engineering discipline. It would take approximately a 20 percent increase for us to match our competitors. However, absent a legislative change, the most we could offer in the next year would be the percentage increase to the General Schedule (perhaps two or three percent in January 1987).
>
> A continuing infusion of recent college graduates is critical to the continued success of NASA's mission and accomplishing this has become increasingly difficult. Inadequate salaries are an equally significant problem at the executive levels in the agency.[165]

While outside witnesses did not fully concur as to the prevalence of departures for the private sector, all acknowledged the need to create incentives for quality people to enter and remain with the agency.[166] To this end, NASA Administrator Fletcher is examining means by which his organization can retain its highly skilled technical employees through a "more motivational type of organizational structure" and premium pay scheduled.[167] The Committee shares NASA's concern that it maintain a strong in-house technical capability and support staff.

In addition to the number of technical managers, it is also necessary to examine their technical performance. Insight into NASA

---

[161] Ibid.
[162] Ibid.
[163] Ibid.
[164] Ibid.
[165] Ibid.
[166] Cmte Hgs, Transcript, July 24, 1986, pp. 119-23.
[167] Ibid., pp. 123, 125.

headquarters' technical ability to discern and react to emerging problems may be gained from an examination of the manner in which it addressed the growing concerns with the O-rings in the summer of 1985. Prior to that time the problems with the O-rings had been briefed at all levels of the agency and had been presented to headquarters on at least two occasions.[168] However, increasing problems with case-to-case erosion prompted headquarters to request a complete briefing "to go over the situation in detail."[169]

The meeting was chaired by Mr. Moore's deputy for technical matters, L. Michael Weeks, and attended by a number of other headquarters personnel which Mr. Moore characterized as having "some knowledge about the SRB."[170] In testimony before the Rogers Commission, Mr. Moore described the composition of the meeting:

> Mr. Winterhalter, who was Shuttle Propulsion Division Acting Director at that time, Mr. Bill Hamby was the STS program integration Deputy Director, Mr. Paul Wetzel, who was the Solid Rocket Booster programs chief, Mr. Paul Herr, who was the Solid Rocket Motor program manager, and Mr. Henry Quong, who was the reliability, maintainability and quality assurance director of the chief engineer's office.
>
> Those were the group of people at NASA headquarters who attended the meeting. Mr. Mulloy of Marshall Space Flight Center, who was a Solid Rocket Booster program manager, attended and Mr. Bob Swinghammer of Marshall also attended, who is the material and processes laboratory director at Marshall. Thiokol had a total of six people

---

[168] See, e.g., testimony of L. Michael Weeks, Cmte Hgs, Transcript, June 12, 1986, p. 130; Rogers Commission Report, Volume I, pp. 120–140.

[169] Rogers Commission Report, Volume V, p. 1051, testimony of Jesse Moore. A somewhat different version of the genesis of the August 19 briefing was presented by Allen McDonald, Thiokol's Director of Solid Rocket Motor Project, in testimony before the Rogers Commission (Ibid., pp. 1591–92):

Mr. MCDONALD. The meeting that occurred on August 19 came about as a result of this problem with the nozzle eroding through, and that is what drove that meeting. Headquarters wanted to hear about that. We lost the primary seal and eroded some secondary.

We all sat down together and got together with the engineering people and put together that presentation and collectively said, you know, we ought to address the whole seal issue, not just that failure, because we all felt that if that ever happened in the field joint we were in bad trouble because the nozzle has a much better secondary seal than the field joint does.

* * * * * * *

Mr. SUTTER. At this meeting on August 19th at headquarters, that was called because of Thiokol's concern that the joint was really in trouble?

Mr. MCDONALD. No, it was called—we had had another meeting scheduled at Washington headquarters at that time that had a problem with the mixer fire earlier in the year, and there was a review of that.

I believe Mike Weeks either called Joe or I or one of us and said well, you're here, you ought to come and address a couple of other issues that have happened recently that we are very interested in.

One of them is we had broken the structural test article on the filament would case I believe in July down at Marshall, and they wanted to hear about that.

The other one was they were made aware that we had violated the primary seal in the nozzle and wanted to hear about that and what our rationale was to continue.

See also, Cmte Hgs, Transcript, June 17, 1986, pp. 98–101.

[170] Cmte Hgs, Transcript, July 24, 1986, p. 97.

there, including Mr. Mason, Mr. Wiggins, Mr. Kilminster, Mr. McDonald and Mr. Speas.[171]

The briefing documents prepared by Thiokol included a detailed history of seal erosion which noted, *inter alia,* that the "frequency of O-ring damage has increased since incorporation of Randolph putty; higher stabilization pressures in leak test procedures; and high performance motors." The briefing documents also listed MTI's primary concerns; the highest concern was "Field joint—joint deflection and secondary O-ring resiliency."

It is suggested that the August 19th briefing failed to give a complete picture of the seriousness of the O-ring problem because it did not include data on the effect that temperature would have on resiliency of the seals. As Michael Weeks noted in his testimony before the Committee:

> When the briefing was presented to us on August 19th of 1985—as you will look in the briefing that was provided to the Commission on February 10th—there was no temperature data presented that showed that the resiliency was such a critical factor. It wasn't until after the disaster of 51-L that I actually saw the resiliency data that showed that Viton, which is the O-ring material that we've been using, is so slow to recover at very low temperatures—[172]

Mr. Weeks correctly notes that the briefing documents did not include data which resulted from bench testing which concluded that resiliency is a function of temperature.[173]

Other participants in the meeting felt that the temperature issue had been presented at the briefing.

> General KUTYNA. Secondly, there has been some question that people understood that there was a temperature problem. I remember your conclusions chart, your file chart, and the very first bullet of that chart had the word "resiliency" in it.
>
> Do you feel when you talked about resiliency at that meeting people got the connection between resiliency and temperature, that resiliency was a function of temperature, or was that lost?
>
> Mr. MCDONALD. It may have gotten lost because we hadn't run a very long range of temperatures when we got that data.
>
> General KUTYNA. So it is possible that people at headquarters from that briefing did not understand temperature was a concern?
>
> Mr. MCDONALD. I guess it is possible they could have.
>
> General KUTYNA. Is it probable?
>
> Mr. MCDONALD. I don't know if it is probable, because we put it as the first bullet of why we thought that was

---

[171] Rogers Commission Report, Volume V, pp. 1051-52.

[172] Cmte Hgs, Transcript, June 12, 1986, p. 138.

[173] Letter from Brian Russell, Manager MTI SRM Ignition System to James W. Thomas, Marshall Space Flight Center, August 9, 1985.

> our highest concern, and if that hadn't have happened, we wouldn't have had that concern.[174]

The briefing recommended an "accelerated pace" to eliminate SRM seal erosion but concluded that "it is safe to continue flying existing design as long as all joints are leak checked with a 200 psig stabilization pressure, are free of contamination in the seal areas and meet O-ring squeeze requirements." Sometime thereafter, Mr. Weeks reported to Mr. Moore on the briefing, indicating that it was safe to continue the program and that it was not "an issue that ought to ground the fleet."[175]

In evaluating the information presented at the August 19, 1985, briefing, the Rogers Commission found:

> The O-ring erosion history presented to Level I at NASA headquarters in August 1985 was sufficiently detailed to require corrective action prior to the next flight.[176]

The current NASA administrator concurs in the finding.[177]

Despite the clarity of the Commission's conclusions, none of the participants at this meeting (all with technical backgrounds)—NASA or Thiokol—recommended that the Shuttle be grounded until the problem with the seals was solved.[178] Rather, as noted above, the unanimous recommendation was to accelerate the efforts to fix the problem but continue flying. In adopting this course, did NASA take steps to seek a solution that was reasonably commensurate with a threatened failure of a criticality 1 item? Mr.

---

[174] Rogers Commission Report, Volume V, pp. 1595-96.

[175] Ibid., p. 1052. See also, Cmte Hgs, Transcript, June 12, 1986, pp. 143-44, and July 24, 1986, p. 90. A conflict in the testimony arose on the question of the briefing Mr. Weeks provided Mr. Moore following the August 19th meeting. According to the testimony presented by Mr. Weeks, "I briefed on the results of that [meeting] and told him about the briefing and showed him the briefing [documents]." Ibid., June 12, 1986, p. 143. Mr. Moore disagreed with this recitation of the facts (Ibid., July 24, 1986, pp. 89-91):

Mr. SCHEUER. Are you telling us that you didn't receive a briefing from Mr. Weeks and that you didn't receive the briefing documents from Mr. Weeks that was given to headquarters by the Thiokol officials?

Mr. MOORE. To my recollection, the first time I remember seeing that document was on August—was on January 29th or January 30th, right after the Challenger accident. I was shown a document which contained the briefing material. It also subsequently came up in one of the earlier discussions with Chairman Rogers and his Commission is the other time I have seen some of that.

Post-accident was the first time I had, to my knowledge, as I said, seen that particular briefing. I had not sat down and been given a briefing on the Thiokol presentation on August 19th.

* * * * * * *

Mr. Weeks verbally said that the meeting was held that day on August 19th and that in effect that he felt comfortable with the overall conclusions, although he did have one more concern. He felt he wanted to talk to somebody else at Marshall and he did, I believe, talk to Mr. Hardy and said that he thought based on the data and also on the Titan success that in fact there was an acceptable position as far as he was concerned and that is where I left the information and that was the information I was given.

Mr. SCHEUER. He didn't indicate the kind of depth of concern that would have led you to believe that additional time was needed or that additional resources needed to apply to some of these problems before lunch?

Mr. MOORE. No sir. I did not get the feeling that we should have grounded the Shuttle fleet prior to the next flight as a result of that particular briefing.

In a subsequent interview with staff, Mr. Weeks recanted his earlier statement and acknowledged that he did not show Mr. Moore a copy of the briefing document and that to the best of his knowledge Mr. Moore did not see this document until after the Challenger accident. Moreover, Mr. Weeks stated that he did not tell Moore specifically that Morton Thiokol was calling for an accelerated pace to eliminate the seal erosion problem nor did he state that additional resources were needed to be committed to solve the problem.

[176] Rogers Commission Report, Volume I, p. 148.

[177] Cmte Hgs, Transcript, June 12, 1986, p. 129.

[178] Ibid., June 17, 1986, pp. 97-8, 101.

Moore, when asked what he would have done had he received the oral briefing and reviewed the briefing document responded:

> I believe that looking at the document and looking at some of the issues that were cited about criticality 1, flight safety issues and mission success issues that came out in the series of the document there, I believe we would have initiated a formal team to go off and take a much more concentrated look at it.
>
> So I believe my actions would have been to form a team of experts to asess this data and to make recommendations on what our course of action should be at this point in time.[179]

Unfortunately this team of experts was not formed until after the Challenger accident. Rather, NASA proceeded on the course summarized in the following exchange between Chairman Roe and Michael Weeks:

> Mr. ROE. Therefore, there are a group of people—whomever they were—that participated at this particular meeting, reviewed these facts that were available, and they determined two things, according to your testimony. One, they determined that if everything—if they had their "druthers," or whatever the case may be—it would take two years in their judgment to be able to correct that; but in spite of that decision they took and made the second judgement. And the second judgement, well, we can continue to fly. We'll start the mechanisms going to get this corrected, but we can continue to fly until we get that done. Isn't that the decision that was made, according to what you're saying?
>
> Mr. WEEKS. That is correct.
>
> Mr. ROE. Therefore, some people who were at that specific meeting had to be the people who made that specific decision.[180]

In attempting to assess the reasons for NASA Level 1 managers not adopting a more aggressive posture to the O-ring problem, it is suggested that insufficient information was communicated to top.[181] However, as Deputy Acting Administrator Graham observed:

> They could have transmitted the information in a higher profile way, but also as engineers, as managers at headquarters, there was certainly a responsibility to perceive the significance of this.[182]

There was plainly a failure of NASA technical managers, and for that matter those at Thiokol, to grasp the seriousness of the problem. As former Shuttle Program Manager Robert Thompson observed:

---

[179] Ibid., July 24, 1986, pp. 91-2.
[180] Ibid., June 12, 1986, p. 141.
[181] The issue of whether communications are filtered so that important information is prevented from reaching decision-makers is addressed in Section VI.B.2.b.
[182] Cmte Hgs, Transcript, June 17, 1986, p. 207.

> Sometimes these problems are very subtle. Sometimes they stand up and shout louder than at other times. Frankly, this time I think it was standing up and shouting pretty loudly.[183]

Why then did top technical managers in the Office of Space Flight at NASA Headquarters (Level I), Johnson Space Flight Center (Level II), and the Marshall Space Flight Center (Level III) fail to take stronger action? (See VI. A.1.f.) The answer may be simply poor technical decision-making, perhaps in combination with a type of collective rationalization described by Larry Mulloy:

> You asked why wasn't more done. You know, in the six years previous. And I have had that question posted to me many times in the last four months, and I have asked it of myself many times since the tragic accident. And my answer has been in hindsight, obviously, more should have been done.
>
> The turning, I think we started down a road where we had a design deficiency. When we recognized that it had design deficiency, we did not fix it. Then we continued to fly with it, and rationalized why it was safe, and eventually concluded and convinced ourselves that it was an acceptable risk.
>
> That was—when we started down that road, we started down the road to eventually having the inevitable accident. I believe that.[184]

d. Change Control Process

Issue 1

Has the pressure to maintain operational flight rates and schedules for the Shuttle compromised the hardware Change Control Process?

Findings

1. When NASA declared the Space Shuttle to be an operational system, additional pressure to increase flight rates impacted other aspects of the overall program such as the ability to implement, evaluate, test, and certify changes in hardware design.
2. As a result of attempting to operate the Shuttle at increased flight rates, controlling other aspects of the program such as the flight production process and manifest also became a more complex and difficult aspect of program administration.

Recommendations

1. NASA must reconsider its efforts to categorize the Shuttle as an operational transportation system.
2. The Configuration Management System designed to control such changes must be reexamined by NASA as to its effectiveness in assuring that all hardware changes take place in a safe and reliable fashion.

---

[183] Ibid., July 24, 1986, p. 117.
[184] Ibid., June 17, 1986, pp. 215–16.

Discussion

The Rogers Commission noted that, "Following successful completion of the orbital flight test phase of the Shuttle program, the system was declared to be operational."[185] The Commission found that as a result, NASA reduced its safety, reliability and quality assurance activities related to the Shuttle. The Commission report goes on to note that this reasoning was faulty; "The machinery is highly complex, and the requirements are exacting. The Space Shuttle remains a totally new system with little or no history."

Program officials frequently find it necessary to consider changing existing hardware designs or production processes. Such changes can be required for a number of reasons, including: to correct the deficiency in a component; to improve a component's performance or the length of this operating life; to enhance the ease of maintaining the component; or to reduce the cost of manufacturing, servicing, or processing the component. Typically, change proposals originate from a manufacturer and are reviewed by the cognizant NASA field center and frequently by the Level II Program Office at the Johnson Space Center as well. In his review process, NASA compares the cost and schedule impacts of the proposed change against the performance improvement that is anticipated. Of particular concern are the safety aspects related to the change (e.g., What analyses and tests must be conducted to insure that the change does not directly or indirectly have a negative impact on the systems safety or reliability?).

It is clear that these activities or steps in the process of implementing essential changes are complex and time consuming, especially if the components to be evaluated are some of the larger and critical elements of the Space Shuttle. Therefore, it is the Committee's view that until such time as all elements of the Space Transportation System can be fully evaluated through extensive flight testing and trend analyses, it is premature to impose an operational flight schedule on the system in a manner comparable to that imposed upon, for example, an air transportation system.

Issue 2

Is the change control process sufficiently defined for all elements of the Shuttle system?

Findings

1. The NSTS engineering and process change guidelines are, for the most part, sufficiently well-defined for the majority of the subsystems that comprise the Space Shuttle.
2. NASA gives the same level of scrutiny to changes involving a minor component (such as moving velcro strips in the Orbiter) as those involving mission critical elements of flight hardware.

Recommendation

NASA should review its change control process to determine the usefulness of differentiating between minor changes and significant changes.

---

[185] Rogers Commission Report, Volume I, p. 159.

Discussion

NASA's Change Control System is shown in Figure VI–4. From the chart, it is evident that the success of the system is highly dependent on the information flow among the various levels of management control.

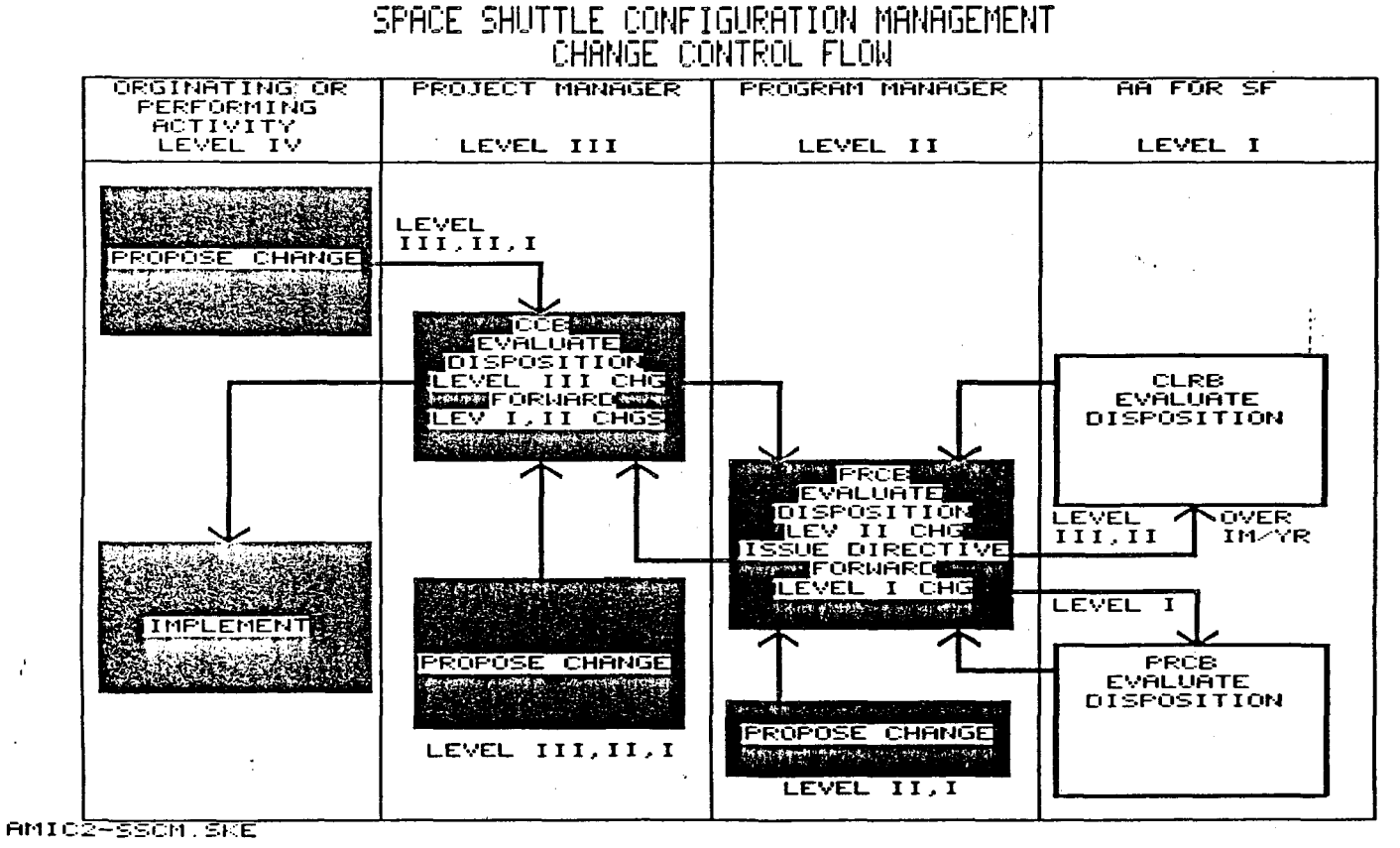

SPACE SHUTTLE CONFIGURATION MANAGEMENT
CHANGE CONTROL FLOW
ORGINATING OR PERFORMING ACTIVITY LEVEL IV
PROJECT MANAGER LEVEL III
PROGRAM MANAGER LEVEL II
AA FOR SF LEVEL I
PROPOSE CHANGE
LEVEL III,II,I
CCB EVALUATE DISPOSITION LEVEL III CHG FORWARD LEV I,II CHGS
CLRB EVALUATE DISPOSITION
PRCB EVALUATE DISPOSITION LEV II CHG ISSUE DIRECTIVE FORWARD LEVEL I CHG
LEVEL III,II
OVER 1M/YR
LEVEL I
IMPLEMENT
PROPOSE CHANGE
LEVEL III,II,I
PRCB EVALUATE DISPOSITION
PROPOSE CHANGE
LEVEL II,I
AMIC2-SSCM.SKE

FIGURE VI-4

The Configuration Management System Requirements are documented in JSC 07700 Volume 4, entitled "Configuration Management Requirements," dated March 2, 1973. Changes to this document have periodically been issued over the course of the program. The configuration management system defines requirements for all levels of management within the NSTS program. A baseline set of requirements is defined for each level of management (Level I through Level IV). This baseline establishes what is to be accomplished at each level of management and established the controlling procedures that supposedly prevent deviations from the baseline program. This baseline program is specified for each flight and includes specifications on payloads for each flight as well.

Changes to the flight and system requirements and the acceptance baselines are made, according to NASA, only by directives issued by the Program Requirements Control Board at Level I and Level II and the Change Control Boards. For example, there is an Orbiter Avionics Software Control Board (OASCB) that has joint Level II and Level III authority for managing the program-wide requirements for Shuttle computer hardware and software systems as part of the Orbiter project. The Board also assures the correct configuration of the software within the Orbiter avionics system for all vehicle and test operations.

Design changes at the contractor level are processed through several levels of technical and managerial reviews. Design and engineering changes on the Orbiter, for example, undergo Technical Status Reviews (TSR's), Avionics Status Reviews (ASR's), Preliminary Design Reviews (PDR's), Critical Design Reviews, (CDR's), Design Certification Reviews (DCR's), and numerous special meetings of NASA and the Rockwell management are utilized to review issues and concerns about any design drawing or specification. According to Rockwell,[186] "Changes are reviewed at a TSR or ASR and the Change Control Board for approval. Any outstanding design dispute is tracked as an open action until it is resolved by Rockwell and NASA management."

The Committee questions, however, whether the complex and extensive processes involved in NASA's change control management system allow for sufficient distinction between minor changes and the significant changes. For example, the systems requires the same level of management attention to as minor a change as moving velcro strips on the Orbiter as it is applied to all Criticality 1 item such as changing a turbo-pump on the SSME.

2. ORGANIZATION AND POLICY MANAGEMENT

a. Management Structure

Issue 1

Does the management of the Shuttle Program adequately define the lines of authority and are managers given authority commensurate with their responsibilities?

---

[186] Responses to Committee Questions, dated August 22, 1986.

Finding

The management of the Shuttle Program is complex and diversified and it is not always clear who has authority or responsibility. NASA's "lead center" concept has resulted in placing the management of the program at JSC, one of three centers participating in the program; however, because Johnson does not have control of the other centers' resources, the NSTS program manager's authority to manage the program is limited and the responsibility is unclear.

Recommendation

NASA should restructure the Shuttle Program management to define clear lines of authority and responsibilities. This restructuring should take into account the special role each center must play and be especially sensitive to the need for the cooperation and support of all the participants to achieve a common goal. NASA should give special consideration to moving the Program Manager to NASA Headquarters to avoid the confusion and inter-center rivalry that result from having a large multi-center program managed out of one of the participating centers.

Discussion

The line of management responsibility, authority, and accountability for NASA programs is from the Administrator to the Associate Administrator to the Field Center Directors, who delegate implementation authority to a program/project manager. JSC has been designated the "lead" center for the NSTS Program and has the responsibility for systems engineering and integration, operations integration, and management integration. Marshall has the responsibility for the propulsion system and Kennedy has the responsibility for launch operations.

The Associate Administrator for Space Flight (the Level I program manager) performs oversight over the program but doesn't have the technical staff to effectively manage the program. The NSTS Program Manager, i.e. the Level II manager at JSC, functions as a program coordinator; he is responsible for integrating the various program elements and he controls all the project interfaces. He clearly does not control all the program elements since the individual (Level III) Project Managers are accountable to their Center Directors who are in turn accountable to the Associate Administrator who controls the funding. For example, the Level II manager told the Rogers Commission that he was unaware that the SRB Project Office had procured additional Solid Rocket Motor casings to be used for testing of the joints;

> Now it turns out that the budget for that kind of work does not come through my level II office. It is worked directly between the Marshall Center and NASA Headquarters and there again had I been responsible for the budget for that sort of work, it would have to come through me, ...[187]

[187] Rogers Commission Report, Volume V, p. 1490.

The witnesses who addressed the management issues at the Committee hearings had differing philosophies regarding the best possible solution; however there was general agreement that the present system tended to cause confusion. There was also strong sentiment for strengthening the headquarter's role. Mr. Jesse Moore, former Associate Administrator for Space Flight, testified:

> I think we need to go back and make sure we clearly define the roles of NASA headquarters, the roles of the centers in the overall management of the STS.
>
> I think we need to re-look at that kind of interaction and the kind of specific roles, responsibilities, to ensure that authority and responsibility is commensurate in terms of the role definitions for the varous levels of management in NASA.
>
> I think we need to look at strengthening NASA headquarters. I would say that in my tenure a NASA headquarters we had a decline in staff in the Office of Space Flight. It was a decline in the number of staff, and I think we need to look at what is the proper level of staffing requirements to do this particular job.
>
> I also think we need to look to make sure we get as much technical expertise into the Office of Space Flight as we possibly can.[188]

General Stafford, a former Gemini and Apollo astronaut, also stated: ". . . I guess I was never comfortable with the lead center type of management structure, after having seen how satisfactorily Apollo worked." [189] (Note: The Apollo program was managed out of headquarters.)

With regard to the appropriate role of the Program Manager (Level II), there was not a clear consensus. In discussing the Rogers Commission's recommendations, General Abrahamson stated:

> . . . However, I would also like to point out that many of these recommendations have long been incorporated in NASA management procedures. The Program Manager, by definition, has the necessary authority to get the job done.[190]

When asked if the program management should remain at JSC, Mr. Moore, who is currently Director of JSC, replied:

> I think that is certainly a topic that is going to be studied very, very carefully.
>
> I think there are a couple of options that can be looked at that would keep the major parts of program management that has been in operation at the Johnson Space Center at the Johnson Center.
>
> There are a lot of tools, roots and capabilities. I think, on the other hand, there should be some looks at the Office of Space Flight for finding some way to strengthen

---

[188] Cmte Hgs, Transcript, July 24, 1986, pp. 80, 87.
[189] Ibid., June 25, 1986, p. 128.
[190] Ibid., July 24, 1986, p. 11.

the overall program management in the Office of Space Flight.

And one concept might be to have a Shuttle Program Director within the Office of Space Flight and working with the Level II program office at the Johnson Space Center.

My answer is, I believe the Level II program office, with some strengthening, and the level I program office, with some strengthening—we can make it work and it should remain at the Johnson Space Center.[191]

This was in direct contrast to the view held by John Yardley, former Associate Administrator for Space Flight. In discussing the Rogers Commission's recommendations, he stated:

> The one in particular that I think I have some background in that I think is not correct is they are trying to strengthen the authority and responsibility of the Program Manager at Johnson. Let me just relate what happened when I went to NASA.
>
> I hadn't been there but a couple of weeks and one of the other centers called me and said, "Hey, the Program Manager wants to take 15 million of my money and put it on the Orbiter." It became immediately apparent to me to have one of the center people handle the funding decisions was not going to be in the best interests of cooperative technical activity.
>
> So I pulled the final decisions on the money to Washington, where I think they still are . . .[192]

Major Slayton made a similar observation concerning the problems with having a multi-center program managed at one of the field centers:

> . . . I think when you look at relationships between the centers and how the organization is structured; and you could say it could be restructured so you don't have inter-center jealousies interfering with the communications channel.
>
> A lead center concept where Level II is viewed by the other centers as being another center instead of having its headquarters' level is one reference I would make.[193]

Major Slayton went on to say that any organization could work with the proper people:

> A lot of it is in the management attitude; but again, my opinion is, you can make any organization work if you got the right people, and if you don't have the right people I don't care how you organize it, it will not work, so you still end up dealing with individuals.[194]

---

[191] Ibid., p. 99.
[192] Ibid., July 23, 1986, p. 10.
[193] Ibid., June 25, 1986, p. 101.
[194] Ibid.

Other witnesses also alluded to the problems with inter-center rivalries under the current system and the break down of esprit de corps. Mr. Moore called for a new effort to re-instill the team spirit:

> I believe an approach to that has got to be building team work, again, to make sure—the Shuttle program involves many elements, many contractors, many NASA centers, all playing together as a team.
>
> I believe we have to go back and re-instill in our people, in our participants, a team work approach . . .
>
> I think the overall structure of the Shuttle program is obviously built upon people and, you know, there are humans all the way up the chain, all the way from the engineers at the subcontractors to the engineers at the contractors, the NASA centers and so forth.
>
> I think we have got to make sure that each of those participants in the program feel a dedication, feel a dedication to safety, feel a dedication to the program that they are making a valuable contribution and I think we need to do that by personal communications as well as trying to look at our structure to make sure we have not defined something that will at least maybe encourage, tend to encourage communications breakdown.[195]

General Abrahamson made the following observations with regard to changing organization:

> It is true that when any organization is formed, it is formed to help you accomplish a particular task. By the same token, once it is there, it develops momentum and procedures and impediments sometimes to exactly what you would like to have, a dynamic and modifying organization for the challenges of the future. This is always difficult.
>
> I believe that we had an organization that was designed for the development of the Shuttle, and when we got there, since it was only the second flight, that we had a tremendous change of attitude that we had to be able to create, and that was to create an organization that would think in terms of operations of the Shuttle and overcome the flight test problems.[196]

Issue 2

Are astronauts adequately represented in management?

Finding

The Committee finds no evidence that astronauts are denied the opportunity to enter management if they so choose.

Discussion

The Rogers Commission has suggested that NASA should make greater use of astronauts in management; however, the Commis-

---

[195] Ibid., July 24, 1986, pp. 69-70.
[196] Ibid., p. 17.

sion report provides no basis for that recommendation. Astronauts generally have shown little interest in going into any kind of desk job, including management positions, until such time as their active flying days end. At that time, management jobs within the astronaut program become attractive alternatives to some; however, opportunities in this area are naturally limited. Major Slayton expressed this very well when he testified about his experience with the Mercury Program:

> I had the misfortune at that time of having been grounded due to a medical problem so I was elected to take over the management of the astronaut corps, a job I didn't particularly care about, but it was the next best thing.[197]

Mr. Nelson asked the astronauts whether any of them felt there was a "modus operandi" within NASA that excluded either active or former astronauts from the management structure. General McDivitt stated he had seen no bias in his three years as Program manager for the Apollo Program. Mr. Nelson then asked Deke Slayton if he had ever seen any bias in NASA and Major Slayton confirmed that he too saw no evidence of bias against astronauts in management. General Abrahamson observed:

> Throughout my tenure, astronauts were in key program office positions and one served as an Assistant Associate Administrator in the Office of Space Flight. . . .[198]

There was agreement among the astronauts that the astronaut office should be moved up higher in the organization. General McDivitt summed up the astronauts' position:

> I think I would recommend that the Flight Crew Operations Directorate be moved up to report to the Center Director as well as the Flight Operations Director.
>
> I think both of those organizations are very key to flying, and having them go through another layer of management before they get to the Center Director creates a filter which is not necessary or desirable for either one of them.
>
> I think it also gets them on the same level as the engineering organizations within the manned spacecraft center, and gives them better access to the program.[199]

b. Communication

Issue 1

Are there adequate opportunities to communicate problems within the Shuttle Program management structure?

Finding

There are many regularly scheduled meetings and teleconferences at all levels of management throughout the Shuttle Program. In addition, "special" meetings and telecons are routine. No evi-

---

[197] Ibid., June 25, 1986, p. 11.
[198] Ibid., July 24, 1986, p. 11.
[199] Ibid., June 25, 1986, p. 52.

dence was found to support a conclusion that the system inhibited communication or that it was difficult to surface problems.

Discussion

Every day at noon central time a teleconference is held among all NASA Space Shuttle Program participants. This is the daily "special" Level II PRCB (Program Requirements Change Board) meeting and includes, among others, all the managers of the various program elements, the JSC Directors of Flight Crew Operations, Mission Operations, Engineering, Mission Support, SR&QA, and Space and Life Sciences. Program status, urgent problems, and program requirements are brought up at this meeting. The PRCB convenes by teleconference on alternate Fridays to discuss all other (less urgent) program issues; in addition, other special meetings are called by the PRCB secretary when deemed necessary.

Each of the supporting organizations also has regularly scheduled meetings, often by teleconference when they involve more than one location. Regularly scheduled (often daily) teleconferences are also held between various directors and managers.

Level I at headquarters conducts daily status meetings and also participates in the noon teleconference. These meetings plus all the Flight Readiness Reviews provide ample opportunity to surface problems.

Issue 2

Is too much information being disseminated so that important information is lost?

Finding

Large amounts of information are disseminated on a routine basis, often with little or no indication of its importance to all of the recipients.

Recommendation

NASA management should review the process of providing information on significant actions so that awareness by concerned managers is assured.

Discussion

In a NASA briefing to staff on Mission Operations (May 21, 1986), NASA managers revealed that they routinely received information copies of all sorts of memoranda, such as directives, requests, approvals for changes, etc. Often the individual receiving these copies had no direct involvement with the specific subject of the memoranda, and they acknowledged that it was entirely likely that an important piece of information could cross their desk without their awareness.

Issue 3

Are communications filtered so that important information is prevented from reaching the decision makers?

Finding

NASA managers delegated the responsibility for making technical judgments to lower level managers or assistants. Therefore, the information that reached the top decision makers was "filtered" in that it was interpreted by others that were presumed to have more specialized experience or expertise in a given area. There is no evidence that middle level managers suppressed information that they themselves deemed to be significant. In fact, as discussed in the Section on Technical Expertise, the failure was not the problem of technical communications, but rather a failure of technical decisionmaking.

Discussion

It is typical in any large, complex organization that as managers rise higher in the organization the scope of their responsibilities broadens to encompass technical areas beyond their own specialized expertise. Therefore they must rely increasingly on the technical judgments of lower level managers or assistants. There is the additional risk of subordinates' reluctance to transmit unpleasant information upwards; however, it is not evident that NASA managers suppressed information about problems they themselves understood.

Throughout the hearings, witnesses said that had they known about the seriousness of the problem with the SRM joint, they would have stopped the flights; or (in their opinion), had the decision makers known about it the flights would have stopped. The witnesses acknowledged that the problems with the SRM joint had been briefed at all levels, but always in a way that didn't communicate the seriousness of the problem; it was not viewed as life-threatening. Yet the witnesses appeared reluctant to attribute this to poor technical judgments on the part of the managers or technical staff with expertise in propulsion, preferring instead to blame it on poor communications or a poor "decision-making process."

Mr. Scheuer questioned Jesse Moore specifically on this point when he asked, referring to Mr. Weeks' summary of the August 19 meeting, "Was it a failure of decision-making on his part or communications on his part?" Mr. Moore responded:

> Sir, I think that in a position like Mr. Weeks is in, we have to work as a team, for example, and people have to make assessments on situations and I think Mr. Weeks looked at the data and his assessment was that he thought we had a program adequate to cover the activities in the SRB and he believed that after he had talked to the people at Thiokol and he also believed that, I think, after talking to the people at Marshall and I believe his position was that in fact was an acceptable posture for him to take. Part of his responsibility is to make technical judgments.[200]

Mr. Moore went on to explain that he believed the lack of understanding of the SRB joint extended throughout the agency:

[200] Ibid., July 24, 1986, p. 94.

> I would say, sir, in looking up and down the system and what has been determined about the SRB from the many analyses and work that has been done in the past, I don't think the system all the way from day one of the program really understood all the implications of how the SRB joints worked and I think that we have learned, all of us have learned, an awful lot about the SRB . . .[201]

And again referring to the August 19 meeting:

> That was a report from my deputy (Mr. Weeks), that he believed the situation was acceptable as far as assessment of the data presented to him, and I trust the people in the organization to make those kinds of judgments.
>
> We have to make those judgments on a day-to-day kind of basis, but I did hear at Flight Readiness Reviews, as everybody as a member of the overall Shuttle team heard about issues associated with the O-ring problem. I believe the first time this was experienced on the Shuttle Program was all the way back to flight 2 . . . I did not, as the head of the Level I office, believe the problem with the SRB O-rings was serious enough to consider stopping the launches.
>
> If I did, I would have stopped the launches, sir.[202]

Mr. Scheuer again asked Mr. Moore to identify where the failure was, "Was it in your being communicated with by Mr. Weeks? Was it a failure of judgment on Mr. Weeks' part that all systems were go? Where was the failure?" At that point, Mr. Moore blamed the failure on communications:

> I think in looking at the whole situation, I think there was a failure to communicate the technical seriousness from the contractors involved in this program through . . . .[203]

But, Mr. Scheuer suggested that the contractors had communicated the problem at the August 19th meeting. Mr. Moore then suggested that perhaps someone should have made a stronger statement; however, in their collective judgment it was not a serious problem:

> On the basis of the specific August 19 briefing that was presented, I believe there should have been a stronger statement made to me that we have a much more serious problem by Mr. Weeks or any of the people who attended that briefing. Mr. Weeks was not the only one at the briefing. There were others at the briefing who had some knowledge about the SRB . . . .
>
> I don't recall the specific list of attendees at that particular meeting, but people that were in the overall propulsion area of the office of space flight—and the office of space flight is level one—that is the level one—people who had experience in this thing.

---

[201] Ibid., p. 95.
[202] Ibid.
[203] Ibid, pp. 96–97.

> I believe if they felt after that August 19th briefing that we had a problem, that the system should be grounded, that somebody would have come and said, "We have got a problem serious enough to ground the Shuttle flight."
>
> That did not occur, and I believe it was based on a collective set of judgments that we did not believe the problem was as serious.[204]

The Committee finds no reason to doubt Mr. Moore's observations that no one within NASA understood the problem with the O-ring and accepts his conclusions:

> In hindsight, I think we should have taken much stronger action after the August 19th briefing . . . if I had the knowledge then that I have today, we would have grounded the fleet.
>
> I did not have it at the time.[205]

In hindsight, the August 19th briefing, as well as the January 27th telephone conversation clearly identified a serious problem. Perhaps the Thiokol engineers understood the seriousness of the problem; however, Thiokol's own summary and recommendation at the conclusion of the August 19th briefing stated:

> Analysis of existing data indicates that it is safe to continue flying existing design as long as all joints are leak checked with a 200 psig stabilization pressure, are free of contamination in the seal areas and meet O-ring squeeze requirements.[206]

This conclusion was accepted by all who heard the briefing, and this was the information that was transmitted throughout NASA. The evidence does not support a conclusion that the top decision makers would have arrived at a different conclusion from the managers at Marshall and the Level I managers with propulsion backgrounds. (For additional discussion on this issue, see Section VI.B.1.c.)

c. Safety, Reliability and Quality Assurance

Issue 1

Is NASA's decision to establish a new Office of Safety, Reliability, and Quality Assurance appropriate and, if so, what should its role be?

Finding

The Committee finds that the Rogers Commission recommendation that NASA should establish an Office of Safety, Reliability and Quality Assurance that reports directly to the Administrator is indeed appropriate. However it is not clear what the activities of this office will encompass.

---

[204] Ibid, p. 98.
[205] Ibid.
[206] Rogers Commission Report., Volume II, p. H-95.

Recommendations

1. The Associate Administrator for Safety, Reliability and Quality Assurance (SR&QA) should provide to the Committee the agency's draft plan delineating the organization, goals, implementation strategies and resource requirements of the office of SR&QA.

2. After the Office of SR&QA is fully operational, the Committee will wish to continue oversight over its activities.

Discussion

Chapter 7 of the Rogers Commission report deals with the subject entitled "The Silent Safety Program." The Commission identified shortcomings in NASA's overall Safety, Reliability and Quality Assurance Programs, and recommended the formation of a separate Office of Safety, Reliability and Quality Assurance that would report directly to the Administrator. The role of safety and quality assurance in the decisionmaking processes associated with Shuttle flight production requirements has been relatively undefined and ambiguous. The formation of a centralized coordination and control organization should serve to remedy the situation. As the Rogers Commission report notes, ". . . No one thought to invite a safety representative or a reliability and quality assurance engineer to the January 27, 1986 teleconference between Marshall and Thiokol." [207]

On July 8, 1986, the Administrator established the position of Associate Administrator for Safety, Reliability and Quality Assurance, and briefly delineated the responsibilities of this office in NASA's responsive document to the Rogers Commission report.[208]

According to NASA, the purpose of this office is to strengthen the role of the SR&QA functions across all the the NASA programs. This will be accomplished by establishing centralized coordination under the Associate Administrator for SR&QA who reports directly to the Administrator on all pertinent matters related to the NSTS. The Associate Administrator is chartered to examine the adequacy of the agencies resources in these areas and to make recommendations for improvements as appropriate. Functional organizations that were previously under the purview of the Chief Engineer's office will now report directly to the Associate Administrator for SR&QA.

The major contractors to the NSTS agree with the Commission's recommendation to form a separate NASA SR&QA organization reporting directly to the Administrator. They are, however, of the opinion that responsibility for the work required to recommend or implement changes or modifications in the quality assurance area must remain with Level III and the contractors themselves.

The Committee does not argue with the contention that strong SR&QA capabilities must reside at the contractors' plants. Further, the Committee suports NASA's efforts to enhance its in-house capabilities in order to improve the agency's monitoring and oversight capabilities in the areas of SR&QA. Strengthening Headquarter's ability to provide guidance and centralized coordination in the

---

[207] Ibid., Volume I, p. 152.
[208] NASA Response to Rogers Commission, July 14, 1986, p. 20.

areas of configuration management, product reliability and quality assurance and risk management, are essential to returnng the Shuttle to flight readiness condition.

Issue 2

Has NASA applied sufficient resources to support adequate SR&QA efforts within the NSTS program?

Findings

1. The Committee finds that reductions in NASA civil service personnel that have occurred over the past decade have adversely impacted the agency's ability to maintain the appropriate level of oversight control of the Safety, Reliability and Quality Assurance activities within the NSTS.

2. NASA has become increasingly dependent upon outside SR&QA support from the Department of Defense (Defense Contract Administration Services [DCAS] and Air Force Plant Representative Office [AFPRO]) and contractors.

3. NASA has reduced or reassigned to other program areas in-house safety, reliability and quality assurance tasks such as testing, analyses and instrumentation and has reduced or shut down in-house facilities for performing SR&QA research and technology development. The degree to which these factors have adversely impacted the safety, reliability and quality assurance activities within the NSTS program has not been adequately assessed.

Recommendations

1. NASA should establish and maintain a strong and effective SR&QA Program. Continuing support for such a program must come directly from the Administrator.

2. Although it is appropriate to establish strong contractor capabilitie in the areas of SR&QA the internal oversight responsibilities and coordination of SR&QA tasks must be the responsibility of NASA itself. In order to assure that the appropriate interfaces among the various subsystem elements that comprise the NSTS, are maintained, a sufficient complement of NASA SR&QA management and support staff must be available to perform the necessary oversight and coordination tasks.

Discussion

Reductions in force over the past several years have reduced personnel across the agency from a complement of some thirty-six thousand people down to twenty-two thousand people. A disproportionate decline in Reliability and Quality Assurance (R&QA) staffing occured as a result of these reductions. In the Shuttle program, many of the quality control functions and government inspection activities have been performed by contractors in conjunction with the Department of Defense support personnel (DCAS and AFPRO). NASA has expressed some concern about their ability to maintain adequate in-house staffing in these areas. The total number of civil servant employees within NASA dedicated to the SR&QA program

is presently about 500 professionals.[209] This represents a reduction of 71% from the 1970 complement.

NASA attributes this reduction to the termination of "in-house flight programs, along with the transfer of certain functions . . . to other organizations within the NASA centers." In their response to Mr. Roe's inquiry,[210] NASA makes the following statement:

> "Even though we had a reduction in R&QA personnel, our detailed review of the quality operation did not reveal that we missed any of the quality control check points which may be contributed to the accident."

The Committee cannot support NASA's assessment on this matter. Although NASA may argue that the quality control check points for the certification tests required on the ambient and induced temperature effects on the O-ring seals were checked off by the QA representative at Thiokol as having been satisfactorily completed, in actuality these tests were never performed. To what extent this failure of the QA function to do its job conbributed to the accident may be questioned, but the fact that the control didn't work in this case cannot be denied.

It should be noted, however, that according to some of the prime contractors, SR&QA staffing has actually improved over the years.[211] For example, at the Rocketdyne Corporation, there has been an increase in QA staffing to a level that represents nearly 40% of the corporation's manufacturing staff.

Issue 3

Are the responsibilities of safety engineers and design engineers adequately specified within NASA's "risk management" program?

Finding

The roles of safety, design as well as reliability engineers are not adequately and uniformly defined throughout the NSTS program. In some cases, the Committee learned that safety engineers were not participating in major decisions related to flights of the Shuttle.

Recommendations

It should be the responsibility of the new Associate Administrator for SR&QA to fully specify the roles of safety and reliability engineering as well as quality assurance personnel within the NSTS program so that all critical aspects of the program and decisions related to the adequacy of hardware and subsystem performance are fully reviewed by these disciplines.

Discussion

The function of the safety engineers within the NSTS program has been to determine whether or not certain prescribed tests, analyses, and design descriptions have been followed appropriately

---

[209] Discussion with the NASA Chief Engineer's Office, May 13, 1986. Also, Cmte Hgs, response to question by Mr. Roe, Transcript, June 11, 1986, pp. 59–60.

[210] Cmte Hgs, response to question by Mr. Roe, Transcript, June 11, 1986, pp. 59–60.

[211] Telephone Conversation, August 13, 1986.

as they relate to safety concerns, using the techniques of HA. The safety office has not been significantly involved in the engineering design efforts. If an engineering problem arises that could effect the safety of the overall system, it is the responsibility of design engineering teams to perform technical evaluations rather than having these analyses performed by the safety engineers. Prior to the Challenger accident, the safety program did not have the personnel, facilities or expertise to review decisions by design engineers that the O-ring erosion problem was a manageable risk. Even though this erosion was a continuing problem, there was, according to testimony provided to the Rogers Commission, no second set of "eyes" available to question waiver applied to this problem.[212]

Issue 4

Does the SR&QA program require improved coordination between centers, contractors and NASA Headquarters?

Findings

1. Although guidelines have been published that describe the responsibility of contractors' in the areas of SR&QA,[213] NASA's guidelines do not adequately distinguish these various activities as distinct disciplines requiring specialized skills and centralized coordination.
2. In its review of the agency's reliability and quality assurance programs as they relate to the Space Shuttle, the Committee found there was little commonality among the cognizant officials at MSFC, JSC, KSC, and Headquarters in the perception of the various responsibilities associated with these separate and distinct disciplines.

Recommendations

1. It is important that a clear delineation of responsibilities for the separate SR&QA disciplines be appropriately documented. It is also essential that the relative importance of each of the three separate disciplines be established as an integral part of the NSTS program. These functions are the responsibility of NASA Headquarters.
2. NASA must carefully review the staff and resources devoted to the SR&QA function within NASA and contractor organizations for adequacy. The Administrator shall report to the Committee with his findings and recommendations.

Discussion

Although the controlling document describing the SR&QA functions for the Shuttle contractors was provided to the Committee, no corresponding document was identified that describes the implementation of these functions for the SR&QA engineers that are direct employees of NASA. NASA contends that the same controlling document applies to agency employees. The specific oversight

---

[212] Testimony before the Rogers Commission, Mr. Jack Walker, Deputy Director, MSFC Safety Office, April, 1986.
[213] NASA, "Safety, Reliability, Maintainability and Quality Provisions for the Space Shuttle Program": NHB 5300.4 (1D–2), October, 1979

responsibilities of these employees and their independent reviews and analyses requires a more complete delineation in the Committee's view. The Rogers Commission report provides definitions for the SR&QA disciplines.[214] An expansion upon these definitions is required in order to establish a commonality of understanding of the various functions as they apply to the Shuttle program.

The management structure within NASA that coordinates and performs the activities associated with the SR&QA tasks for the NSTS has become decentralized over the past decade. Until recently many of the oversight duties that at one time were handled through Level I were moved to the field centers. Responsibilities for various systems that comprise the Space Shuttle are delegated to the Level III field centers. These centers establish and coordinate SR&QA activities at the contractor facilities. They are also responsible for reporting any anomalies, inconsistencies, or problems to Level II program management.

Until recently, the Office of the Chief Engineer had responsibility for SR&QA activities. For various reasons, the operations of this office in the areas of SR&QA appear to have lost effectiveness, either through reductions of personnel and support of these programs at the Headquarters level or through the diffusion of these functions into various organizations within the operating divisions at the field centers.[215] These changes reduced Headquarter's ability to participate in field center status reviews with the prime contractors, limited the Level I manager's ability to survey the effectiveness of the SR&QA programs agency-wide and reduced the colocation of SR&QA personnel within Headquarter's program offices. The Committee expects that the new Office of SR&QA will be chartered to make appropriate corrections to augment the safety, reliability and quality assurance functions within the NSTS Program.

d. Contractor Incentives

Issue

Key Shuttle contracts (e.g., the Solid Rocket Booster Production Contract and the Shuttle Processing Contract (SPC)) provide incentives both for reliability, integrity, and safety of products and services on the one hand, and for cost and schedule on the other. Do these contracts provide an appropriate balance between the two types of incentives? That is, does NASA utilize contracts to reward and promote operational safety?

Findings

1. The SPC provides far greater incentives to the contractor for minimizing costs and meeting schedules than for features related to safety and performance. SPC is a cost-plus, incentive/award fee contract. The amount of the incentive fee is based on contract costs (lower costs yields a larger incentive fee) and on safe and successful launch and recovery of the Orbiter. The award fee is designed to permit NASA to focus on those areas of concern which are not sen-

[214] Rogers Commission Report, Volume I, p. 152.
[215] Discussion with the NASA Chief Engineer's Office, May 13, 1986.

sitive to the incentive fee provisions, including the safety record of the contractor. However, the incentive fee dwarfs the award fee—while the maximum value of the award fee is only one percent of the value of the SPC, the incentive fee could total as much as 14 percent of the SPC.

2. During the developmental phases of the Thiokol contract for Solid Rocket Booster production (1980–1983), the contractor received consistent ratings of "Excellent-Plus" or "Superior" under the cost-plus, award-fee contract. NASA contracted with Thiokol on a cost-plus, incentive-fee (CPIF) basis beginning in July, 1983. The CPIF contract pays strictly on the basis of costs, although penalties may be invoked for delays in delivery or for Shuttle accidents due to SRB failure. At the time of the Challenger accident, Thiokol was eligible to receive a very large incentive fee, probably on the order of $75 million.

Recommendations

1. NASA should reexamine all Shuttle contracts and report to the Committee with its findings and recommendations on whether more incentives for safety and quality can be built into these contracts. This report should address, *inter alia,* the SRB Production Contract and the SPC.

2. NASA's new Office of SR&QA should be involved in the procurement and award fee processes, both to establish reasonable guidelines and rewards in new contract and to judge performance of ongoing contracts.

Discussion

Mr. Robert Thompson, Vice President of McDonnell Douglas, summarized the position of several Committee witnesses when he stated:

> I have never detected that a contractor would deliberately infringe on safety for a profit motive.[216]

On the other hand, Thompson also admitted that contracts do vary in the extent of their safety incentives and that, to a certain degree, such incentives can make a difference in operational safety:

> . . . the type of safety that we are looking for, for a system like the Shuttle, I think they can be enhanced with these kind of stipulations in a contract. They can't truly be bought that way.
>
> Certainly you [could] hang a larger incentive toward safety. You may enhance a strong focus on safety and I would not say that it wouldn't do some good to enlarge those enhancements.[217]

The more difficult question is whether existing NASA contracts, such as the SPC and the SRB Production Contact, strike an appropriate balance between safety incentives and cost/schedule incentives. This question is particularly critical in light of reductions in

[216] Cmte Hgs, Transcript, July 24, 1986, p. 135.
[217] Ibid.

NASA's SR & QA programs detailed in Section VI.B.2.c. of this report.

Both Thiokol and NASA witnesses on June 17, 1986, argued that the penalties inherent in the Thiokol contract with Marshall Space Flight Center provided more than adequate incentives for Thiokol to deliver safe, reliable products. These penalties are of two types. Late-delivery penalties amount to $100–200 thousand per unit. Penalties for mission failures are much larger:

> If findings of this Board of Investigation determines (sic) that the cause of the failure is attributed to the Solid Rocket Motor/Motors not performing in compliance with the specification requirements of the contract, a fee reduction of $10,000,000 for each category I failure and $5,000,000 for each category II failure shall be deducted from any fee otherwise earned under this contract.[218]

Similarly, in briefings for Committee staff, NASA contract managers have stated that the award fee portion of the SPC, though small, is highly visible and that contractors take the award fee and the semiannual contract ratings very seriously. In NASA's view, high ratings enhance a company's reputation and, therefore, its likelihood of competing effectively for additional contracts.

Nevertheless, there are several reasons to believe that NASA could utilize contractual terms more effectively to enhance program safety. First, there can be no argument, for both the SPC and the SRB contract, that absent a major mission failure, virtually all the financial incentives are tilted toward cost-savings and timely delivery.

Secondly, because of the complex and overlapping division of responsibilities between NASA and its contractors, it is not clear that contractors will be fully penalized even in cases where their actions or their hardware appear to be directly responsible for a mission failure. Mr. Scheuer's questioning of Mr. Charles Locke, Chairman of the Board of Morton Thiokol, showed that Thiokol is not prepared to accept the full contractual penalties for the Challenger accident.[219]

Finally, it is revealing that, under its NASA contract, Thiokol was never penalized for any of the numerous SRB flight anomalies.[220] The booster joint had never worked as intended, nor was its behavior at ignition ever clearly understood. Occurrences of O-ring erosion and/or blow-by exceeded twenty-five at the time of the Challenger accident. In fact, the *rate* of erosion/blow-by had increased steadily since the beginning of the SRB contract in 1983. The seal problem was serious enough to lead both to briefings at Headquarters and to establishment of a re-design task force. Yet, in spite of all these problems, Thiokol was eligible to receive a near-maximum incentive fee of approximately $75 million. But, in the final analysis, it was NASA that both approved the SRB design

---

[218] NASA, Schedule D, NAS8-30490, April 12, 1983, p. 71.

[219] Cmte Hgs, Transcript, June 17, 1986, pp. 55–56.

[220] While this discussion focuses on possible contract penalties related to flight anomalies, it is also interesting to note that Thiokol has never been penalized for numerous safety and process violations at its Utah facilities. Several of these violations have resulted in serious fires and/or explosions.

and drew up an SRB contract which contained no provisions for performance penalties or flight-anomaly penalties. One must not fault Thiokol for collecting the bonus; one must fault NASA for allowing the bonus to be collected at all.

The problem with the kinds of penalties that were contained in the SRB contract is that, so long as management is convinced that a festering problem like the seal problem is not likely to cause mission failure, there is little incentive for the company to spend resources to fix the problem. In fact, if the solution involves significant delays in delivery, there may be a strong financial dis-incentive for the company to pursue a short-term solution aggressively. For example, Thiokol engineer R.M. Boisjoly provided a clear warning of the seriousness of the O-ring problem in July, 1985, and Thiokol engineer A. R. Thompson laid out a plan for a possible short-term solution to the problem.[221] Whatever its efficacy, why was Thompson's plan apparently dismissed so summarily? Part of the answer may be found in the June 18, 1986, exchange between Mr. Scheuer and Mr. Thompson:

> Mr. SCHEUER. Would the research and development of your fixes have delayed the delivery of ths SRMs to NASA?
>
> Mr. THOMPSON. . . . It probably would have delayed it a month or two, at least for the hardware and some of the research work. . . .[222]

The Committee is certainly not suggesting that anyone in NASA or Thiokol would recommend launch or would refuse to spend resources fixing a problem if it was known that the problem constituted a real threat to mission safety. However, in the case of the SRB joint, both NASA and Thiokol managers clearly misjudged the threat to mission safety. In situations of this sort, contractual provisions rewarding performance rather than cost and schedule would have provided a far stronger incentive to fix a long-festering problem. Ultimately, the balance between safety incentives and cost/schedule incentives in the SRB contract may illuminate a number of issues raised by the Challenger accident.

---

[221] Rogers Commission Report, Volume I, pp. 249–51.
[222] Cmte Hgs, Transcript, June 18, 1986, p. 18.

VII. CASING JOINT DESIGN

Discussion

(a) Introduction

The fact that the aft field joint of the right-hand Solid Rocket Booster failed at the 300 degree location is overwhelmingly supported by the evidence. Retrieval of two large pieces of the joint clearly show that they were destroyed by the heat and velocity of the gas flame emanating from the right-hand booster. Additional supporting evidence was found by reviewing the telemetry data and the photographs taken during launch and flight.[1]

For the purpose of redesigning the joint it is important that the way in which the joint failed be determined as closely as possible. This determination, however, is difficult, if not impossible, to make with one hundred percent certainty. The evidence to support progress of the failure through the joint is incomplete. However, based on the recorded history of the joint problems encountered in flight and in test, based on the laws of physics, and based on behavior of the materials used in the joint, the following PROBABLE CAUSE is offered.

(b) Probable Cause of Failure

1. Both the primary O-ring and the secondary O-ring were seated when the steel casings were mated. The pressure check verified this fact. However, from experience, the primary O-ring was seated in the upstream position as had been previously recognized by NASA and Thiokol engineers. (See Figure VII-1.)

[1] Rogers Commission Report, Volume I, pp. 22-23 and 78-79.

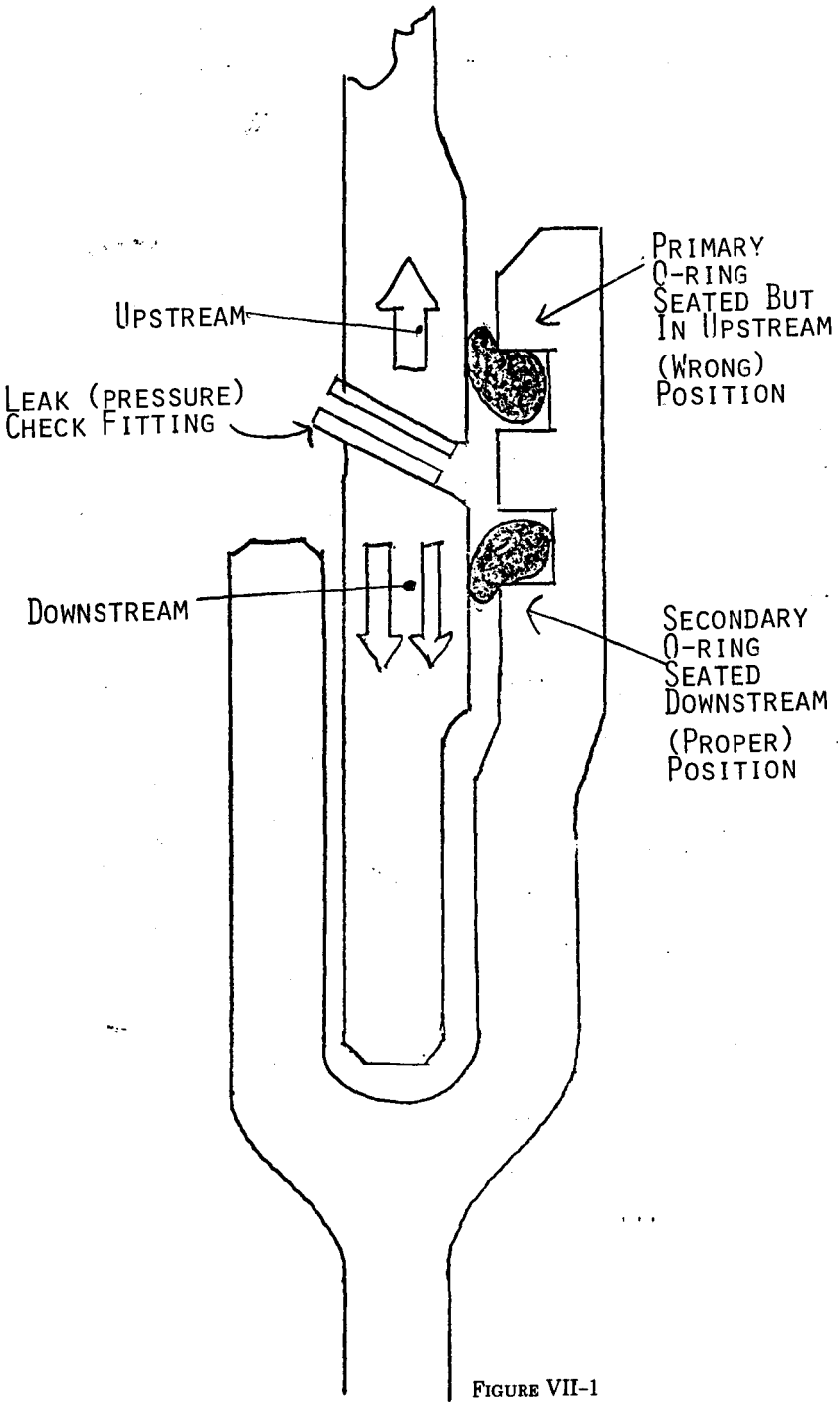

PRIMARY
O-RING
SEATED BUT
IN UPSTREAM
(WRONG)
POSITION
UPSTREAM
LEAK (PRESSURE)
CHECK FITTING
DOWNSTREAM
SECONDARY
O-RING
SEATED
DOWNSTREAM
(PROPER)
POSITION

FIGURE VII-1

2. Upon ignition, the primary O-ring could not reseat at the 300 degree location in the downstream position, where it needed to be in time to prevent blowby. At this point there were too many deficiencies acting in unison which prevented the O-ring from reseating in that location. First, proper spacing between the inner face of the tang and the opposing face of the inner leg of the clevis approximately 0.020 inches, is critical. That spacing for Flight 51-L was too small, at the 300 degree location where the smoke was observed, to facilitate prompt reseating of the primary O-ring. Calculations of segment diameters indicate the gap spacing was only 0.004 inches, near metal-to-metal contact. The ignition gases passed the O-ring at this location (See Figure VII-2). This condition did not exist elsewhere in the joint around the casings since the primary O-ring was able to seat around the joint in other locations.

Second, the low temperature throughout the night prior to launch left the fluorocarbon elastomer primary O-ring stiff and lacking in ability to spring into the downstream (seated) position at the 300 degree location in time, relative to the buildup of motor pressure, to provide a tight seal. The temperature of the aft field joint at time of launch was calculated by Thiokol after the accident to be 16 degree F. Part of the reason for this low temperature was the heat transfer away from the joint, by conduction through the aft attachment strut. The conduction was driven by liquid hydrogen, which remained in the external tank overnight. The supercold fuel created a 430 degree temperature differential across the ship, drawing heat out of the joint and O-rings.

At ignition, blowby occurred, either with erosion of the primary O-ring or without erosion.

64-420 0 - 86 - 7

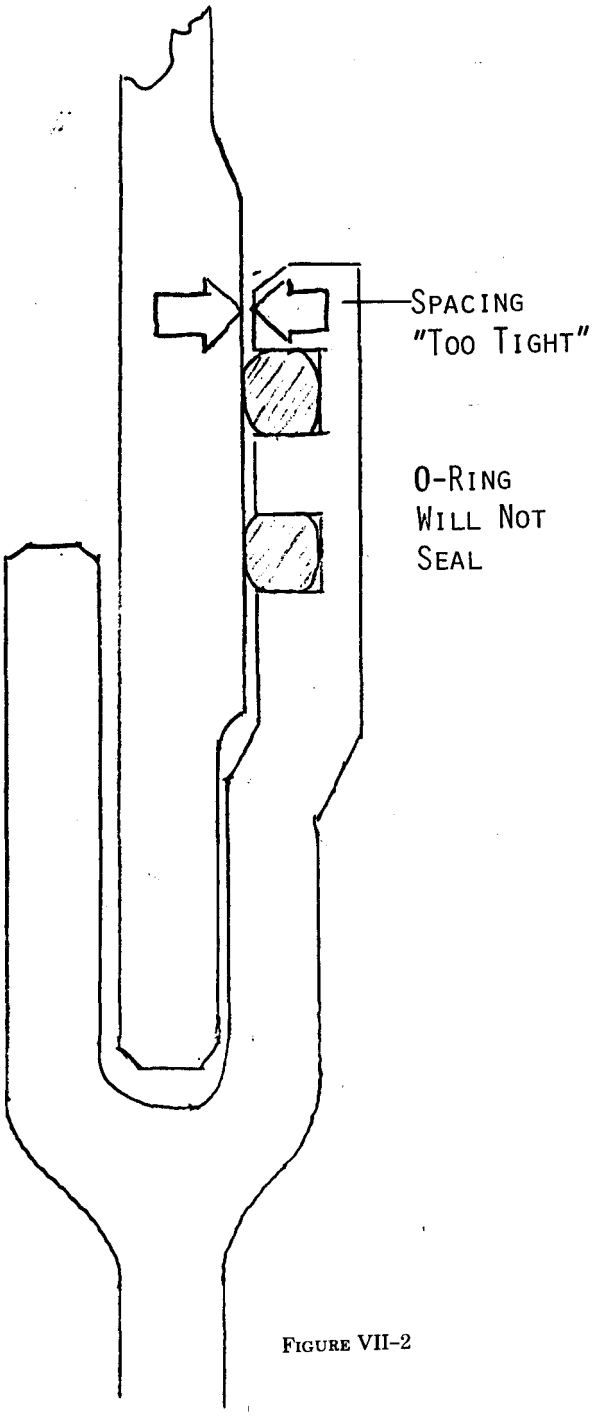


FIGURE VII-2

3. However, there could have been one or more than one blowhole through the zinc chromate putty before ignition. One such blowhole could have been made at the 300 degree location either during the leak check at 200 psi prior to the seating of the primary O-ring or prior to the leak check when the casings were brought together. If so, there was a high probability that the primary O-ring eroded at this point. The phenomena of blowing holes in the putty had been observed many times at post-flight dismantlement and had dramatically increased when the procedures were changed to increase the test pressure to 200 psi. Additionally, the Randolph putty had been found unsatisfactory on numerous occasions.[2]

4. Upon ignition of the Solid Rocket Motor, this blowhole would have facilitated and concentrated the hot propellant gas flame on the primary O-ring, and possibly the secondary O-ring as well. Alignment of O-ring erosion with the location of blowholes had been observed on numerous occasions.[3]

5. Between the time the casings were assembled and the launch, the secondary O-ring was unseated from its previously sealed position. The fact that it had been sealed has been verified by the pressure check made 28 days before when the casings were joined. Either the secondary O-ring was unseated by joint rotation coupled with O-ring stiffness or by the formation of ice in the joint.

During the intervening period, as the Shuttle stood on Pad 39B waiting for the launch, 7 inches of rain had fallen and some could have easily penetrated the joints. The access of rain water into the joints was proved when STS-9 was disassembled and water poured out of the assembly pin holes. In tests conducted after the accident, it was confirmed that the water in the aft field joint would have turned to ice, and that the ice could have dislodged the secondary O-ring, pushing it upstream into a non-sealed position. In this position, it is doubtful that the secondary O-ring could have sealed at ignition.

6. One of the three Solid Rocket Booster to External Tank aft attachment struts is also connected at the 300 degree location, just a few inches below the aft field joint. As the Space Shuttle system stood on the launch platform at Pad B on January 28, the large External Tank was gradually filled with liquid hydrogen and liquid oxygen. Liquid hydrogen, at a temperature of 423 deg F below zero, and liquid oxygen, at a temperature of 297 deg F below zero, caused the tank to contract as it was filled. Since the Solid Rocket Boosters are firmly bolted to the launch platform, a lateral force of approximately 190,000 pounds pulled sideways on the aft attachment strut and the Solid Rocket Motor casing, including the joint that failed.[4] Refueling of the tank was accomplished early on the morning of January 28.

At ignition, the 190,000-pound force was instantly released when the SRB hold-down bolts were blown loose. For the next two and a half seconds the right Solid Rocket Motor field joints experienced a 3 cycle per second vibratory load caused by the sudden release of

---

[2] NASA, MSFC Memo, Miller to Horton, April 12, 1984.

[3] Thiokol, "Erosion of SRM Pressure Seals," TWR 15160, Chart A-9, August 19, 1985: "Seal damage always has associated putty blowhole."

[4] NASA, MSFC, "51-L Analysis Overview," April 25, 1986, p. H-203.

the lateral force.[5] The ignition pressure increased the joint spacing. Also, the flow of motor gases through the blowhole at the 300 degree location could have resulted in damage to the primary O-ring. The evidence of smoke at the 300 degree location is unlikely without O-ring damage.

7. Smoke at launch, clearly visible in the photographs, stopped at 2.7 seconds when the vibratory load damped out and the joint sealed. The sealing of the breach at the 300 degree location was made possible by blockage from burned material, probably consisting of a mixture of insulation and aluminum oxide. Post-accident tests performed by Morton Thiokol proved that aluminum oxide could have successfully plugged the joint at 2.7 seconds. While the smoke at ignition appeared to be intermittent, that appearance was probably a result of air and main engine exhaust currents.

8. At T+37 seconds into the flight, the Shuttle encountered wind gust loads in conjunction with planned maneuvers. Components of these gust and maneuvering loads were transmitted to the Solid Rocket Booster through the External Tank attachment strut. Based on the prescence of smoke at liftoff, these forces were transmitted to a joint already weakened by erosion and heat damage.

9. At 43 seconds into the flight, the main engines throttled back as the Shuttle reached "Max q" (maximum dynamic pressure). Four seconds later, the main engines had throttled up to 104% power and the geometry of the Solid Rocket Motor propellants had increased thrust. At this point, the motor pressure increased to 609 psi.

Additional structural loads resulted from turbulence. Flight 51-L experienced the most severe turbulence of any Shuttle flight and, although the loads were within the allowable design limits, those design limits did not consider a joint that had already failed.[6] It is unknown how much the combined effect of wind gust loads, maneuvering loads and an increase in thrust contributed to the accident. But the combined effects of these forces could have dislodged the burned material at the previously breached section of the joint.

10. Shortly after the vehicle was loaded by these turbulent forces, at T+58 seconds, a flame appeared from the same general region where the puffs of smoke had been seen. But, this time the joint was continuously breached by the burning propellant gases. In a little over two seconds, the flame had grown and acted as a blowtorch to burn through the hydrogen tank. The appearance of the flame at this time is also indicative of a damaged primary O-ring and failure of the secondary O-ring to seal, for reasons explained in the Critical Items List dated December 17, 1982.[6a]

The telemetry data, photographs and cockpit voice recordings support evidence of turbulent conditions and the manner in which the Shuttle failed.

---

[5] The joint was designed to accommodate these loads.
[6] NASA, MSFC, "51-L Analysis Overview, STS 51-L—Wind Shears, April 25, 1986, p. H-597.
[6a] NASA, "SRB Critical Items List," December 17, 1982, page A-6A, sheet 1.

Time (min:sec)	Crew position[7]	Crew comment
T+19	PLT	Looks like we've got a lotta wind here today.
T+20	CDR	Yeah.
T+22	CDR	It's a little hard to see out my window here.
T+28	PLT	There's ten thousand feet and Mach point five.
T+30		(Garble)
T+35	CDR	Point nine.
T+37		[High thrust vector control (steering) activity noted. This was caused by upper atmoshpere wind gusts and planned maneuvers.]
T+40	PLT	There's Mach one.
T+41	CDR	Going through nineteen thousand.
T+43	CDR	OK, we're throttling down.
T+57	CDR	Throttling up.
T+58	PLT	Throttle up.
T+58		[Vehicle loaded by dynamic pressures.]
T+58		[A flame appeared from same general region where the puffs of smoke had been seen earlier.]
T+59	CDR	Roger.
T+60	PLT	Woooohoooo.
T+60		[Right SRM internal pressure began to diverge from that of left SRM]
T+61		[Well-defined plume was deflected indicating the plume had burned through the liquid hydrogen tank structure.]
T+62	PLT	Thirty-five thousand going through one point five.
T+64.7		[Liquid hydrogen leak noted.]
T+66.8	CDR	Reading four eighty six on mine.
66.800		[Leak confirmed when hydrogem tank leak pressurization system was unable to maintain normal pressurization rate.]
T+67	PLT	Yep, that's what I've got, too.
T+70	CDR	Roger, go at throttle up.
T+72.2		[Right Solid Rocket Booster motion differed from Orbiter and left Solid Rocket Booster, indicating failure of lower attachment structure.]
T+72.6		[Liquid hydrogen tank pressure fell. Leak was growing rapidly.]
T+73		[Liquid hydrogen and liquid oxygen pressure to main engines showed significant drop.]
T+73	PLT	Uh Oh
T+73.1		[Circumferential white pattern around the External Tank aft bulkhead suggested liquid hydrogen tank structure failure.]
T+73.1		[Vapor observed at inter-tank which was indicative of the liquid oxygen tank failing. Liquid oxygen then observed.][8]

[7] (CDR) Commander Scobee, (PLT) Pilot Smith, (MS 1) Mission Specialist Onizuka, (MS 2) Mission Specialist Resnik.
[8] NASA, D.M. Germany, STS 51L Incident Investigation, Integrated Events Time Line, Johnson Space Flight Center, June 4, 1986, as modified.

(c) Problems Discovered

The design of the joint was based on the successful design of the joints used on the Titan III booster rocket.[9] That design was similar except that the tang pointed upward, instead of down, and the clevis pointed downward, instead of up, as in the case of the Shuttle booster. Another difference was that the design of the Shuttle joint included two O-rings instead of one as provided for in the

[9] NASA, "SRB Critical Items List," December 17, 1982, p. A-6A, Sheet 1.

Titan design. But, the most important difference was the use of putty in the Shuttle design. While the Titan employed the NBR insulation to close the gap between segments, the Shuttle design called for filling a gap between insulation with putty.

The Shuttle design was changed to accommodate manufacturing constraints.[10] The Shuttle booster is larger, 146 inches in diameter as compared to 120 inches for the Titan. As a result of its larger size, the Shuttle booster uses more steel. While this requirement for more steel had no impact on other booster components, it did have an impact on the joint design. The maximum billet size (a piece of metal made from an ingot) commercially available to manufacture the large, one-piece, weld free forward dome with an integral forward skirt tang was less than that needed for the Shuttle Solid Rocket Boosters. However, it was found that by turning the casings upside down, there would be just enough metal to manufacture a forward dome because that component would then only have to incorporate the single joint element, the tang, instead of the double joint element, the clevis.

It is good engineering practice to design products to accommodate manufacturing tooling capabilities and methods. Furthermore, with the clevis facing up and the tang down, field assembly at the Kennedy Space Center was simplified. Combined with the extra O-ring, the design change appeared reasonable. But it is also good engineering practice to accommodate all the forces and conditions that the product must perform under during its useful life. The design of the Shuttle Solid Rocket Motor, as opposed to the Titan, had to provide for reuse of the propellant casings, including the wearing of joint surfaces and distortion of the case in handling and shipment. It had to accommodate heavier propellant loads. The design was more susceptible to water entry during storms. And, most significant, the design had to accommodate a combination of dynamic structural loads significantly different than those encountered by the Titan.

It is always a simple task to find fault with someone else's work; especially after an accident occurs. It is quite another matter to originate the work and produce a useful product.

The joint design provided a direct path between the combustion chamber, consisting of an annulus with propellant surrounding it, and the outside of the steel motor casings. That path was sealed with putty and two circular fluorocarbon elastomer (rubber-like) bands called O-rings. While O-rings are frequently used to retain pressures much higher than those present in the Shuttle Solid Rocket Motor, thermal and structural forces acting on the Shuttle joints are formidable. These joints must carry and transfer these loads between the casings.

Another essential ingredient of good engineering practice is to use material suited to the function. Some O-rings can withstand high temperatures. But "all . . . elastomers become brittle at low [temperatures]. . . . Elastomers, like natural rubber, nitrile rubber, and Viton A . . . that become brittle at low [temperature]

[10] Staff discussion with E.G. Dorsey, Thiokol Wasatch Operations, Brigham City, Utah, September 4, 1986.

can be used for static seal gaskets when highly compressed at room temperature prior to cooling."[11]

But the Shuttle's O-rings were not used in a "static" system as evidenced by the variations in gap spacing between the tang and clevis. Nor would they always be highly compressed at room temperature prior to the cooling. Furthermore, the O-rings could not withstand burning propellant temperatures in the range of 5800° F. The design of the joint therefore provided for putty to insulate the O-rings from the burning gases.

This putty did not always perform as had been expected, and evidence of hot gas passing the putty and getting to the first, or primary, O-ring along the path to the outside of the rocket chamber was discovered. Once the putty was breached, the joint was not working as it had been designed. This failure, although recognized by NASA and its contractor Morton Thiokol, was neglected on March 8, 1984 when they chose to accept an "allowable degree of erosion," which meant there was an allowable percentage of failure.[12]

O-rings become effective (are seated) when pressure is applied to them as they sit in a groove provided to house them.

One question that the design was intended to answer was whether or not the O-ring was seated properly in its groove. An opening, with a fitting much like a valve stem on a tire, was provided to allow pressure testing between two such O-rings, the primary and the secondary.

But this design did not always answer the question: was the primary O-ring seated? Did it seal or not? Notice how the primary O-ring in Figure VII-1 (p. 176) is forced upward (shown by the single arrow). That is opposite to the normal direction that the propellant pressure acts (notice the double arrows). Even with an acceptable pressure check result, the primary O-ring would still be unseated for a fraction of a second when the motor pressure pushed the O-ring in the opposite direction from that which took place during the leak check.

The second reason the assumption concerning the leak check as "proof of sealing" could be erroneous was that the primary O-ring did not really have to seat at all if the putty behind it (toward the inside of the case) held the pressure during the leak check. So earlier in the program there really was no way to know whether the primary O-ring seated or not.

What appeared to be a rather straightforward joint was far from simple. If the primary O-ring did not seat during the leak check, and the pressure test succeeded, then the putty was doing the work of sealing. But it still was not possible to determine from outside the casings whether the putty or the O-ring was holding the pressure. But, if the leak check failed, then the O-ring was not seated and there was a blow-hole through the putty.

To resolve this concern, NASA and its contractor, Morton Thiokol, changed the leak-check procedure by increasing the pressure until a pressure of 200 pounds per square inch (psi) was accepted as

---

[11] Theodore Baumeister, Editor, *Standard Handbook for Mechanical Engineers*, 7th Ed., (New York: McGraw-Hill, 1967), pp. 18-35.

[12] Rogers Commission Report, Volume II, p. H-1.

the standard. They had ascertained that this was sufficient pressure to blow a hole through the putty.[13] Then, if the O-ring failed to seat, the pressure would blow a hole through the putty and the test would disclose an unseated O-ring (a failed seal). But if the O-ring held the higher pressure, the O-ring would still have been seated in the upper position instead of the downward position. That would be contrary to the way the O-ring would have to be seated to contain the propellant pressure during launch of the Shuttle.

In summary, there was still no way to verify whether the primary O-ring was seated properly, meaning in the downstream position after the cases were joined together in the field. In the beginning of the development program the concept was that the putty would act somewhat like a "piston in a cylinder" when the propellant was ignited. As the chamber pressure built up, the putty was to move downstream and compress the air in the path between it and the primary O-ring. The compressed gas was to seat the O-ring and thereby seal the joint. Besides, even if the primary O-ring didn't seal, surely the secondary O-ring would, since it had already been pressure checked, which verified it was seated in the downstream position.

There was no direct evidence that the primary O-ring was not holding the pressure off the secondary ring until Flight 51-B. That was the first flight when erosion of the secondary O-ring had been observed, even though erosion of the primary O-ring had occurred before.[14]

Thiokol had considered the joint design to be Criticality 1R,[15] meaning that there was redundancy. While the second O-ring was redundant by design, the joint as a whole was still Criticality 1, since if it failed, it would mean the loss of the Shuttle and crew. In other words, there was no backup for the joint.

The joint was designed to mate two rocket motor segment cases, one to the other, where the lower edge of the upper case consisted of a tang and the upper edge of the lower case consisted of a clevis. After the tang was inserted into the clevis (which housed the two O-rings), 177 steel pins, each approximately 1 inch in diameter, were inserted from the outside through aligned holes which went through the outer leg of the clevis, the tang and partly into the inner leg of the clevis. The spacing between the inner face of the tang and the mating face of the inner leg of the clevis where the O-rings were housed was critical to the integrity of the joint because that spacing, in part, determined whether the O-rings could function properly to seal against the propellant gas pressures. Not only was the initial static spacing critical, but maintaining the proper spacing during launch and flight under dynamic structural loadings was necessary for an effective seal.

Upon ignition of the Solid Rocket Motor fuel the operating pressure increases to 922 psi at 40 degrees F within a little over one-half second (0.648 sec).[16] The effect of this pressure increase is to

[13] NASA, MSFC, Problem Assessment System Record No. A07934, January 23, 1986, p. 6.
[14] Rogers Commission Report, Volume V, p. 1510.
[15] Cmte Hgs, Transcript, June 18, 1986, p. 51.
[16] Morton Thiokol, TWR-10212 (CD), Table 4-9, *Typical Propellant Design Data.*

cause the casings to bulge out around their midsections while being constrained by the thicker steel sections at the ends, much like a can of soda after freezing. The casings change shape during the buildup of motor pressure. This bulging has an effect on the joint. As in the case of the frozen soda can, the wall of the casing near the joint is no longer vertical, or perpendicular to the bottom, but angles out to meet the larger diameter in the center of the casing. NASA calls this change in angle at the joint "joint rotation."

This joint rotation is a component of an overall spacing problem that includes: changes caused by casing wear and tear experienced during refurbishment; case growth (swelling) from pressurizing the casings; distortion that occurs during shipment of the loaded casings; and the physical handling of the casings during stacking operations.

The joint rotation problem was aggravated when the steel casings were made thinner to achieve a reduction in weight and thus an increase in payload. The rotation problem was further aggravated by changing the design of the propellant geometry to achieve greater thrust. This increased the pressure within the casings and thereby increased the "gap opening."[17] These changes compromised the integrity of the joint seals because joint rotation increases the spacing (gap) between the tang and the O-ring grooves in the clevis.

When the increase in the gap occurs, it can open the O-ring seal, leaving the path from the propellant combustion chamber open to the outside of the casing, except for any blockage by the putty. But, as noted above, the putty frequently has holes blown through it. If there were blowholes in the putty, and the original spacing between the metal parts of the joint was such that the joint rotation left open spaces between the O-rings and the tang, then the joint would fail and burning gases would escape to the outside.

(d) Joint Behavior

In a memo from John Miller to Mr. Eudy of NASA on June 16, 1980, the following statement was made:

> STA-1 test data shows that the secondary O-ring can become unseated from the tang due to joint rotation at approximately 40 percent of MEOP [Mean Effective Operating Pressure], and therefore, is not likely to assume a sealing position should the above primary seal failure occur. The SRM has never been tested to evaluate the above failure condition, nor has credibility of such a failure been officially declared.[18]

In March of 1984 Thiokol had completed its SRM O-ring assembly test plan, which was to confirm the O-ring erosion scenario, provide data for heat transfer predictions and establish the effec-

[17] The Light Weight Casings, first used on STS-6, had thinner casing walls than the standard steel casings. Light weight casings permitted flight with heavier payloads. On STS-8, NASA began using the High Performance Motor (HPM) which developed higher internal pressures while using the light weight casings. The purpose of the HPM was to further increase payload capacity.

[18] NASA, "Evaluation of TWR-12690 CD, Test Plan for Space Shuttle SRM Lightweight Cylinder Segment Joint Verification, dated June 10, 1980", EP 25 (80-70), June 16, 1980, p. 2.

tiveness of the vacuum putty. The introduction to that plan included the statement:

> O-ring seals in rocket motors in general and the Space Shuttle SRMs in particular, can suffer thermal degradation because of exposure to the high temperature motor chamber gases. Although none of the SRM primary O-rings to date have failed to perform their design function, there is some concern because of isolated events which show localized erosion as high as 0.053 inches. The postulated scenario for this thermal degradation effect is a short-time duration impingement of a high energy jet which is induced during ignition pressurization by a combination of voids in the protective vacuum putty and the filling of available free volumes created by the tolerances of mating parts and the O-ring slots. Unfortunately, the overall assembly and the vacuum putty layup does not lend itself to a well-defined geometry for predicting the hot gas flow and associated heat transfer to the O-rings.[19]

A subsequent report, dated May 7, 1984, contained a statement:

> Symptom of failure: a vaccum putty exhibited gas paths located at 319 deg., 338 deg., and 347 deg. Erosion of the primary O-ring occurred at 319 deg. only. The damaged region was approximately 5.6 inches long with a .034 inch maximum depth and involved 136 deg. of the O-ring cross section diameter.[20]

In a memo from Larry Mulloy to Bob Lindstrom, Director, MSFC Shuttle Projects Office, in November of 1984 it was noted:

> . . . it was determined that shims could be used to make the case joint sufficiently concentric to consistantly achieve a 7.54 percent minimum O-ring squeeze. Therefore the 7.54 percent has been established as the minimum acceptable requirement for both case and nozzle O-ring joints and verified by subscale testing and full scale experience.[21]

On a 0.280 inch diameter O-ring a 7.54 percent squeeze would be equal to a compression distance of 0.021 inches.[22]

On July 17, 1985, Irv Davids, Manager of the Solid Rocket Booster Program at NASA Headquarters, sent a memo to the Associate Administrator for Space Flight, the subject of which was case-to-

---

[19] Thiokol, Philip Shadlesky, "Performance Characteristics of the SRM O-ring Assembly Test Plan", TWR-14336, dated March 1984, p. 1.

[20] Thiokol, S. Rodgers, "Significant Problem Report DR4-5/35: 5 Day Report; O-ring Erosion at Nozzle/Aft Segment Joint of SRM 11A (STS 41-B/Mission 41-C)", TWR-14370-1, May 7, 1984, pp. 1-2.

[21] O-ring squeeze is the distance, in fractions of an inch, that an O-ring is compressed from its normally round shape. This dimension can also be expressed as a percentage of the total diameter before compression. In 1984 NASA was using a term "minimum O-ring squeeze." During an SRM design analysis of the case and nozzle O-ring joints it was concluded that the 146 inch diameter case cylinders would not meet the design standard of 15 percent minimum O-ring squeeze at zero pressure. The various problems that prevented this included flaws in the O-ring grooves and sealing surfaces and differences in the spacing between tang and clevis on various casings.

[22] NASA, Larry Mulloy, "ECP SRM 1197, Nozzle Nose Inlet Housing O-ring Squeeze," SA 42-562-84, November 20, 1984, p. 1.

case and nozzle-to-case O-ring seal erosion problems.[23] Davids sent copies to Messrs. Weeks, Hamby, Herrington and Winterhalter.[23a] In the memo it was noted that there has been twelve instances of primary O-ring erosion during Shuttle flights. In addition, in one specific case there had also been erosion of the secondary O-ring seal. There were also two primary O-ring seals that were heat affected without erosion and two cases in which soot blewby the primary seals. In this memo it was noted that the prime suspect for the cause of erosion on the primary O-ring seals was the type of putty being used. It was Thiokol's position that during assembly leak check, or ignition, a hole could be formed through the putty which then initiated O-ring erosion due to a "jetting effect." It was even mentioned in this memo that Thiokol was seriously considering the deletion of putty on the QM-5 nozzle/case joint since they believed the putty was the prime cause of the erosion. Davids, however, had reservations about deleting the putty because he recognized the significance of the QM-5 firing in qualifying the FWC (Filament Wound Case) for flight.

In the matter of case-to-case O-ring erosion the memo noted that there had been five occurrences during flight where there was primary field joint O-ring erosion. There was also one case where the secondary O-ring was heat damaged with no erosion. The memo stated:

> The erosion with the field joint primary O-ring is considered by some to be more critical than the nozzle joint due to the fact that during the pressure build up on the primary O-ring the unpressurized field joint secondary seal unseats due to joint rotation.[24]

The memo continued:

> The problem with the unseating of the secondary O-ring during joint rotation has been known for quite some time. In order to eliminate this problem on the FWC field joints a capture feature was designed which prevents the secondary seal from lifting off.[25]

Lastly the memo noted:

> The present consensus is that if the primary O-ring seats during ignition, and subsequently fails, the unseated secondary O-ring will not serve its intended purpose as a redundant seal. However, redundancy does exist during the ignition cycle, which is the most critical time.[26] (See Appendices VII-B and VII-C.)

On August 2, 1985, Larry Wear, MSFC's SRM Element Manager, sent a letter to Joseph Kilminster, Thiokol's Vice President for Space Booster Programs, on the subject of SRM field joint second-

---

[23] NASA, Irving Davids, "Case to Case and Nozzle to Case 'O' Ring Seal Erosion Problems," July 11, 1985.

[23a] Mr. Weeks, Dep. Assoc. Administrator for Space Flight (Technical); Mr. Hamby, Dep. Dir., STS Program Integration; Mr. Herrington, Dep. Dir. of Launch & Landing Operations; and Mr. Winterhalter, Acting Dir., Shuttle Propulsion Div.

[24] Ibid., p. 2.

[25] Ibid.

[26] Ibid.

ary O-ring lift-off during pressurization. The letter concerned the situation wherein one O-ring might not seal subsequent to joint rotation. The letter stated:

> Because of recent experiences of flight and ground test motors having increasing incidences of putty blow-holes and the associated burning of primary O-ring, it would seem prudent for us to attempt to assure that the secondary O-ring is capable of sealing during the entire SRM burn.[27]

The letter requested an assessment of the possibility of lift-off of the secondary O-ring.

In August of 1985 Jim Thomas, MSFC's Deputy SRM Element Manager, wrote a memo for Mr. Mulloy to Mr. Hamby at NASA Headquarters, which was apparently never signed or sent. The subject of the memo was SRM Joint/O-ring Erosion. The memo stated:

> On July 11, 1985, you and Irv Davids were briefed by Jim Thomas of my office on the history of the effort underway to resolve the issues and concerns of the above subject.

The memo than went on to discuss a number of questions.

1. What would happen if the secondary seal lifted off the mating surface during motor pressurization, and, also, how long it would take for the seal to return to a position where contact was made? The answer to that question stated that bench test data indicated that the O-ring resiliency, that is, its capability to fill the gap between the tang and the clevis, was a function of temperature and the rate at which the gap opened.

The memo stated, "at 100 deg. F the O-ring maintained contact. At 75 deg. F the O-ring lost contact for 2.4 seconds. At 50 deg. F the O-ring did not reestablish contact in 10 minutes at which time the test was terminated." The memo then stated, "the conclusion is that secondary sealing capability in the SRM field joint cannot be guaranteed." [28]

2. Another question concerned whether or not the secondary O-ring would seal in sufficient time to prevent joint leakage if the primary O-ring had not sealed. The answer to that question was as follows:

> MTI has no reason to suspect that the primary seal would ever fail after pressure equilibrium is reached, i.e., after the ignition transient. If the primary O-ring were to fail from 0 to 170 milliseconds, there is a very high probability that the secondary O-ring would hold pressure since the case has not expanded appreciable at this point. If the primary seal were to fail from 170 to 330 milliseconds, the probability of the secondary seal holding is reduced. From 330 to 600 milliseconds the chance of the secondary seal

---

[27] NASA, Larry Wear, "SRM Field Joint Secondary O-ring Lift-Off During Pressurization," SA 41-326-85, August 2, 1985.

[28] Engineering consultants to the Committee have serious questions as to how this test relates to actual O-ring performance in flight hardware.

holding is small. This is direct result of the O-ring's slow response compared to the metal-case segments as the joint rotates.

3. The third question indicated that NASA Headquarters was not aware that the secondary O-ring may not seat due to joint rotation, and they wanted to know when this data was incorporated into the FMEA/CIL? The answer noted that Thiokol had submitted a TWR-13520 to MSFC in December of 1982. This was approved by NASA Level III on January 21, 1983. NASA Level II authorized a change request March 2, 1983 and Level II issued a PRCBD to implement approved Level I change request on May 2, 1983.[29]

Thiokol completed their engineering study of O-ring compression set and dated the report October 2, 1985.[30] (Compression set relates to the ability of a material, in this case, O-rings, to rebound to its original dimensions after having being subjected to compression for various periods of time and or at various temperatures.) That report contained the following information. There was a concern of the ability of the O-ring to rebound to or near its original dimensions after having been subjected to compression for various periods of time and at various temperatures. The Parker Seal Company of Culver City, California, tested several O-rings to determine the properties of the material. Two compression set tests in accordance with ASTM (American Society for Testing and Materials) D-395 method B were performed.[31] The first test was conducted at a constant temperature of 75 deg. and the time that the ring was in compression was varied. In the second test the temperature was varied and the compression was held constant. A small O-ring of 0.139 inch diameter was used for test purposes. The test showed that the percentage of compression increased with an increase of temperature. However, these tests were not conducted at low temperatures. Rather, they were conducted at temperatures of 212 deg. F and above and therefore, they have little relevance to ambient conditions.

A status report from Thiokol's SRM O-ring Task Force, presented on November 20, 1985, recommended that a slightly larger Viton O-ring of 0.292 inch diameter, along with thicker shims, be used as a short-term solution. The current O-rings were 0.280 inches. Thiokol pointed out that there would be more erosion margin due to greater material thickness at the sealing surface. They noted that the thicker shims would reduce the initial and absolute final gap opening dimension, resulting in more O-ring "squeeze" initially. Thiokol stated that the greater initial squeeze would be better for compression set and resiliency, and would give a higher probability of maintaining a secondary seal longer into the ignition transient. Thiokol also noted that various tests were conducted on the Randolph putty using hot five-inch char motors.[32] Two tests were conducted, which determined that the

[29] NASA, Larry Mulloy, "SRM Joint/O-ring Erosion," SA 42-349-85, August 1985, pp. 1-2.
[30] Thickol, B.L. Orme, "Enginering Study of O-ring Compression Set," TWR-15218, October 2, 1985.
[31] Refer to Appendix VII-A for ASTM specification.
[32] Small scale test motors.

putty erosion could take place at a rate between 5.5 and 13.0 mils per second. Two other tests noted that the erosion on GS-43 [33] putty was ten times higher than that on the Randolph.[34]

Primary concerns drawn from the charts provided by Thiokol on January 27, 1986, centered around the following items. During the ignition transient, 0 to 170 milliseconds, there is a high probability of a reliable secondary seal. Between 170 and 330 milliseconds there is a reduced probability of a reliable secondary seal and between 330 and 600 milliseconds there is a high probability of no secondary seal capability. Under steady state conditions, between 600 milliseconds and two minutes, the notes states "if erosion penetrates primary O-ring seal—high probability of no secondary seal capability." [35]

A. Bench testing showed O-ring not capable of maintaining contact with metal parts gap opening rate to MEOP.

B. Bench testing showed capability to maintain O-ring contact during initial phase (0 to 170 ms) of transient.[36]

What follows is taken from Chart 2-2:

1. A temperature lower than current data base results in changing primary O-ring sealing timing function.
2. SRM 15-A 80 deg. arc black grease between O-rings. SRM 15-B 110 deg. arc black grease between O-rings.
3. Lower O-ring squeeze due to lower temperature.
4. Higher O-ring Shore hardness.
5. Thicker grease viscosity.
6. Higher O-ring pressure activation time.
7. Activation time increases, threshold of secondary seal pressurization capability is approached.
8. If threshold is reached then secondary seal may not be capable of being pressurized.[37]

The presentation went on to included the following blow-by history:

SRM 15 worst blow-by.

A. Two case joints (80 deg.), (110 deg.) arc.

B. Much worse visually than SRM 22. SRM blow-by.[38]

The presentation then included a chart titled "O-ring (Viton) Shore Hardness vs. Temperature." [39]

Degree F	Shore Hardness
70 degrees	77 hardness
60 degrees	81 hardness
50 degrees	84 hardness
40 degrees	88 hardness
30 degrees	92 hardness
20 degrees	94 hardness
10 degrees	96 hardness

[33] A type of putty made by another company that also was considered for use in the SRM.
[34] Thiokol, "SRM O-ring Task Force Status and QM-5 Recommendations," TWR-15349, November 20, 1985.
[35] Thiokol, "Temperature Concern on SRM Joints," January 27, 1986, chart 2-1.
[36] Ibid.
[37] Ibid., Chart 2 2.
[38] Ibid., Chart 3-1.
[39] Ibid., Chart 4-1.

The term Shore Hardness refers to a method of identifying the hardness of materials, and a higher number means a harder material. Regardless, though, it is seen from the above table that the hardness increases as the temperature decreases. Engineers presented a chart titled Secondary O-ring Resiliency, listing the following temperatures.[40]

Temperature degree F	Time to recover (seconds)
50 degree	600 recover
75 degree	2.4 recover
100 degree	*did not separate

The conclusions presented at the end of the teleconference were:

1. Temperature of O-ring is not only parameter controlling blow-by. SRM 15 with blow-by at an O-ring temperature at 53 deg. F. SRM 22 with blow-by at an O-ring temperature at 75 deg. F. Four development motors with no blow-by were tested at O-ring temperature of 47 deg. to 52 deg. F. Development motors had putty packing which resulted in better performance.
2. At about 50 deg. F blow-by could be experienced in case joints.
3. Temperature for SRM 25 on 1/28/86 will be 29 deg. F 9:00 a.m., 32 deg. F. 2:00 p.m.
4. Have no data that would indicate SRM 25 is different than SRM 15 other than temperature.[41]

Recommendations

1. O-ring temperature must be greater than or equal to 53 deg. F at launch. Development motors at 47 deg. to 52 deg. F with putty packing had no blow-by. SRM 15 (the best simulation) worked at 53 deg. F.
2. Project ambient conditons (temperature and wind) to determine launch time.[42]

The effect of Thiokol's recommendations would be that the Shuttle should not be launched unless the O-ring seal temperature was at least 53°F.

(e) Loads Acting on the Joint

There are other loads on the joint in addition to those caused by the pressures of the burning propellant. The following table identifies those loads relative to time.[43]

Time	Activity	Source of load	Static or dynamic	Impact on joint
Days before launch.....	Mating of casing.....	Weight of upper casing contacting lower casing.	Static plus impact.....	Physical contact between tang and clevis.

[40] Ibid., Chart 4-2.
[41] Ibid., Chart "Conclusions."
[42] Ibid., Chart "Recommendations."
[43] This Chart was prepared by the Committee and is based on information obtained by Committee staff during meetings at MSFC on June 30, 1986.

Time	Activity	Source of load	Static or dynamic	Impact on joint
Days before launch	Solid rocket motor assembly (Stacking) at KSC.	Weight of SRB components.	Static	Compressive; shear on pins at faces between tang and each clevis leg.
Days before launch	Joining of the external tank, and orbiter to SRB's.	Additional weight of tank and orbiter.	Static	Compressive; additional shear on pins which connect tang to clevis.
Days before launch	Transport to pad on crawler.	Movement of Transporter	Static and dynamic	Compressive; slight shear changes on pins.
Days before launch	Addition of payloads	Added weight	Static	Additional compressive and shear loads.
Within 24 hours of launch.	Loading of fuel	Weight of liquid hydrogen and liquid oxygen.	Static	Additional compressive and shear loads.
Within 24 hours of launch.	Loading of fuel	External tank contracts in diameter due to reduction in temperature.	Static	Lateral tensile force applied by aft attachment structure between external tank and solid rocket motor casing.
6 seconds to launch	Firing of main engines (SSME's).	Thrust of engines	Static and dynamic	Further moments compressive and vibratory (25 to 30 Hz).
At start of launch	Solid rocket motor ignition before lift-off.	Combustion pressures	Basically static	Bending (Joint rotation) lateral forces perpendicular to casing walls.
At start of launch	Solid rocket motor ignition before liftoff.	Engine thrust	Static and dynamic	Instant load reversal from compressive to tensile in joints and instant shear reversal in pins.
At start of launch	Release of hold down bolts.	Instant release of lateral force at aft External Tank attachment structure.	Dynamic	Instant change in stress, vibratory at 3 Hz.
At liftoff	Launch maneuvering	Thrust, plus nozzle vector forces.	Static and dynamic	Combination: Vibration, tensile, shear, lateral (via attachment structure).
Launch phase	In-flight maneuvering	Thrust plus gimbaling, applied loads at attachments.	Static and dynamic	Combination vibration, tensile, shear, lateral.
Launch phase	Turbulence—wind gust loads.	Impact, thrust nozzle gimbaling (changes in applied loads).	Static and dynamic	Impact loads transmitted to joints.
Launch phase	Reduction of main engine power and Solid Rocket Motor thrust at Max q (maximum dynamic pressure).	Decrease in thrust	Static and dynamic	Changes in bending and stress in joint, changes in frequency of vibration.
Launch phase	Increases in thrust thrust of main engines and solid rocket motors.	Increase in thrust	Static and dynamic	Changes in bending and stress in joint and in vibration.
Separation phase	Burning out of solid propellants and explosive forces at attachment points.	Release of thrust, impact forces at attachment points.	Static and dynamic	Reduction of tensile forces and shear on pins.

Time	Activity	Source of load	Static or dynamic	Impact on joint
Ocean impact and retrieval.	Weight of SRB impacting ocean at about 60 mph (vertical) 25 mph (horizontal).	Impact on joint	Dynamic	Variation in stresses at joints.

How these loads are accommodated by the joint is critical to the seal. In Thiokol's analytical evaluation report (TWR-12019, dated October 6, 1978), S. Stein of the Structures Section included the statement, "except in local area of pin, all stress levels are considerably below yield." [43a] As a result of this information, the Committee will explore this condition as part of its normal oversight work to determine the long-term effect on structural integrity of the casings.[44] Stein also wrote, "at MEOP [maximum expected operating pressure] the primary 'O' ring gap increases 0.052 and the secondary 0.038"." [45] It should be noted however, the analysis was made for no thrust, i.e. internal pressure only.[46] As noted on the forgoing chart, loads on the joint do work in combination and so the analysis should also provide for the combined effect of all loads at the time they occur.

On page 55 of the Rogers Commission Report there is a chart which shows a series of curves which relate maximum aerodynamic force to Mach Number. As a result of a discussion with Dr. Richard Feynman, Department of Physics, California Institute of Technology, and a member of the Rogers Commission, the Committee will review these curves after the completion of this report in an effort to ascertain their validity. There is reason to suspect that the "flight envelope" as repressented in the chart is inaccurate.[47]

As stated previously, the proper choice of materials is critical to attaining performance objectives. The steel casings are designed to withstand the propellant pressures and loads incurred in flight. Secondly, they must accommodate these forces over and over as the casings are reused. Consequently, the choices of the type of steel selected was important.

The steel used to make the casings and the joint is a D-6A. D-6A is a low-alloy steel for aircraft and missile structural applications. It is designed primarily for use at room-temperature tensile strengths of 260 to 290 ksi.[48] D-6A maintains a very high ratio of yield structure to tensile strength up to a tensile strength of 280 ksi, combined with good ductility.

Typical mechanical properties of D-6A steel: [49]

[43a] Thiokol, S. Stein, "Analytical Evaluation of the Space Shuttle Solid Rocket Motor Tang/Clevis Joint Behavior", TWR-12019, October 6, 1978, p. 1.

[44] Thiokol, S. Stein, "Analytical Evaluation of Space Shuttle SRM Tang/Clevis Joint Behavior", TWR-12019, October 6, 1978.

[45] Ibid.

[46] Ibid.

[47] Discussion with Dr. Richard Feynman, California Institute of Technology, Pasadena, California, September 3, 1986.

[48] 1000 pounds per square inch equals 1 ksi.

[49] American Society for Metals, Handbook Edited by H.E. Boyer and T.L. Gall, November, 1984.

Tempering temperature		Tensile strength ksi	Yield strength ksi	Elongation in 50 mm or 2 in., percent	Reduction in area, percent	V=notch impact energy ft-lb
°C	°F					
150	300	299	211	8.5	19.0	10
205	400	290	235	8.9	25.7	11
315	600	267	247	8.1	30.0	12
425	800	236	228	9.6	36.8	12
540	1,000	210	204	13.0	45.5	19
650	1,200	150	141	18.4	60.8	30

Normalized at 900°C (1650°F) and tempered at various temperatures.

In addition to the steel, other principal materials in the joint design that were to seal in the propellant gases were the zinc chromate putty and the O-rings.

On April 12, 1984, John Miller, Chief of the Solid Motor Branch of NASA, wrote a memo to Mr. Horton, Chief Engineer, SRB Engineering Office, MSFC which referred to concerns with putty made by Randolph. The Randolph putty was selected on the basis that it had several desirable performance characteristics. The change in putty was made after Fuller-O'Brien discontinued making putty because their product contained asbestos. Mr. Miller noted, "Stacking difficulties and observed O-ring anomalies appear to be more frequent with Randolph putty than with the previously used Fuller-O'Brien putty." [50] Miller requested that Thiokol expedite development and qualification of a putty with properties similar to those of Fuller-O'Brien.

On June 18, 1984, Miller wrote Horton again, mentioning erosion/heat exposure O-ring experience on QM-4, STS-2, STS-6, STS-11, and STS-13 and citing Deficiency Reports which violated specifications.[51]

By June 29, 1984, 5 inch motor tests has been completed. These tests substantiated the concept of hot gas jet impingement against O-rings. Interestingly, a simulation of "no putty" yielded no O-ring damage. This information was conveyed to NASA via telecon from Thiokol, which also stated that there was no second source for the Randolph putty. Thiokol had abandoned their program to mix the putty themselves. Measures taken to correct the putty problems included changes in the putty layup to reduce air entrapment, use of a porous sacrificial heat barrier such as carborundum fiberfrax or removing the putty and reducing joint gaps were introduced.[52]

A new joint design was forwarded to NASA by Thiokol on July 19, 1984, which included a fill capture feature. This feature looked similar to the "capture feature" proposed for future Shuttle flights. The fill capture feature, however, was to be filled with grease. A thermal analysis had shown that "severe heat effects would result if the cavity were not filled."

As stated previously, the putty was to insulate the O-ring seals from the hot propellant gases. It was also to remain flexible enough to move outward under the pressure of the burning propellant, thereby compressing the gas in the joint which, in turn, was

[50] NASA, John Miller, "Concerns with Randolph Vacuum Putty," EP-25 (84-35), April 12, 1984.

[51] NASA, John Miller, "Zinc Chromate Putty Installation in Nozzle to Case Joint Discrepancy," EP-25 (84-53), June 18, 1984.

[52] Thiokol, "Vacuum Putty Telecon," June 29, 1984.

to seat the primary O-ring. O-rings require some pressure from the working fluid, in the case of the SRM, this was gas, in order to seat properly and provide a effective seal. In practice, this design philosophy did not prove to be correct because the putty frequently held the pressure off of the O-rings, or if it did not, the putty had blowholes in it. It was then postulated that these holes might actually benefit the seating of the O-ring by allowing more pressure to reach it sooner. It was even suggested that holes might be deliberately made through the putty. However, it was then learned that blowholes served to concentrate propellant gas on small segments of the primary O-ring and caused the ring to erode.

The unacceptable heat erosion damage to both primary and secondary O-rings on SRM-16A resulted in an evaluation of the putty produced by Randolph Products. In July 1985, L. Thompson of MSFC made a presentation which noted that five different types of putty from four companies were under study in an effort to solve the putty performance problem. As late as 1985 twelve different types of tests had been performed and six more were in progress. The only putty to survive the water tests was General Sealants No. 43, which was a non-asbestos formulation. The Randolph putty had disintegrated in all three water tests. However, in comparing dynamic viscosity to temperature, the General Sealants product, at 25,000 poise,[53] was not viscous above 125 deg C. It was slightly better than the Randolph product and another product made by Inmont. The previously used Fuller-O'Brien product, however, increased in dynamic viscosity with an increase in temperature. It was 100,000 poise at 250 deg C, while it was less than 50,000 at 50 deg C.[54] Consequently, no product met all the design requirements as well as the Fuller-O'Brien did.

The Randolph putty is hydroscopic and its behavior is unsuited to use in the dry climate of Utah, as well as the humid climate of the Florida coast. In one case the putty was too stiff and in the other, too sticky. Since both factory and field joints required the use of the putty, a product with consistent performance in both climates was required.

The materials used in the manufacture of the O-rings was also critical to the safe operation of the Shuttle system. The O-rings had to be serviceable at the high temperatures in the joint which would result from heat transfer from the rocket combustion chamber. However, the use of NBR insulation around the propellant, and the use of putty, was to protect the steel casings and the O-rings from the direct heat of the propellant gases. This protection was not always successful when blowholes in the putty occurred, however, and the O-rings would frequently be damaged by heat. The lower temperatures that occur in Florida during the winter months was not covered by NASA's specifications. While elastomers are known to become brittle at low temperatures, a product specification sheet on Viton Fluroelastomer claimed, "Cold-VITON is generally serviceable in dynamic applications down to −18 to −23 deg C (0 to 10 deg F)."[55] The sheet added: "The brittle point of Viton at a thick-

[53] Poise is a measure of viscosity or resistance to flow.
[54] NASA, L.M. Thompson, "SRM/SRB Putty Evaluation," July, 1985.
[55] 3M-Chemical Division, "Viton Fluoroelastomer," Undated, p. 1.

ness of 0.075 inches is in the neighborhood of 50 deg F. Yet, as with other elastomers, thickness has a marked effect upon low temperature flexibility. Thinner cross-sections are more flexible than thicker ones at every temperature." The thickness of the O-rings on the Shuttle is 0.280 inches, thicker than the 0.075 inch article with a brittle point of 50 deg F noted above.[56] Consequently, the brittle point of Viton was misleading since the O-rings were much larger the the test specimen.

Military Specification MIL-R-83248A, 17 Feb. 84, "Rubber, Fluorocarbon Elastomer, High Temperature, Fluid, and Compression Set Resistant" set the specification for the O-rings that Thiokol had to meet.[57] They included:

> Type I-O-rings and compression seals Class I-75 +/−5 Hardness[58]

This specification then included other specifications issued by the Society of Automotive Engineers and the American Society for Testing Materials. One of the ASTM Specifications listed was ASTM 1329, "Evaluating Rubber Property, Retraction at Low Temperatures."[59] It was these referenced specifications which defined the significant characteristics required.

On February 6, 1979, Mr. William Ray of NASA's Marshall Space Flight Center, sent a memo to Messrs Hardy, Rice, Eudy, and McCool (See Appendix V-I). That memo was essentially a trip report of Mr. Ray's visits to the Precision Rubber Products Company and the Parker Seal Company, in search of information on the performance of O-rings. Some of the points covered in the memo were:

> The purpose of the visits was to present the O-ring seal manufacturers with data concerning the large O-ring extrusion gaps being experienced on the Space Shuttle Solid Rocket Motor clevis joints and to seek opinions regarding the potential risks involved.[60]
>
> With regard to the visit with company officials at Precision Rubber Products, "they voiced concern for the design, stating that the SRM O-ring extrusion gap was larger than that covered by their experience." [61]
>
> In response to the data presented to Parker Seal Company officials by Mr. Ray, Parker officials "also expressed surprise that the seal had performed so well in the present application." [62]
>
> Regarding the visit with the Parker officials, the memo stated, "their first thought was the O-ring was being asked to perform beyond its intended design and that a different type of seal should be considered." [63]

---

[56] Ibid.
[57] Department of Defense, "Military Specification: Rubber, Fluorocarbon Elastomer, High Temperature, Fluid, and Compression Set Resistant", MIL-R-83248A, February 17, 1984.
[58] Ibid, p. 1.
[59] Ibid., p. 3.
[60] NASA memorandum, William Ray, "Visit to Precision Rubber Products Corporation and Parker Seal Company", EP 25 (19-23), February 6, 1979, p. 1.
[61] Ibid.
[62] Ibid., p. 2.
[63] Ibid.

> The need for additional testing of the present design was also discussed and it was agreed that tests which more closely simulate actual conditions should be done.[64]

As a result of the foregoing data, the Committee has arrived at the specific Findings and Recommendations contained in Chapter V.

[64] Ibid.

VIII. LAUNCH OPERATIONS

INTRODUCTION

The purpose of this section is to document the series of decisions that culminated in the launch of STS 51-L on January 28, 1986. In Section A, the discussion details the Flight Readiness Reviews used to assess the mission's readiness, and also describes the teleconference on the night of January 27 when Thiokol engineers attempted to delay the launch. Also discussed are the circumstances surrounding the uncertainty represented by ice covering the launch pad's gantry.

Section B describes a specific example where the launch crew in the Firing Room waived a launch commit criterion. The discussions that took place on the subject indicate that the alternate procedure used as a justification for the waiver should not have been allowed, since the environmental conditions on the morning of January 28 were outside the limits specified for the alternate procedure.

A. THE STS 51-L LAUNCH DECISION

Discussion

Before each flight of the Space Shuttle, the ground support team carries out a series of meetings that are collectively known as the Flight Readiness Review. Policy guidance for this procedure is supplied by NASA Program Directive 710.5A, which states:

> It is the policy of the Associate Administrator for Space Flight (AA-SF) to make an assessment of mission readiness prior to each flight. This will be accomplished by a consolidated Flight Readiness Review (FRR) of all activities/elements necessary for safe and successful conduct of the launch, flight, and post-landing operations. . . . The FRR will be preceded by detailed readiness reviews (pre-FRR's) on individual elements, including cargo, under the cognizance of the responsible Managers.[1]

The FRR policy directive offers the following guidance to project and program managers regarding the expected content of their presentations:

> The Project/Element Managers will conduct pre-FRR's to develop their readiness assessment and are responsible for the FRR briefing content in their particular area.[2]

As for the agenda at these reviews, the directive has this to say:

---

[1] James Abrahamson, NASA, Headquarters, "Space Shuttle Flight Readiness Reviews," SFO-PD 710.5A, September 26, 1983, p. 1. See Appendix VIII-A.

[2] Ibid., pp. 1-2.

The presentation of agenda items will normally include a brief status summary with appropriate supporting detail on significant items and conclude with a readiness assessment. The presentation topics and scope should be developed from the pre-FRR's and should:

(1) be that required to provide the AA-SF with the information needed to make a judgment as to flight readiness;

(2) review recent significant resolved problems and prior flight anomalies when necessary to establish confidence;

(3) cover all problems, open items and constraints remaining to be resolved before the mission;

(4) establish the mission baseline configuration in terms of all significant changes since the last STS mission (changes to be considered include hardware, software, vehicle servicing/checkout, launch commit criteria, flight plans, flight rules and crew procedures);

Within the above guidelines, the scope of the review should cover status and issues in areas such as: vehicle checkout, shortages and open work, unexplained anomalies, hardware failures, prior flight anomalies, certification/verification, as-built hardware configuration versus certified hardware list, Critical Items List (CIL), development, qualification and reliability testing, waviers and deviations, limited life components, launch critical spares, sneak circuits, system safety/hazards and flight margins. . . .[3]

In the case of STS 51-L, no deviation from normal procedure apparently occurred. This means that the Solid Rocket Motor, containing the seal that apparently failed, proceeded through the usual eight levels of review at Thiokol's Wasatch Division, Marshall Space Flight Center, the STS Program Office at Johnson Space Flight Center, and the Associate Administrator's review at NASA Headquarters.[4]

The Flight Readiness Review process for STS 51-L began on December 11, 1985, at Thiokol's Utah plant. No information is presented in the briefing charts used that day regarding the continuing failure of the SRM joint seals. The chart entitled "STS-61C (STS-32) (SRM-24) Performance "has only one entry: "TBD [to be determined]." [5]

Post-flight disassembly of STS 61-C SRB hardware following its launch on January 12 revealed that erosion of the primary O-ring had occurred in the aft field joint of the left motor. Hot gas had also bypassed the primary seal in the left nozzle joint. Erosion of the primary seal had also occurred in the nozzle joint of the right motor.[6]

Under the terms of the FRR Policy Directive, such damage would appear to require discussion: "the scope of the review should

---

[3] Ibid., pp. 2-3.

[4] Table I (page 44) indicates the date and scope for each of these eight reviews.

[5] Thiokol, "STS-51L (STS-33) Solid Rocket Motor (SRM-25) Flight Readiness Review," TWR-15380, December 11, 1985, chart 1-1. See Appendix VIII-B.

[6] Rogers Commission Report, Volume II, p. H-3.

cover status and issues in areas such as . . . prior flight anomalies. . . ." [7]

However, according to Mr. McDonald and Mr. Kennedy, Thiokol normally took about one week to prepare a discussion of problems noted in the initial inspection following SRB hardware disassembly.[8]

It would seem logical, when faced with the lack of data from the previous hardware set, to expand the search to other previously-flown hardware. On 61-A, hot gas had bypassed the primary seals in both the center and aft field joints of the left motor.[9]

The right motor suffered erosion of the primary O-ring in the nozzle joint.[10]

The SRBs from 61-B suffered erosion of the seals in both nozzle joints, with gas bypassing the primary seal of the left nozzle.[11]

The Associate Administrator's policy directive is not alone in stressing that any available information capable of assisting with an assessment of flight readiness should be presented at a readiness review. Marshall's Shuttle Projects Office policy guidance uses virtually identical language. Under "Shuttle Policy Guidance," it states, "*Review Concept*: The Shuttle Projects FRR will employ a delta review concept from prior reviews and previous STS missions." [12]

In his letter announcing the STS 51-L Marshall Center FRR, Dr. Lucas wrote:

> Each project manager must certify the flight readiness of his hardware and present supporting rationale and data so the Board can independently assess the flight readiness . . . Emphasis will be placed on safety of flight and mission success, including potential impact of prior flight anomalies.[13]

Apparent in the STS 51-L process, however, is that the continuing SRM seal problem did not receive such treatment. The "delta review concept" referred to above, according to Mr. McDonald, meant that the contractor was obligated to step back only to the previous mission for comparison.[14]

For 51-L, there was no previous mission to compare data with, since 61-C had not yet flown. Anomalies on STS 61-A and 61-B were not discussed, Mr. McDonald said, because they had already been dispositioned in the FRR's for 61-B and 61-C.[15]

The Marshall Space Flight Center FRR conducted by Dr. Lucas occurred only one day after the 61-C launch. Mulloy's presentation

---

[7] SFO-PD 710.5A, p. 3.
[8] Discussions with Allan McDonald and Carver Kennedy, Thiokol (Wasatch Operations), Brigham City, Utah, September 4, 1986.
[9] Rogers Commission Report, *loc. cit.*
[10] Ibid.
[11] Ibid.
[12] Robert Lindstrom, NASA, Marshall Space Flight Center, "Shuttle Project Flight Readiness Review," SOP 8000.1, December 29, 1983, p. 2. See Appendix VIII-C.
[13] William Lucas, NASA, Marshall Space Flight Center, "MSFC Flight Readiness Review (FRR) Board for MSFC Elements for Mission 51-L," January 7, 1986, pp. 1-2. See Appendix VIII-D.
[14] Discussion with Mr. McDonald, September 4, 1986.
[15] Ibid.

under STS 61-C performance noted that "all SRB systems functioned normally."[16]

Under "ascent," the chart shows "no anomalies."[17]

There is no indication in the documentation for this FRR that the continuing problem with the SRM seals was raised. The parachute recovery system was discussed at some length. Much of the presentation appears to be drawn from the booster assembly presentation made at Mulloy's Level III FRR on January 3. The only relevant item that might refer to the O-rings appears under "Certification/Verification Status," where Mulloy stated that there were "no findings from continuing analyses that changes previously established rationale for flight."[18]

Mr. Mulloy's presentation at the January 15, 1986, Level 1 FRR does not indicate any serious problems with the SRB's. Documentation under "Problems/Anomalies" lists "[n]o 61-C flight anomalies."[19]

Again, the focus of his presentation involved the changes made in the parachute recovery system. The SRM booster nozzle on STS 51-L would be separated at the apogee of the SRB flight path (following separation of the boosters from the Shuttle vehicle) to protect the drogue parachute from debris, and the main parachutes were to be separated at water impact to reduce risks to the divers that assisted with recovery.[20]

Mulloy's presentation to the Associate Administrator was not noticeably different from the presentation be made to Mr. Aldrich, the Shuttle Program Manager, at the Level II readiness review the day before. In fact, the briefing charts are identical.[21]

SRM seal erosion was ultimately raised during the STS-51L Flight Readiness Review cycle. Mr. McDonald stated that at the L-1 FRR Mr. Mulloy informed the Mission Management Team of the erosion damage seen on STS 61-C, characterizing it as "within the experience base."[22]

This evaluation of the seal erosion problem does not indicate the seriousness of the issue, and would not lead senior managers to a conclusion that the seal problem was a threat to the safety of the Shuttle. Given the historical treatment of the SRM seal problem in this process, however, it is not surprising that the STS 51-L reviews did not raise any new concerns about the integrity of the joint.

This point is readily apparent in the Commission's report. There is no implication that a serious problem exists, if Mr. Mulloy's presentations are examined. The presentation made to Level 1 during the STS 41-C FRR indicated that erosion was "acceptable," and offered a rationale for accepting the possibility that the phenomenon would recur.[23]

---

[16] Larry Mulloy, NASA, Marshall Space Fight Center, "Center Board: STS-51L Flight Readiness Review Solid Rocket Booster," January 13, 1986, Chart SRB-3.

[17] Ibid.

[18] Ibid, Chart SRB-28.

[19] Larry Mulloy, NASA, Marshall Space Flight Center, "STS-51L Level I Flight Readiness Review," January 15, 1986, Chart SRB-3. See Appendix VIII-E.

[20] Ibid., Chart SRB-4.

[21] Larry Mulloy, NASA, Marshall Space Flight Center, "STS-51L Level II Flight Readiness Review," January 14, 1986. See Appendix VIII-F.

[22] Discussion with Allan McDonald, September 4, 1986.

[23] Rogers Commission Report, Volume II. See Chart 15 (p. H-10) and Chart 19 (p. H-12).

Indeed, Level 1 displayed more concern about the problem than did Marshall, since Dr. Hans Mark, then Deputy Administrator, directed Marshall to prepare a review of the seal erosion problems.[24]

For mission 41-G, Mulloy argued that "test shows maximum erosion possible less than erosion allowable."[25]

In his presentation to the STS 51-E Level 1 FRR, Mulloy is seen on a videotape stating that:

> The rationale that was developed after observing this erosion on STS-11 [41-B] was that it was a limited duration that was self-limiting in that as soon as the pressure in the cavity between the putty and the primary O-ring after the primary O-ring seats, or the pressure between the primary and the secondary O-ring equals the motor pressure, the flow stops and the erosion stops. The maximum erosion that we have seen previously is 53 thousandths [of an inch]—that was back on STS-2. The erosion that we saw on 51-C was 10 thousandths of an inch on one O-ring and 38 thousandths on the other, so we believe that because of the limited exposure and the fact that the leak check assures that the secondary O-ring is properly sealing against motor pressure and the fact that the duration is limited, and that we can take 95 thousandths erosion on a primary O-ring and seal against 3,000 psi which is three times the motor pressure, that this represents an acceptable risk.[26]

Mulloy's confidence that the SRM seal could take "95 thousandths erosion on a primary O-ring and seal against 3,000 psi which is three times the motor pressure, . . ." is based on computer modelling of the joint performance. Dr. Feynman's analysis of the model, however, questions its use as the basis for declaring the seal problem "an acceptable risk."

> . . . This was a model based not on physical understanding but on empirical curve fitting. To be more detailed, it was supposed a stream of hot gas impinged on the O-ring material, and the heat was determind at the point of stagnation (so far, with reasonable physical thermodynamic laws). But to determine how much rubber eroded it was assumed this depended only on this heat by a formula suggested by data on a similar material. A logarithmic plot suggested a straight line, so it was supposed that the erosion varied as the .58 power of the heat, the .58 being determined by a nearest fit. At any rate, adjusting some other numbers, it was determined that the model agreed with erosion (to depth of one-third the radius of the ring). There is nothing much so wrong with this as believing the answer! Uncertainties appear everywhere. How strong the gas stream might be was unpredictable, it depended on holes formed in the putty. Blow-by showed that the ring might fail even though not, or only partially eroded

[24] Ibid., p. H-13.
[25] Ibid., Chart 30 (p. H-18).
[26] Ibid., p. H-42.

> through. The empirical formula was known to be uncertain, for it did not go directly through the very data points by which it was determined. There were a cloud of points some twice above, and some twice below the fitted curve, so erosions twice predicted were reasonable from that cause alone. Similar uncertainties surrounded the other constants in the formula, etc., etc. When using a mathematical model careful attention must be given to uncertainties in the model.[27]

Mr. Mulloy's analysis of erosion also notes that it was a "self-limiting" phenomenon, assuming that the damage ceased after the pressure built up against the seal. His analysis demonstrates that either the primary or the secondary seal would serve the purpose.

On December 17, 1982, an amended version of the SRB Critical Items List was approved. It stated, "Leakage of the primary O-ring seal is classified as a single failure point due to possibility of loss of sealing at the secondary O-ring because of joint rotation *after motor pressurization.* [emphasis added]"[28]

In the "Rationale for Retention," the document states, "Full redundancy exists at the moment of initial pressurization."[29]

Mr. Mulloy read this to indicate that during the ignition transient, the seal was a Critically 1R system, a redundant seal existed, and the secondary O-ring could be relied upon. After completion of the ignition transient (approximately 600 milliseconds), the joint became a Critically 1 system.[30]

Congressman Roe, however, said,

> We don't buy the point of view, do we measure other criticality points in degrees? My father taught me . . . [i]t is or it isn't . . . you took it from a R1 position and made it a number one position. You didn't qualify that, there is nothing in the record that qualifies it as half an R1 or three-quarters of an R1 in terms of temperature. . . . We didn't say we put them in there in number of degrees. We either did or we didn't.[31]

Implied in the presentation by Mr. Mulloy is that the Marshall and Thiokol engineers understood the joint's performance during the ignition transient. But, as Congressman Volkmer noted,

> Mr. VOLKMER. ". . . Mr. Mulloy, it says on page 148 [of the Commission's report] that prior to the accident neither NASA or Thiokol clearly understood the mechanism by which the joint sealing took place. Do you agree or disagree with that?"
>
> Mr. MULLOY. "I totally agree, sir."[32]

Also notable by its absence in Mulloy's presentation is the fact that STS 51-L had demonstrated an extreme example of blowby in the nozzle joint. It was this case that led Thiokol engineers to the

---

[27] Rogers Commission Report, Volume II, p. F-2.
[28] NASA, "SRB Critical Items List," Page A-6A, December 17, 1982, Sheet 1.
[29] Ibid.
[30] Rogers Commission Report, Volume V, p. 834.
[31] Cmte Hgs., Transcript, June 17, 1986, p. 185.
[32] Ibid., p. 291.

conclusion that temperature was a contributing factor to joint damage. Mr. Boisjoly explained to General Kutyna that, based on photographs of the joints on 51-C (launched at a seal temperature of 53°F) and 61-A (launched at a seal temperature of 75°F), he "concluded, and so presented on the night before the launch . . . that it was telling us that temperature was indeed a discriminator . . ."[33]

The appearance of the material that had bypassed the joint, according to Boisjoly, was significantly worse for 51C in appearance and extent.

For mission 51-F, even after the failure of the primary seal in the 51-B nozzle joint, Mulloy's presentation to Level 1 listed the problem as "closed."[34]

Chairman Roe, questioning witnesses from NASA on 17 June, learned that managers at Johnson and at Headquarters had not necessarily perceived the seriousness of the situation represented by the seal problem.

> Mr. ROE. "I would like to get Mr. Mulloy to answer the question—would you repeat the nine flights and tell the Committee at what level the O-ring problem was discussed and who was at that level? . . . You mentioned again, you listed the whole nine, and tell the Committee at what level the O-ring problem was discussed. We have been going on this for seven years and then who was at that meeting?"
>
> Mr. MULLOY. "Yes sir. I can answer part of your question. . . . I am reading from what was provided to me. It looks like it fits within the erosion. STS-11 [41-B], 41-C, 41-G, 51-E . . ., 51-F . . . , 51-I, 51-J, 61-A . . ., and 61 Bravo."
>
> Mr. ROE. "These were a problem with the O-rings and they were discussed at Level 1?"
>
> Mr. MULLOY. "Level 1 and Level 2."
>
> Mr. ROE. "Therefore it is inconceivable that Level 1, which is top management, would not have understood the issue?"
>
> Mr. MULLOY. "That is right, and I believe that has been acknowledged. . . ."
>
> Dr. GRAHAM. "We are in fact, reviewing the records to see who was at the various Flight Readiness Reviews that occurred when the O-ring data was mentioned, and we have not yet been able to pull that together. . . ."
>
> Mr. ROE. "So what you are basically saying is that Washington level knew of part of the problems; is that a fair comment?"
>
> Dr. GRAHAM. "There are two pieces to this: one, what was transmitted; and what was understood. I believe what Mr. Mulloy and Dr. Lucas are addressing is what was transmitted. I don't know that they are the most appropriate people to express what was understood. That was a Headquarters issue and, in some cases, a Johnson Space

[33] Rogers Commission Report, Volume V, p. 784.
[34] Ibid., Chart 130 (p. H-66).

> Center issue. It is clear the issue was not perceived at the seriousness with which it actually affected the system. However, the information was transmitted to these agencies." [35]

Dr. Graham's statement is important when discussing the August 19, 1985, briefing on the joint seal problem. Thiokol and Marshall personnel did not communicate that the situation required a halt in operations until the problem of seal erosion had been solved. It should also be noted that Mr. Moore was then occupied with the failure of the SSME temperature sensors (a failure which had led to premature shutdown of a main engine on flight STS 51-F and caused the first abort-to-orbit in the program's history), and so the briefing was attended by Mr. L. Michael Weeks, Deputy Associate Administrator (Technical) for Space Flight. A more complete analysis of his descripton of the situation to Mr. Moore is discussed in the section on Technical Expertise.[36]

Mr. Aldrich was not made aware of the briefing at all, removing Level 2 from the information flow.[37]

Levels 1 and 2 were not alone in their misapprehensions. the lack of understanding of the seal problem also appears in the presentations to Dr. Lucas and Mr. Reinartz at Marshall made by Mr. Mulloy. In the STS 61-C FRR cycle, immediately preceding 51-L, Mr. Mulloy was given an extensive discussion of the information obtained from STS 61-B, describing the damage to the seals, at the Level 3 SRB Project Office briefing he chaired.[38]

Mulloy's presentation to the Shuttle Projects Board then noted "SRM joint O-ring performance within experience base."[39]

In his presentation to the Level 1 FRR, however, Mulloy stated there were "no 61-B flight anomalies."[40]

In hindsight, a fundamental error that pervades the history of the seal erosion problem is this reliance on the "experience base" argument. Unwarranted confidence existed in the analysis of the joint seal erosion problem developed by Thiokol engineers and agreed to by Marshall's program office. There is a vital lesson to be learned in this episode, and it is best expressed by Henry Petroski, from the School of Engineering at Duke University.

> *Dismissing the single structural failure as an anomaly is never a wise course* (emphasis added). The failure of any engineering structure is cause for concern, for a single incident can indicate a material flaw or design error that renders myriad structural successes irrelevant. . . . In en-

---

[35] Cmte Hgs, Transcript, June 17, 1986, pp. 203-206.

[36] Section VI.B.1.c. of this report.

[37] "The second breakdown in communications. . . ," Mr. Aldrich testified before the Commission, "is the situation of the variety of reviews that were conducted last summer between the NASA Headquarters Organization and the Marshall Organization on the [joint seal problem] and the fact that that was not brought through my office in either direction—that is, it was not worked through by the NASA Headquarters Organization nor when the Marshall Organization brought these concerns to be reported were we involved. And I believe that is a critical breakdown in process and I think it is also against the documented reporting channels that the program is supposed to operate to." Rogers Commission Report, Volume V, p. 1490.

[38] Larry Wear, NASA, Marshall Space Flight Center, "Flight Readiness Review SRM-24 (STS 61-C)," December 2, 1985, Charts 3-2; 3-2B. See Appendix VIII-G.

[39] Rogers Commission Report, Volume II, p. H-3.

[40] Ibid.

> gineering, numbers are means, not ends, and it ought rightly to have taken only the failure of a single bridge to bring into question the integrity of every other span. . . . The common expectation of the engineer and the layman is that the road will not lead to bridges that collapse.[41]

Clearly evident is the fact that the Flight Readiness Review procedure cannot compensate for poor engineering analysis. The FRR is similar to a checklist and will not necessarily discover problems not on the list. The technical rationale presented by Level 3 managers is assumed to reflect the best engineering judgment available. Relying on this expertise, managers in more senior positions at NASA were misinformed regarding the severity of the problem of seal erosion and its critical importance to flight safety.

The Committee is also concerned about the so-called "launch constraint" imposed on the Shuttle system following STS 51-B, and the role this constraint was expected to play in the Flight Readiness Review process. The term "launch constraint" would seem to indicate that the Shuttle should not be launched until the problem giving rise to the constraint was solved. This is apparently not the case, according to Mr. Mulloy:

> The problem assessment system is in place at the Marshall Center as a tool to assure—it is a tool used by our quality and a reliability assurance organization to assure that problems that occur in flights and in ground test, in development, our qualification motor tests that would have a bearing on the flight or the upcoming flights, that that is documented and tracked. That problem assessment system shows in the case of the O-ring erosion, it shows essentially the same information, in many cases identical information to what is in the Flight Readiness Reviews. It is the basis for continuing to fly given the observations we are seeing.[42]

Testifying before the Commission, Mulloy had also made this distinction.

> Chairman ROGERS. "Let's go back just a bit, because I think it is helpful to me if you—you use words that I understand a little bit. What caused the constraint to be put on in the first place?"
>
> Mr. MULLOY. "The constraint was put on after we saw the secondary O-ring erosion on the nozzle, I believe."
>
> Chairman ROGERS. "Who decided that?"
>
> Mr. MULLOY. "I decided that, that that [the joint seal erosion] would be addressed, until that problem was resolved, it would be considered a launch constraint, and addressed at Flight Readiness Reviews to assure that we were staying within our flight experience base. . . ."
>
> Dr. RIDE. "Why didn't you put a launch constraint on the field joint at the same time?"

---

[41] Henry Petroski, *To Engineer is Human: The Role of Failure in Successful Design* (New York: St. Martin's Press, 1985), pp. 69–73.

[42] Cmte Hgs, Transcript, June 17, 1986, p. 205.

> Mr. MULLOY. "I think at that point . . . the logic was that we had been discussing the field joint, the field and nozzle joint primary O-ring erosion. This erosion of [STS 51-B's] secondary O-ring was a new and significant event, very new and significant event that we certainly did not understand. Everything up to that point had been that the primary O-ring, even though it does experience some erosion, does seal. What we had evidence of was that here was a case where the primary O-ring was violated and the secondary O-ring was eroded, and that was considered to be a more serious observation than previously observed." [43]

The Marshall Space Flight System Problem Assessment System (PAS) was tracking the problem of nozzle joint primary O-ring erosion in Record A09288, "O-Ring Erosion in the Case to Nozzle Joint."[44] The record was apparently opened on July 10, 1985, some two months following the launch of STS 51-B on April 29, 1985. The last entry in this record is dated January 23, 1986, and begins "Resolution." It continues with a rationale for closing out the tracking record. Part of the rationale is quoted here:

> Analytical studies based on both impingement erosion and blowby erosion show that this phenomenon has an acceptable ceiling since implementing the above changes [in performance of the seal leak check and in stacking procedures]. Recent experience has been within the program data base. The seal improvement program plan will continue until the problem has been isolated and damage eliminated to the SRM seals.[45]

An identical entry appears in PAS Record A07934, "Segment Joint Primary O-ring Charred." Though tracking problems with the field joint seals, work on the nozzle joint problem was included "as they are the same generic problem."[46]

The logic behind this "resolution" of the O-ring problem is not readily apparent. As the field joint tracking report notes, "The O-rings in the SRM segment ass[embl]y joints are designed as press[ure] seals & are not intended to be exposed to hot gases."[47]

Yet, in the rationale for closing out these tracking reports, it is stated that "[p]rimary O-ring erosion is expected to continue since no corrective action has been established that will prevent hot gases from reaching the primary O-ring cavity."[48]

The rationale ad described also appears to violate the directive, issued in 1980, that addresses the question of launch constraints. "All open problems coded criticality, 1, 1R, 2 or 2R," it stated:

> will be considered launch constraints until resolved (recurrence control established and its implementation effectivity determined) or sufficient rationale, i.e., different con-

---

[43] Rogers Commission Report, Volume V, p. 1510.
[44] NASA, Marshall Space Flight Center, "O-Ring Erosion in the Case to Nozzle Joint," Problem Assessment System Record Number A09288, February 26, 1986. See Appendix VIII-H.
[45] Ibid., p. 2.
[46] NASA, Marshall Space Flight Center, "Segment Joint Primary O-ring Charred," Problem Assessment System Record Number A07934, March 7, 1986, p. 4. See Appendix VIII-I.
[47] Ibid., p. 1.
[48] Ibid., p. 5.

figuration, etc., exists to conclude that this problem will not occur on the flight vehicle during prelaunch, launch or flight.[49]

The Committee attributes this situation to the concept of "acceptable erosion," which is more fully discussed in Chapter VII of this report.

According to testimony before the Commission, these tracking records for O-ring erosion were closed out upon receipt of a letter from Mr. McDonald dated December 10, 1985.[51]

This was apparently a mistake. As Mr. Mulloy and Mr. Wear explained to the Commission,

> Mr. MULLOY. ". . . Now, the entry that is shown in there that the problem was closed prior to 51-L is in error. What happened there was, one of your documents here which we did not discuss is the letter from Mr. McDonald to Mr. Wear which proposed that this problem be dropped from the problem assessment system and no longer be trapped [tracked] for the reasons stated in Mr. McDonald's letter. That letter was in the review cycle . . . . After Mr. Wear brought this letter to my attention, my reaction was, 'we are not going to drop this from the problem assessment system because the problem is not resolved and it has to be dealt with on a flight-by-flight basis.' Since that was going through the review cycle, the people who run this problem assessment system erroneously entered a closure for the problem on the basis of this submittal from Thiokol. Having done that, then for the 51-L review, this did not come up in the Flight Readiness Review as an open launch constraint, so you won't find a project signature because the PAS system showed the problem was closed, and that was an error."
>
> Chairman ROGERS. "Who made the error? Do you know?"
>
> Mr. MULLOY. "The people who do the problem assessment system."
>
> Mr. WEAR. "Mr. Fletcher, and he reports within our quality organization at the Flight Readiness Review, at the incremental Flight Readiness Reviews . . . . At my review and at Larry's review, there is a heads up given to the quality representative at that board for what problems the system has open, and they cross-check to make sure that we address that problem in the readiness review. On this particular occasion, there was no heads up given because their PAS system considered that action closed. That is unfortunate." [52]

Mr. Mulloy's discussion with Chairman Roe, and his description provided to the Commission, indicate that the NASA Safety, Reliability and Quality Assurance (SR&QA) organization should play a

---

[49] Robert Lindstrom, NASA, Marshall Space Flight Center, "Assigning Launch Constraints on Open Problems Submitted to MSFC PAS," September 15, 1980, p. 1. See Appendix VIII-J.
[51] Rogers Commission Report, Volume V, p. 1509.
[52] Ibid.

significant role in the Flight Readiness Review process. The information available on this topic suggests that this was not the case. The Committee is concerned by the invisibility of SR&QA in this area.[53]

First, the number of people operating in the SR&QA organization has apparently been declining since the Apollo program. The decline at Marshall is among the most severe, according to NASA's internal estimate.[54]

The obvious conclusion is that SR&QA has fewer people to oversee the myriad details involved in preparing for Flight Readiness Reviews, in addition to their other duties.

Second, there is nothing to show what evaluation SR&QA personnel had made of the joint seal erosion problem. The recurring nature of this problem, and the Criticality-1 status of the joint, argue that SR&QA should have paid close attention to this situation, including the execution of an independent test program and presentations of their evaluation at Level 3 Flight Readiness Reviews.

Third, though management of the PAS system is apparently the responsibility of the SR&QA organization, they appear to exercise little control. The system is operated under contract by Rockwell, data is entered by hardware manufacturers, and the only technical analysis that appears in the reports on joint seal erosion was developed by Mr. Wear or his deputy, James Thomas, in the SRM program office. There is no input, either concurrence or dispute, registered by SR&QA personnel. Even more important, as illustrated by Mr. Mulloy's testimony, problem reports can too easily be removed from the system.

Fourth, the PAS tracking records do not support the testimony of Mr. Mulloy and Mr. Wear before the Commission. The entry entitled "Resolution" is dated January 23, 1986, while the FRR in Mr. Wear's SRM Project Office occurred on December 17, 1985, and the FRR in Mr. Mulloy's SRB Project Office took place on January 3, 1986. It is also interesting to note that the PAS Record, dated February 26, 1986, shows "Status Open."[55]

If Mr. Mulloy's and Mr. Wear's testimony is accurate, SR&QA should still have raised the issue of seal erosion as a concern at their reviews. No evidence exists that this occurred.

The Committee, however, is concerned not only about the SR&QA organization at Marshall. The Commission noted in its report that "[t]he Problem Reporting and Corrective Action document (JSC 08126A, paragraph 3.2d) requires project offices to inform Level II of launch constraints. That requirement was not met. Neither Level II nor Level I was informed."[56]

Testifying before the Committee, however, Mr. Mulloy argued that both levels were informed.

> Mr. VOLKMER. "Even though . . . you continued to see erosion of the O-ring, you continued to waive the launch constraint?"

---

[53] See also "Safety, Reliability and Quality Assurance," Section VI.B.2.c(2) of this report.
[54] This is documented in Section VI.B.2.c(1) of this report.
[55] PAS Record A09288, p. 1.
[56] Rogers Commission Report, Volume I, p. 159.

Mr. MULLOY. "That is correct, sir, on the basis of the rationale or the explanation as to why that was an acceptable risk that was presented to me by Morton Thiokol, reviewed and approved by my management. . . ."

Mr. VOLKMER. "Was JSC [the Johnson Space Center] informed of the problem?"

Mr. MULLOY. "Through the Flight Readiness Review and through the submission of this problem to the problem tracking system at JSC. I do not know what distribution was made at JSC when it goes down there. The report also goes to the Chief Engineer's office at Headquarters."

Mr. VOLKMER. "It is my understanding that we had some testimony earlier from Mr. Aldrich that he wasn't knowledgeable that there was a launch constraint."

Mr. MULLOY. "That is entirely possible, sir, I don't know what distribution was made, and I have testified, and I think—I have testified that it wasn't briefed in the Levels 2 and the Level 1. When I went—"

Mr. VOLKMER. "That is right."

Mr. MULLOY. "—that 'we have a problem, the concern is flight safety, the rationale for continuing to fly is this.' That was not briefed in the context of 'this is a launch constraint in the problem assessment system,' and it is entirely possible that if that report, whatever distribution is made of that report at Houston, that he might not have seen that."[57]

It is quite likely that neither Mr. Moore nor Mr. Aldrich was made aware of the launch constraint on the SRM. SR&QA personnel at Johnson and at NASA Headquarters should have received these reports described by Mr. Mulloy and tracked them as they did similar launch constraints on other Shuttle hardware. What steps they took to assure that these constraints were raised at the Flight Readiness Review for these management levels is less clear. The Committee's review of the FRR's for the six missions subject to the launch constraint on SRM nozzle joint seals does not indicate a greater level of discussion took place because the constraint was in force, except for the STS 51-F FRR where an explanation of the 51-B failure was required.

The rationale for closing out these problems on the problem assessment system stated that "status will continue to be provided in the Flight Readiness Reviews and in formal technical reviews at Thiokol and MSFC." [58]

Without a change in the prevailing technical evaluation of the problem, however, it is unlikely that proper action to correct the overall problem would have been undertaken.

On the eve of the launch, Thiokol engineers attempted to change this prevailing technical evaluation. In a teleconference on the night of January 27, they presented data demonstrating that the low temperatures in the area would impair the function of the O-ring seals inside the joint. After NASA managers expressed con-

[57] Cmte Hgs Transcript, June 17, 1986, pp. 284-85.
[58] PAS Record A09288, p. 3.

cern about the delay such constraints would have on the flight schedules, Thiokol's management withdrew and met with their engineers again; only this time, the managers would not listen to further argument about aborting the flight. Thiokol then recommended that the launch be allowed to proceed.

Launch operations were terminated at 12:36 p.m. Eastern Standard Time (EST) on January 27 because of high crosswinds at the launch site. At 2:00 p.m. EST, the Mission Management Team met and decided to attempt a launch at 9:38 a.m. EST January 28. The weather was expected to be clear but cold with temperatures in the low twenties. There were concerns about the facilities and various water drains but no concerns were expressed about the O-ring and the Solid Rocket Boosters. All members of the team were asked to review the situation and call if any problems arose.

At Thiokol's Wasatch Division in Utah, Mr. Robert Ebeling [59] met with Mr. Boisjoly at about 2:30 p.m. Mountain Standard Time (MST) on the afternoon of January 27. They were joind by other Thiokol engineers. Mr. Ebeling was concerned about predicted cold temperatures at the Kennedy Space Center. When he was questioned by the Rogers Commission he responded:

> . . . The meeting lasted one hour, but the conclusion of that meeting was engineering, epecially Arnie [Thompson], Roger Boisjoly, Brian Russell, myself, Jerry Burns, they come to mind, were very adamant about their concerns on this lower temperature, because we were way below our data base and we were way below what we qualified for.[60]

Later Mr. Ebeling called Mr. McDonald at the Kennedy Space Center. Mr. McDonald remembered the call, saying:

> He called me and said they had just received some word earlier that the weatherman was projecting temperatures as low as 18 degrees F [Fahrenheit] sometime in the early morning hours of the 28th and that they had some meeting with some of the engineering people and had some concerns about the O-rings getting to those kinds of temperatures.[61]

Mr. Ebeling wanted Mr. McDonald to get some accurate predicted temperatures for the Cape so he could make some calculations to determine what could be expected of the O-rings. McDonald told him he would get the temperature data for him and call him back. Mr. Carver Kennedy, Vice President of Space Services for Thiokol, working at the Kennedy Space Center, obtained the information. Mr. McDonald then relayed the information to Mr. Ebeling in Utah. The information indicated that the temperature was to get as low as 22° in the early morning hours, probably around 6:00 a.m., and that they were predicting a temperature of about 26° at the intended time of launch, 9:38 a.m. on the 28th.[62]

---

[59] Solid Rocket Motor Igniter and Final Assembly Manager, Thiokol.
[60] Thiokol, Robert Ebeling, Interview before the Presidential Commission on the Space Shuttle Challenger Accident, March 19, 1986, p. 13.
[61] Rogers Commission Report, Volume IV, p. 715.
[62] Ibid.

Mr. McDonald then called Mr. Cecil Houston, the Resident Manager for the Marshall office at Kennedy Space Center, and told him of Thiokol's concerns with the low temperature and potential problems with the O-rings. Mr. Houston said he would set up a teleconference including Marshall and personnel at Thiokol in Utah.[62a]

[62a] Table II lists the principal participants in the teleconference on January 27, 1986.

TABLE II

PRINCIPAL PARTICIPANTS IN THE TELECONFERENCE
ON JANUARY 27, 1986

	At Kennedy Space Center, Florida	
Thiokol	Allan McDonald	Director, Solid Rocket Motor Program Office
Thiokol	Jack Buchanan	Manager, KSC Resident Office
MSFC	Lawrence Mulloy	Manager, Solid Rocket Booster Project Office
MSFC	Stanley Reinartz	Manager, Shuttle Projects Office
MSFC	Judson Lovingood	Deputy Manager, Shuttle Projects Office
	At Marshall Space Flight Center, Alabama	
MSFC	George Hardy	Deputy Director for Science and Engineering
	At Thiokol Wasatch Operations, Utah	
Thiokol	Jerald Mason	Senior Vice President, Wasatch Operations
Thiokol	C.G. Wiggins	Vice President and General Manager, Space Division
Thiokol	Robert Lund	Vice President for Engineering
Thiokol	Joseph Kilminster	Vice President, Space Booster Programs
Thiokol	Roger Boisjoly	Staff Engineer, Applied Mechanics
Thiokol	Robert Ebeling	Manager, SRM Ignition System, Final Assembly, Special Projects and Test
Thiokol	Arnold Thompson	Supervisor, Structures Design

Mr. Houston then called Dr. Judson Lovingood, Deputy Shuttle Project Manager at Marshall, to inform him of the concerns about the O-rings. Mr. Houston asked Dr. Lovingood to set up a teleconference with senior project management personnel, including Mr. George Hardy, Deputy Director of Science and Engineering at Marshall, and with Thiokol personnel. Dr. Lovingood called Mr. Stanley Reinartz, Marshall's Shuttle Project Office Manager, a few minutes later and informed him of the planned teleconference.

The first phase of the teleconference began at 5:54 p.m. EST and included Messrs. Reinartz, Lovingood, Hardy, and others at Kennedy, Marshall and Thiokol's Wasatch plant. Concerns about the effect of low temperature of the O-rings and the joint seal were presented by Thiokol personnel, along with an opinion that launch should be delayed.

A recommendation was also made that Arnold Aldrich, the Space Transportation System Program Manager, be told of the upcoming telecon and that the fact that Thiokol had expressed some concerns. Mr. Reinartz testified before the Commission that "we did not have a full understanding of the situation as I understood it at that time, and felt that it was appropriate to do before we involved the Level II into the system."[63]

Testifying before the Rogers Commission, Dr. Lovingood was asked whether the possibility of a launch delay had been mentioned in this telcon on January 27. Dr. Lovingood replied:

> That is the way I heard it, and they were talking about the 51-C experience and the fact that they had experienced the worst case blow-by as far as arc and the soot and so forth. And also, they talked about the resiliency data that they had.
>
> So it appeared to me—and we didn't have all the people there. That was another aspect of this. It appeared to me we had better sit down and get the data so that we could understand exactly what they were talking about and assess that data.
>
> And that is why I suggested that we go ahead and have a telecon within the center, so that we can review that.[64]

Dr. Keel, the Staff Director for the Rogers Commission, asked,

> Dr. KEEL. "So as early as after that first afternoon conference at 5:45, it appeared that Thiokol was basically saying delay. Is that right?"
>
> Dr. LOVINGOOD. "That is the way it came across to me. I don't know how other people perceive it, but that's the way it came across to me."
>
> Dr. KEEL. "Mr. Reinartz, how did you perceive it?"
>
> Mr. REINARTZ. "I did not perceive it that way. I perceived that they were raising some questions and issues which required looking into by all the parties, but I did not perceive it as a recommendation to delay."
>
> Dr. KEEL. "Some prospect for delay?"

---

[63] Rogers Commission Report, Volume V, p. 919.
[64] Ibid., p. 923.

Mr. REINARTZ. "Yes, sir, that possibility is always there."

Dr. KEEL. "Did you convey that to Mr. Mulloy and Mr. Hardy before the 8:15 teleconference?"

Mr. REINARTZ. "Yes I did. And as a matter of fact, we had a discussion. Mr. Mulloy was just out of communication for about an hour, and then after that I got in contact with him, and we both had a short discussion relating to the general nature of the concerns with Dr. Lucas and Mr. Kingsbury at the motel before we both departed for the telecon that we had set up out at the Cape."[65]

At approximately 8:45 p.m. EST, the second phase of the teleconference commenced, Thiokol's charts and written data having arrived at the Kennedy Space Center by telefax. The charts presented a history of the O-ring erosion and blow-by in the Solid Rocket Booster joints of previous flights, presented the results of subscale testing at Thiokol and the results of static tests of Solid Rocket Motors.

Mr. Boisjoly testified:

I expressed deep concern about launching at low temperature. I presented Chart 2-1 with emphasis—now, 2-1, if you want to see it, I have it, but basically that was the chart that summarized the primary concerns, and that was the chart that I pulled right out of the [August 19] Washington presentation without changing one word of it because it was still applicable, and it addresses the highest concern of the field joint in both the ignition transient condition and the steady state condition, and it really sets down the rationale for why we were continuing to fly. Basically, if erosion penetrates the primary O-ring seal, there is a higher probability of no secondary seal capability in the steady state condition. And I had two sub-bullets under that which stated bench testing showed O-ring not capable of maintaining contact with metal parts gap opening rate to maximum operating pressure. I had another bullet which stated bench testing showed capability to maintain O-ring contact during initial phase (O to 170 milliseconds of transient). That was my comfort basis of continuing to fly under normal circumstances, normal being within the data base we had.

I emphasized, when I presented that chart about the changing of the timing function of the O-ring as it attempted to seal. I was concerned that we may go from that first beginning region into that intermediate region, from 0 to 170 being the first region, and 170 to 330 being the intermediate region where we didn't have a high probability of sealing or seating.[66]

---

[65] Ibid.

[66] Ibid., Volume IV, p. 790.

Mr. Boisjoly then presented Chart 2-2 with his added concerns related to the timing function. He mentioned in his testimony to the Rogers Commission:

> We would have low O-ring squeeze due to low temperature which I calculated earlier in the day. We should have higher O-ring Shore hardness. . . . Now, that would be harder. And what that material really is, it would be likened to trying to shove a brick into a crack versus a sponge. That is a good analogy for purposes of this discussion. I also mentioned that thicker grease, as a result of lower temperatures, would have higher viscosity. It wouldn't be as slick and slippery as it would be at room temperature. And so it would be a little bit more difficult to move across it.
>
> We would have higher O-ring pressure actuation time, in my opinion, and that is what I presented. . . . These are the sum and substance of what I just presented. If action time increases, then the threshold of secondary seal pressurization capability is approached. That was my fear. If the threshold is reached, then secondary seal may not be capable of being pressurized, and that was the bottom line of everything that had been presented up to that point.[67]

Asked by Chairman Rogers, "Did anybody take issue with you?." Mr. Boisjoly responded:

> Well, I am coming to that. I also showed a chart of the joint with an exaggerated cross section to show the seal lifted off, which has been shown to everybody. I was asked, yes, at that point in time I was asked to quantify my concerns, and I said I couldn't. I couldn't quantify it. I had no data to quantify it, but I did say I knew that it was away from goodness in the current data base. Someone on the net commented that we had soot blow-by on SRM-22 [Flight 61-A, October, 1985] which was launched at 75 degrees, I don't remember who made the comment, but that is where the first comment came in about the disparity between my conclusion and the observed data because SRM-22 had blow-by at essentially a room temperature launch.
>
> I then said that SRM-15 [Flight 51-C, January, 1985] had much more blow-by indication and that is was indeed telling use that lower temperature was a factor. This was supported by inspection of flown hardware by myself. I was asked again for data to support my claim, and I said I have none other than what is being presented, and I had been trying to get resilience data, Arnie [Thompson] and I both, since last October, and that statement was mentioned on the net.[68]

This second phasse of the telecon on the evening of January 27 concluded with statements from Robert Lund, Thiokol's Vice President of Engineering. His conclusion at that time was that the Shut-

---

[67] Ibid., p. 791.
[68] Ibid.

tle should not fly outside Thiokol's database; that, is, that the O-ring seals should be above 53 degrees Fahrenheit before lift-off.

NASA participants in the telecon were not pleased with these conclusions and recommendations, according to Mr. Boisjoly and Mr. McDonald. Mr. Hardy, when asked what he thought about Thiokol's recommendation, was quoted to the effect that he was "appalled" at Mr. Lund's decision.[69]

Boisjoly also testified that Mr. Hardy said, "No, not if the contractor recommended not launching, he would not go against the contractor and launch."[70]

Shortly thereafter, Mr. Joseph Kilminster, Thiokol's Vice President for Space Booster Programs, was asked by NASA if he would launch and "he said no because the engineering recommendation was not to launch."[71]

Then, according, to Mr. Boisjoly, someone in Thiokol management asked for a five-minute caucus, and at that point Thiokol cut their speakerphone off.

> Chairman ROGERS. "Mr. Boisjoly, at the time that you made the—that Thiokol made the recommendation not to launch, was that the unanimous recommendation as far as you knew?"
>
> Mr. BOISJOLY. "Yes. I have to make something clear. I have been distressed by the things that have been appearing in the paper and things that have been said in general, and there was never one positive, pro-launch statement ever made by anybody. There have been some feelings since then that folks have expressed that they would support the decision, but there was not one positive statement for launch ever made in that room."[72]

Asked for his recollection of these incidents, Mr. McDonald commented,

> . . . And the bottom line was that the engineering people would not recommend a launch below 53 degrees F. The basis for that recommendation was primarily our concern with the launch that had occurred about a year earlier, in January of 1985, I believe it was 51-C.[73]

Mr. Mulloy testified:

> The bottom line of that, though, initially was that Thiokol engineering, Bob Lund, who is the Vice President and Director of Engineering, who is here today, recommended that 51-L not be launched if the O-ring temperature predicted at launch time would be lower than any previous launch, and that was 53 degrees. . . .[74]

At 10:30 p.m. EST, the teleconference between NASA and Thiokol was recessed. The off-net caucus of Thiokol personnel lasted ap-

---

[69] Ibid.
[70] Ibid.
[71] Ibid.
[72] Ibid.
[73] Ibid., Volume IV, p. 717.
[74] Ibid., Volume IV, p. 604.

proximately thirty minutes at the Wasatch office. Jerald Mason, Senior Vice President for Wasatch Operations, remembered that the conversation during the caucus centered around O-rings and the history of eroosion of the O-rings. Mr. Mason testified:

> Now, in the caucus we revisited all of our previous discussions, and the important things that came out of that was, as we had recognized, we did have the possibility that the primary O-ring might be slower to move into the seating position and that was our concern, and that is what we had focused on originally. . . . The fact that we couldn't show direct correlation with the O-ring temperature was discussed, but we still felt that there was some concern about it being colder.[75]

Ten engineers participated in the caucus, along with Mr. Mason, Mr. Kilminster, Mr. Lund and Mr. C.G. Wiggins (Vice President and General Manager for Thiokol's Space Division). Arnold Thompson [76] and Mr. Boisjoly voiced very strong objections to launch, and the suggestion in thieir testimony was that Lund was also reluctant to launch.

Mr. Boisjoly, in testifying before the Rogers Commission, stated:

> Okay, the caucus was started by Mr. Mason stating that a management decision was necessary. Those of us who opposed the launch continued to speak out, and I am specifically speaking of Mr. Thompson and myself because in my recollection he and I were the only ones that vigorously continued to oppose the launch. And we were attempting to go back and rereview and try to make clear what we were trying to get across, and we couldn't understand why it was going to be reversed. So we spoke out and tried to explain once again the effects of low temperature. Arnie actually got up from his position which was down the table, and walked up the table and put a quarter pad down in front of the table, in front of management folks, and tried to sketch out once again what his concern was with the joint, and when he realized he wasn't getting through, he just stopped.
>
> I tried one more time with the photos. I grabbed the photos, and I went up and discussed the photos once again and tried to make the point that it was my opinion from actual observation that temperature was indeed a discriminator and we should not ignore the physical evidence that we had observed.
>
> And again, I brought up the point that SRM-15 [Flight 51-C, January, 1985] had a 110 degree arc of black grease while SRM-22 [Flight 61-A, October, 1985] had a relatively different amount, which was less and wasn't quite as black. I also stopped when it was apparent that I couldn't get anybody to listen.[77]

---

[75] Ibid., p. 759.
[76] Supervisor of Structures Design, Thiokol.
[77] Rogers Commission Report, Volume IV, pp. 792-93.

Commissioner Walker asked, "At this point did anyone else speak up in favor of the launch?" Mr. Boisjoly replied:

> No, sir. No one said anything, in my recollection, nobody said a word. It was then being discussed amongst the management folks. After Arnie and I had our last say, Mr. Mason said we have to make a management decision. He turned to Bub Lund and asked him to take off his engineering hat and put on his management hat. From this point on, management formulated the points to base their decision on. There was never one comment in favor, as I have said, of launching by any engineer or other nonmanagement person in the room before or after the caucus. I was not even asked to participate in giving any input to the final decision charts.
>
> I went back on the net with the final charts or final chart, which was the rationale for launching, and that was presented by Mr. Kilminster. It was hand written on a note pad, and he read from the notepad. I did not agree with some of the statements that were being made to support the decision. I was never asked or polled, and it was clearly a management decision from that point. . . .
>
> I left the room feeling badly defeated, but I felt I really did all I could to stop the launch.[78]

In testimony before the Committee on June 18, 1986, concerning the caucus and the decision to overrule engineering recommendations, Boisjoly said:

> When we went off the line and caucused—one of the first statements that was made was that we would have to make a management decision by management people. And we continued very strongly to oppose that and we argued as vigorously as we could argue, and when you look up into people's eyes you know you have gone about as far as you can go.
>
> And so both Mr. Thompson and I just plain frankly backed off. You had to be there and you had to see the looks and feel the experience that it didn't really make any difference what further you were going to say, you were just not going to be heard.[79]

At approximately 11 p.m. EST, the Thiokol/NASA teleconference resumed, with Mr. Kilminster stating that they had reassessed the problem, that the temperature effects were a concern, but that the data were admittedly inconclusive. He read the rationale recommending launch and stated that to be Thiokol's recommendation. Mr. Hardy of NASA requested that it be sent in writing by telefax both to Kennedy and to Marshall, and it was.[80]

---

[78] Ibid.

[79] Cmte Hgs, Transcript, June 18, 1986, p. 85.

[80] The Committee has learned that (apparently due to an error in duplicating the relevant chart) the copy of this telefax that was sent to the Rogers Commission did not contain the standard caveat that was printed below the company logo on the original telefax. The caveat reads, "Information on this page was prepared to support an oral presentation and cannot be considered complete without the oral discussion." At the Committee hearings on June 18, 1986, Mr. U.

Continued

Having heard the debate and this decision, Mr. Reinartz accepted the conclusion and ended the teleconference. He asked whether anyone had any further concerns. None were expressed. Commissioners then asked what steps he had taken after the decision was made.

Chairman ROGERS. "I guess the question that still lingers in my mind is, in the Navy we used to have an expression about going by the book, and I gather you were going by the book. But doesn't the process require some judgment? Don't you have to use common sense? Wouldn't common sense require that you tell the decisionmakers about this serious problem that was different from anything in the past?

Mr. REINARTZ. "In looking at that one, Mr. Chairman, together with Mr. Mulloy when we looked at were there any launch commits, any Level II, as I perceived during the telecon, I got no disagreement concerning the Thiokol launch between any of the Level III elements, the contractor, with Mr. McDonald there. I felt that the Thiokol and Marshall people had fully examined that concern, and that it had been satisfactorily dispositioned based upon the evidence and the data that was supplied to that decision process on that evening, from that material, and not extraneous to what else may have been going on within Thiokol that I had no knowledge of."

Chairman ROGERS. "Okay. Thank you. I'm sorry for the long interruption."

Mr. REINARTZ. "Based upon—and as we skipped over it is only a point to illustrate, Mr. Chairman, that in our discussion about the parachute with KSC and Mr. Aldrich, was to indicate that there was a clear area there where we had a very direct responsibility to inform them of the situation, which Mr. Mulloy did. And after a discussion of that issue, Mr. Aldrich concluded that the launch should proceed in that nature. Based on the results of the meeting and the conclusions out of the meeting, Mr. Mulloy and I informed the Director of Marshall, Dr. Lucas, and the Director of Science and Engineering, Mr. Kingsbury, on the 28th of January—about 5:00—of the initial Thiokol concerns and engineering recommendations, the final Thiokol launch recommendation, that I felt had led to a successful resolution of this concern."

General KUTYNA. "Could I interrupt for a minute? You informed Dr. Lucas. He is not in the reporting chain?"

Mr. REINARTZ. "No, sir."

General KUTYNA. "If I could use an analogy, if you want to report a fire you don't go to the mayor. In his position as center director, Dr. Lucas was cut out of the reporting chain, much like a mayor. If it was important enough to

---

Edwin Garrison, President of the Aerospace Group at Thiokol, testified that the caveat at the bottom of the paper in no way". . . insinuates. . . that the document doesn't mean what it says." (Cmte Hgs, Transcript, June 18, 1986, p. 43.) After further investigation of this deletion, the Committee has concluded that it was not significant.

report to him, why didn't you go through the fire department and go up your decision chain?"

Mr. REINARTZ. "That, General Kutyna, is a normal course of our operating mode within the center, that I keep Dr. Lucas informed of my activities, be they this type of thing or other."

General KUTYNA. "But you did that at 5 o'clock in the morning. That's kind of early. It would seem that's important. Why didn't you go up the chain?"

Mr. REINARTZ. "No, sir. That is the time when we go in, basically go into the launch, and so it was not waking him up to tell him that information. It was when we go into the launch in the morning. And based upon my assessment of the situation as dispositioned that evening, for better or worse, I did not perceive and clear requirement for interaction with Level II, as the concern was worked any dispositioned with full agreement among all reasonable parties as to that agreement."

Chairman ROGERS. "Did I understand what you just said, that you told Dr. Lucas that all the engineers at Thiokol were in accord?

Mr. REINARTZ. "No, sir. What I told him was of the initial Thiokol concerns that we had and the initial recommendation and the final Thiokol recommendation and the rationale associated with that recommendation, and the fact that we had the full support of the senior Marshall engineering [81] and, as George has testified, to the extensiveness of the group of people we had involved in that telecon with the various disciplines, that those three elements made up the final recommendation."

Mr. HOTZ. "Mr. Reinartz, are you telling us that you in fact are the person who made the decision not to escalate this to a Level II item?"

Mr. REINARTZ. "That is correct, sir." [82]

According to NASA's Program Directive SFO-PD 710.5A, Mr. Reinartz may have been required to report the matter to the Associate Administrator for Space Flight. A portion of that directive reads as follows:

> Significant items occurring subsequent to the FRR will also be reported to the AA-SF. Actions that can be easily accomplished without safety, mission, or launch impact and do not violate flight vehicle or launch complex configuration integrity or cause basic changes to launch commit criteria, flight rules, flight plan, or abort and alternate mission plans, need not be reported.[83]

Was this telecon, and the decision reached, "significant?" NASA's request that the Thiokol decision be put in writing indi-

---

[81] Staff review of teleconference materials used by Marshall engineers Wilbur Riehl (Chief, Nonmetallic Materials Division) and John Miller (Technical Assistant to the SRM Manager) indicates that some of the Marshall engineering staff shared the concerns expressed by Thiokol engineers.

[82] Rogers Commission Report, Volume V, pp. 917-18.

[83] SFO-PD 710.5A, p. 3.

cates that MSFC personnel felt the situation was significant, since in effect Thiokol was reconfirming the flight readiness of the SRM.

It is also interesting to note, in light of the directive, that Mr. McDonald testified before the Commission to the effect that Mr. Mulloy had made some "fairly strong comments . . . about trying to institute new launch commit criteria." [84]

Mr. Mulloy responded to this by explaining what he had attempted to say.

> Mr. MULLOY. "The total context, I think, in which those words may have been used is, there are currently no Launch Commit Criteria for joint temperature. What you are proposing to do is to generate a new Launch Commit Criteria on the eve of launch, after we have successfully flown with the existing Launch Commit Criteria 24 previous times. With this LCC, i.e., do not launch with a temperature greater than [*sic*] 53 degrees, we may not be able to launch until next April. We need to consider this carefully before we jump to any conclusions. It is all in the context, again, with challenging your interpretation of the data, what does it mean and is it logical, is it truly logical that we really have a system that has to be 53 degrees to fly? . . ."
>
> General KUTYNA. "Mr. Mulloy, if in fact the criteria were 53 degrees, it would have an impact not only on this launch, but on the shuttle program. . . . It is a fairly important decision to say you can't launch below 53 degrees, isn't it?
>
> Mr. MULLOY. "Yes, sir, I agree with that. I cannot describe the impacts, but, as I say, based upon our previous experience and our actions in flying subsequent vehicles after 51C, I found that to be a surprising conclusion. . . ."
>
> Mr. SUTTER. ". . . [I]nstead of saying you have to wait until next April to launch, the thing that you do is you go and there were three different levels of improvements that were discussed. The thing to do then was to put those improvements in the program, not infer that these engineers are saying, we're throwing a ringer at you that says don't launch until next April. I think that is putting those engineers into a little bit of a hot seat. And if they're trying to do their job and say, hey, we ought to do something about this, there ought to have been more attention paid." [85]

The Rogers Commission report included the statement, "It is clear that crucial information about the O-ring damage in prior flights and about the Thiokol engineers' arguments with the NASA telecon participants never reached Jesse Moore or Arnold Aldrich, the Level I and II program officials, or J. A. Thomas, the Launch Director for 51-L." [86]

Based on the available evidence, the Committee is unable to conclude whether or not Mr. Moore or Mr. Aldrich, with the informa-

---

[84] Rogers Commission Report, Volume IV. p. 721.
[85] Ibid., Volume V. pp. 843-45.
[86] Ibid., Volume I, p. 101.

tion available to them that day, would have reached a decision to stop the launch had they been informed of this meeting. It does appear, however, that a three-hour telecon, in which arguments are raised about launch commit criteria and the contractor is asked to reconfirm the flight readiness of his hardware fall under the definition of the policy directive. At the very least, the STS Program Manager should have been presented the new declaration of flight readiness with an explanation of why it had been developed. This should have been a necessary addition to the Certificate of Flight Readiness prepared after the Flight Readiness Review.

The Committee also reviewed tapes and transcripts of conversations that took place in the Firing Room on January 28th involving discussions of the threat posed by ice on the Fixed Service Structure. Kennedy Space Center managers, in response to the predicted low temperatures that would be seen in the hours before the launch of STS-51L, took action to protect Launch Complex 39B and the Shuttle from freezing and ice buildup. This involved implementing the "freeze protection plan" for launch pad facilities. According to a post-accident report:

> Two actions within the PLAN were intended to limit the ICE DEBRIS which potentially could cause damage to the Shuttle Vehicle during launch. The first action involved adding approximately fourteen hundred gallons of antifreeze into the overpressure water troughs. The water troughs in both SRB exhaust holes have a total capacity of 6,580 gallons. The resultant antifreeze to water ratio was calculated to be 21.3%. According to the manufacturer's specifications, solution protected against freezing down to an ambient temperature of 16 degrees F. The second action involved the draining, where practical, of all water systems. Several systems, such as Firex [fire extinguishing], Deluge, and emergency shower and eyewash, were not drained. These systems were opened slightly and allowed to trickle into drains. The trickling water was found to cause drain overflows. High wind gusts then spread the water over large areas and it then froze.[87]

Soon after the call-to-stations on 28 January, at approximately midnight, cameras on the pad allowed engineers in the Firing Room to see that the gantry was heavily encrusted with ice. Over the Engineering Support Room communications loops, the following conversation took place: [88]

---

[87] NASA, Kennedy Space Center, "STS-33 (51-L) Ice/Frost Team Evaluation Report: ESS/RSS/MLP Deck/Pad Apron Icing," January 30, 1986, p. 1.

[88] Conversations were recorded from the Kennedy Space Center Operational Intercommunication System (OIS), which permits members of the launch crew to discuss problems that occur during the countdown, and permits them to contact various mission support facilities around the country. The transcripts provided to the Committee do not indicate the exact times at which the referenced conversations occurred, and so the flow of conversations has been reconstructed in an attempt to provide logical consistency.

Most of the transcripts in this section are drawn from OIS Channel 245, identified as the coordination channel for JSC/MSFC/KSC Engineering personnel in the Engineering Support Area at Kennedy Space Center. Conversations among Rockwell International personnel were obtained from OIS Channel 216, described as the coordination channel for JSC personnel at Johnson and Kennedy centers and Rockwell engineers in Downey, California. Transcript page numbers are those supplied by NASA.

Continued

BL. This is Bob on 245.
DIRECTOR. Go ahead, Bob.
BL. Did you ever find out any more about that water?
DIRECTOR. No, no I haven't. [89]
BL. Okay, I guess the thing, they don't know yet where that water's coming from and the NTD's [NASA Test Director] got some folks looking into it. I guess the thing you guys need to do is see if you can get certain cameras to look at the vehicle and to determine if there's any water getting on the vehicle that might freeze and cause problems.
DIRECTOR. I think what we need to do, Bob, we need to decide do we feel comfortable enough to let it keep running and forming ice up there, or we ought to stop and send somebody in there and try to shut it off.
BL. Yeah, okay.
DIRECTOR. Do you guys think we could form enough ice there to cause us any problem on liftoff or anything?
BL. I think you've already done that, Horace.
DIRECTOR. Well, then, we ought to stop and go out there and get the water shut off.
BL. Yeah, we're worried about an icicle up high—well, see, camera 108's on the 155 foot. . . .
ITL. 155? Yeah. So you're already getting high up, you know. And if the wind's going to be out of the north-northwest. . . .
DIRECTOR. All right, let's stop them and send the people out there and see if they can shut the water off.
ITL. I think what it is, is the fire hose—if you look right over, if you go to 108 and go like you're going to the elevator, you'll see a fire hose, looks like a fire hose draped across there...
DIRECTOR. Yeah.
ITL. . . . And I think they take the fire hose and carry it over to the shower, the eye shower. And evidently the drains that they're draining into is frozen off, or either the hose has fallen off the drain, one or the other.
DIRECTOR. Okay, we gotta work this in. We're going to tell them to go out there and shut the water off.
ITL. Okay.[90]

The ice/frost team was dispatched to the pad and arrived at approximately 1:45 a.m. What they found during their inspection of the Fixed Service Structure was not very encouraging to the team leader. He reported back to the Firing Room:

ITL. Horace, this is Charlie on 245.
BL. Hello, this is Bob. I think he may still be in that HIM (Hardware Interface Module] meeting [on the fire detector problem]. What do you see out there?
ITL. Okay, starting on about the 235 foot level where the top hose is, the fire hose that was draining into the shower, the hose is not really, the drain is in the shower, the hose is not really draining into the little bowl on the shower and it was spilling over. So we have a lot of hard solid ice from the 235 feet down to 195 feet where I am now. Most of it's on the west side and the north side, and about halfway in-between, the floor is one solid sheet of ice about an inch and a half thick. And down on 195 foot level, the water's on the pipe and plumbing and structure and beams all the way over to the Orbiter Access Arm [OAA]. That's as far down as we got so far.
BL. Copy.
ITL. We have some icicles about 18 inches long.[91]

The Ice Team leader later reported that ice was covering part of the floor on the 195 foot platform where the crew would enter the Orbiter. Part of this discussion follows:

Significant participants in these conversations include:
DIRECTOR: Director of Engineering, Kennedy Space Center.
BL: Chief, Mechanical Systems Division, Shuttle Engineering Directorate.
LD: Launch Director, STS 51-L.
ITL: Ice Team Leader.
[89] OIS Channel 245, p. 570.
[90] Ibid., pp. 570-72.
[91] Ibid., p. 573.

BL. Charlie, is that ice going to be any kind of impediment to the crew?

ITL. Uh, not to the crew. I don't see any out here where the crew walks.

BL. How 'bout to the baskets [the emergency escape system]?

ITL. . . . Up to the what?

BL. If they had to go to the baskets for any reason, do they have a clear path through that ice?

ITL. Oh, no. When you get over that way, we got ice.

BL. So they had to get out in a hurry in order to get to the baskets, they've got to go over a sheet of ice.

ITL. Oh, yeah. On the north side, that's all one hard sheet of ice. Now, they could get to the two baskets on the south side, probably three baskets on the south side. Hold on just a minute, I'll go take a look. . . .[92]

ITL.Okay, Bob, I'm back.

BL. Okay. What's it look like over there?

ITL. Okay—some right at the elevator, right where the camera is, going back toward the baskets, we got ice on the floor. And the ice goes all the way across the west side of the facility, all the way over to the north corner on the floor. So it is slippery. Once you get past the west-most part of the FSS, the ice on the floor ceases, and you got a clear walkway, so all five baskets are, uh, six baskets do have a clear walkway right around the baskets. But to get between the elevator and the camera where you're looking at, there's some ice on the floor.

BL. Okay.

ITL. And including the handrails that they would be holding on to. But out here from the Orbiter Access Arm over to the camera is clear.

BL. What's your Safety guy there think about that? You got a Safety guy with you, don't you?

ITL. Yes, he's concerned. Matter of fact, there's some ice right under my feet now that I look.

BL. Charlie, Horace is back with us now. Why don't you start up your review from the 235 foot level on down again.

ITL. Okay. From the 235 foot level is where we had these litle Firex systems—the hose, the rubber hose which we ran over to the shower back on the northwest corner of the FSS, so we ran it on to the eyewash shower the level below, the 235 foot level, and it was running out of the drain, you know, the little basin—it was overflowing the little basin. So we have over in that area, down to the 195 foot level on the north side of the FSS, icicles that are about 18 inches or so long, about one inch in diameter or more at their maximum diameter. This floor, the grating, over on the north side paralleling the showerway and the elevator in some places are frozen solid about two inches thick. You know, the area is like 10 [by] 10 or more.

DIRECTOR. On the floor, which floors, Charlie?

ITL: The floor of the grating, on like the 215 and there's a level between 215 and right under, between 215 and 195; there's a half-level that you go out to the hatch on the north side.

DIRECTOR. Okay.

ITL. And there's a lot of icicles hanging, you know, under the floor. As far as the east side goes, on the 195 foot level where we are now, we have about one-quarter inch or one-half inch ice. On the beam structures themselves, they go all the way over to the—right at the hinge where the Orbiter Access Arm goes out. I don't see any on the Orbiter Access Arm itself, but there is some here where I'm standing and a little bit on the floor. Just before you came on, we were talking about the slidewire—the baskets. The floor from the Orbiter Access Arm over to, back where the camera is, is fairly clean. And from the camera back to the west side, is some ice on the floor to the west edge of the FSS.

DIRECTOR. Okay.

ITL. So the crew would have to walk across one slick spot. Around the baskets themselves, it's fairly clean. But to the northwest corner of the FSS where the baskets are, there's heavy concentration of ice on the floor.[93]

The discussion concluded:

DIRECTOR. Do you see anything out there that makes it unsafe for the crew?

ITL. At this time, I'd say from the elevator to the Orbiter Access Arm would be fairly good; the floor's in good shape. The elevator's got a little bit of—the doors are real hard to work but everything seems to work in that neighborhood. If they had to

[92] Ibid., p. 574.

[93] Ibid, pp. 574-76.

go to the slidewire, it'd be very slippery from the camera that you're looking at to the slidewire itself. There's an area about ten feet long where the handrails have ice on them, as well as the floor.[94]

This discussion is of concern to the Committee. The slidewire referred to is the means by which the Shuttle crew would evacuate the launch pad if an emergency were to occur that required a rapid exit from the pad area. Crew safety concerns should dictate that the ice situation described by the Ice Team leader is unacceptable, and that some effort to remove the ice from the floor and handrails should be made. There is no indication that this was done. If the ice could not be removed, the mission should have been delayed until the danger represented by the ice could be eliminated.

This situation, admittedly, had nothing to do with the accident that destroyed STS-51-L. However, had this been the one time that use of the pad escape system was required, the crew would very likely have been impeded in their attempt to reach the escape baskets, and the lost time might have proven fatal. This system must be operated with the expectation that it will be used, and the countdown procedure should require that no barriers to its use be present before launch.

As the Ice Team continued its inspection, the discussion in the Firing Room involved the possible threat posed by the ice problem.

BL. We're just going to ask your opinion on the debris concern. If Charlie thinks we have a concern with debris, and I guess I would find it hard to believe that we'd be concerned about it from the FSS, but if we do have a concern, can we go out there and try to clean it up a little bit?

DIRECTOR. Yeah, I think we could. I think when he gets through here, if we think there's some areas that we need to clean up a little bit, we probably could.

BL. I think Safety would probably have to make a call, myself, on the floor if they think it's, that's a concern, but. . . .

DIRECTOR. Okay, he's [ITL] on 108 now.

BL. I see him on camera 108, next to the 155 foot level.

LD. Hey, what kind of debris are you guys talking about?

BL. The icicles on the FSS.

LD. Yeah, and how is that going to hurt you?

BL. Well, that's what I'm saying. I don't think it—personally, I don't think it would, but I just wanted to. . . .

DIRECTOR. [garbled] by there, you're not gonna hit the Orbiter, but Charlie's worried about it, Gene—the acoustics releasing it and it being free when the Orbiter comes by.

LD. Boy, he's really stretching it.

DIRECTOR. Oh no, I don't know whether that's stretching it too much or not.

LD. Well, I mean if we can ignore it, we need to feel comfortable about it.

P. All right, Gene, remember the wind is coming from the northwest.

LD. We need to all know if we don't get back into tanking as soon as possible, we could possibly blow it just for that.

DIRECTOR. Yeah, we understand, Gene. . . .[95]

The Ice Team leader's next report was no more encouraging. Water had spilled over the platform as the drains were unable to cope with the volume of water they were asked to manage. Icicles were found on the platform handrails that could easily be knocked off.[96]

---

[94] Ibid., p. 576.
[95] Ibid., pp. 577-78.
[96] Ibid., p. 578.

As the team leader was explaining that the water could not be completely cut off for fear of making the situation worse.[97] the following conversations occurred:

LD: Hey, we gotta come out of there when you guys telling us you're pretty sure that water system's gonna work.

DIRECTOR. Yeah, you feel comfortable with what you see out there, Charlie, now?

ITL. We have a lot of ice, if that's what you mean. I don't feel comfortable with what's on the FSS.

DIRECTOR. Then what choices we got?

ITL. Well, I'd say that only choice you got today is not to go. We're just taking a chance of hitting the vehicle.

LD. You see that much ice?

ITL. Well, the problem we have is we hve a lot of icicles hanging, you know, even on the west side of the FSS here, which is only 60 feet or more from the Orbiter wing. And I'm sure that stuff is going to fall off as soon as the acoustics get to it. And you got a northwest wind, so you know, somebody will have to make that assessment. If we're worried about that little bit of ice that comes off the hydrogen vent arm, and the GOX [gaseous oxygen] vent arm, what we have over here is considerably more than that, you know—it's a hundred-fold.

DIRECTOR. You got enough ice that's over there that's big enough and got enough density to it that if it hits the Orbiter it could do some significant damage?

ITL. Yes, we do. . . . It's on the east side of the FSS. On the northeast corner of the FSS, which puts you about 65 feet or so from the vehicle. But it comes right to about where this camera is, it's right on the center thin line of the FSS, it comes that far over.

BL. Charlie, I would doubt the wind could blow that over. Are you concerned about during—after engine start, that things should kinda blow around?

ITL. Uh, yes. And the problem is it's so high, too. You know, it's way up to the top. If it were all the way down here to the bottom, it probably wouldn't be any problem.

BL. Can we go along the east side handrails and knock it off now? Isn't that the biggest concern—the east side?

ITL. Well, it's on the handrails and its on the floor underneath, too. You know, I guess it could probably be done but it'd be a job.

BL. It would take a long time, wouldn't it?

ITL. All the FACS pipes, and all the conduit and all the cable trays and then hanging down underneath the floor, you know, everywhere, on all the pieces of grating, you got little icicles hanging down.

DIRECTOR. Who do you have out there with you?

ITL. B.K., as far as my group goes.

DIRECTOR. Okay, why don't you guys go ahead and walk everything down and quick as you get—come on back, let's get with Gene and we'll sit down and talk about what we got.[98]

LD. Okay, why don't you do like Horace said, come on back and we'll go ahead and tank and we'll have you look at it when you go back.

ITL. Okay, and I think—you know, the Rockwell people have a program which says it probably would be all right, so contact them and let them put it in the machine and see what they get.[99]

As the Ice Team returned from its initial inspection, the launch director spoke to KSC's Director of Engineering about the ice situation:

LD. Yeah, we really need to do some head scratching on this ice thing and what we gonna do once we get back in. We've just about used all our hold.

DIRECTOR: Okay.

LD. What we're gonna do—we're not gonna opt to have a hold, we'll let the ice team go in just like normal, but we gonna keep counting the clock, and if we have to get ourselves into eating some of our launch window, we'll do it late.

DIRECTOR. Okay. We can do the ice inspection parallel with counting.

LD. Okay, but we need to have Rockwell in there where we need to ready to talk.

DIRECTOR. We can get Don in and we'll do that.

---

[97] Ibid.
[98] Ibid. pp. 579–81.
[99] Ibid., pp. 581–82.

LD. I don't see how in the world—we need to worry about that ice. I don't know what's going to knock it off. It's going to freeze and stay there.

DIRECTOR: I don't think it will hit the Orbiter, Gene. It'll probably come off just from the acoustics, but. . . .

LD. Yeah, as you go by, I would think . . . I would think after we start the engines, if it starts breaking loose then, I don't know how it would travel that far until, you know, the six seconds or so we are on the ground.

DIRECTOR. Yeah. We'll get everybody here; we'll talking before—while we're tanking before they go out, then we'll be ready to help make an assessment.[100]

The Ice/Frost Team returned to the launch pad during the scheduled countdown hold at T-3 hours. The crew was somewhat larger than usual, since their primary objective was to clear ice from the water troughs on the launch pad. The team would also be making a follow-up assessment of ice on the FSS.

According to the post-accident report, "the team arrived at Pad B at 0654 hours and departed at 0844." A summary of their activity during this time stated:

> Ice in the troughs had thickened and was found to be solid. All secondary troughs except the northern most one in each hole now had ice. The two inboard primary hole troughs were also forming ice. . . . The "shrimp net" was employed to break up the ice and remove it. Approximately 95% of the ice was removed. The ice and unfrozen antifreeze solution was measured using an infra-red pyrometer and found to have a temperature between 8 and 10°F. . . . The pyrometer measured the MLP deck surface temperature as 12°F. On the FSS the quantity of ice had increased but the overall extent of icing was generally the same. In most cases, sheet ice was firmly adhered to the structure. Icicles could very easily be "snapped" off. Water continued to trickle down the facility—including the RSS [Rotating Service Structure].[101]

NASA launch team members were continuing their debate over the risk represented by the ice at this time.

LD. What do you think about the ice now?

DIRECTOR. Well, I don't know, Gene. I keep thinking there is an answer if we can find it, but we got people out talking it and I think we can make a decision on the SRB things [troughs]. I think we have the data there that we can make the right decision on that. The tower is what's going to be the one that is going to be hard to come to a decision on.

LD. But do we have any data that shows a mechanism for moving that ice across there?

DIRECTOR. That's what we're trying to see if we can come up with some kind of rationale why it won't. Charlie says we've got some data that we have moved some pieces across from basically the tower to the vehicle but we're—[Marty Ciofoletti, Vice President for Space Transportation System Integration, Rockwell International] and the guys are working with Downey to see how they feel if they come up with anything on the acoustics and things like that.

LD. Okay.[102]

KSC's Engineering Director then called the Rockwell liaison.[102a]

---

[100] Ibid., pp. 583–84.

[101] "Ice/Frost Team Evaluation Report," p. 4.

[102] OIS Channel 245, p. 595.

[102a] Rockwell personnel appearing in House transcripts are identified as follows:
RI: Kennedy Space Center liaison.
RTI: Director of Technical Integration.

Continued

DIRECTOR. John, on 245.
RI. Go ahead, Horace.
DIRECTOR. You guys still talking to Downey?
RI. Yeah, we haven't go anything new since I last talked to you. Larry Williams just called in. He wants to talk to Bill Hovath [?] so they are getting ready to make a call to his now.
DIRECTOR. Okay.
RI. But we haven't got anybody. You know Jack McTimmons [McClymonds]—I talked to him and all he had to offer was his old program data of how far ice would go at various wind speeds, you know, from the FSS that's not our primary concern, obviously, right now.
DIRECTOR. No, that's right.[103]

In the Mission Support Room in Downey, California, Rockwell support personnel were expressing the same concerns as the Ice Team Leader, and were not confident their computer model could remove the uncertainty presented by the ice. Rockwell's Kennedy Space Center liaison was asked for information.

RTI. KSC, MSR [Mission Support Room]
RI. Go.
RTI. Good morning, John. Uh . . .
RI. It's been a busy morning!
RTI. I bet—looks bad, eh?
RI. Ice does look bad, yeah. The situation we've got right now is that they're working the bags in the SRB hole; they reported slush in those bags and we were watching on TV and some of that slush was pretty big and pretty heavy. But I think we can take care of that part—I think they're gonna get that cleared up. There's a crew out there working on those right now. One of the concerns [Richard] Colonna [Orbiter Project Manager, JSC] had was reflected pressure wave problems if there was a film of ice across those bags, but it looks like they're breaking that up. The big concern is gonna be the mass of ice that is on the FSS, from the 235 foot level all the way down to the MLP [mobile launcher platform]. Every platform had had water running on it all night and they're just a—some of the closeups of the stairwells looks like, uh, something out of Dr. Zhivago. There's sheets of icicles hanging everywhere. We've had reports, back on the northwest corner, of ice, icicles—this is a couple hours ago, the crew are up there walking it down right now, so we'll probably get some updates here shortly—but the initial walkdown said icicles up to two feet long by an inch in diameter. On the northwest corner, kind of graduating down to about three inches by one-quarter inch diameter on the east side, with periodic one-foot icicles on the east side on some of the cross beams.
RTI. Sounds grim.
RI. The big concern is that nobody knows what the hell is going to happen when that thing lights off and all that ice gets shook loose and come tumbling down and—what does it do then? Does it ricochet, does it get into some turbulent condition that throws it against the vehicle? Our general input to date has been basically that there's vehicle jeopardy that we've not prepared to sign up to. . . .
RTI. Okay. We didn't see this when we had icing conditions before?
RI. No, and they didn't run the showers all damn night before. They ran the showers this time and ran'em, pretty heavily by the look of it, the drains froze up and they all overflowed.
RTI. Oh. . . .
RI. And I guess nobody watched it all night or, if they did, they didn't say anything. But, uh—is John [Peller, Rockwell Vice President for Engineering] in yet?
RTI. No.
RI. Okay. We need to—you know, somebody at his level needs to get in and try to get up to speed as fast as they can. They're going to be looking for a final position from Rockwell here very shortly. We got—Bill Frohoff is right now talking to Larry Williams of JSC. I've got Colonna and Bobola sitting here with Al Martin and myself and we're probably going to be the forcing factor on this decision. Until

RSD: Site Director for Launch Support Operations.
RTP: Thermal Protection Project Manager.
RDE: Vice-President for Engineering.
RVP: Vice President and Program Manager for Orbiter Operations Support.
RSR: Senior Representative, Mission Evaluation Room, Johnson Space Center.
[103] Ibid., pp. 596-97.

somebody can come up and tell us that the potential flow path is to the objects on the FSS at liftoff—you know, we're going to have to assume the worst case—but I don't think anybody is going to have that sort of data.

RTI. This is going to be a tough one.[104]

The Ice Team Leader reported that efforts to clear out the water troughs under the SRBs were meeting with success, and that the team was managing to clear the ice that had formed on the left SRB aft skirt.[105]

As this occurred, Rockwell's Site Director for Launch Support Operations was reporting from KSC to Downey. He said,

RSD. . . . [T]he situation here is that—very quickly—when Charlie gives his report, then they are gonna want to reconvene a top level management meeting here, so whatever we want to say in that meeting we're gonna have to come up with it here and now in order to be ready to say it and I guess the situation is that there are icicles all over the stand, that's the fixed service structure, all up and down it, various levels—some of the icicles are two feet long, an inch or two at the base, there are lots of small icicles hanging all over the place. What they say is that when they touch them gently that they break off and for that reason I don't think there is any doubt about the fact in my mind that when we start the SSMEs a lot of these icicles are going to break off and they're going to—and when they do break off, then what's going to happen is that they're gonna come tumbling down, they can ricochet off of the service structure and they can—then some of them wind up on top of the MLP.[106]

The discussion was interrupted at this point by another report from the ice team, indicating that the lower levels of the FSS had ice coverage equal to the levels already discussed. The decision was made to bring the team leader back for a report to managers, in order to decide whether the threat was sufficient to stop the launch.[107]

The discussion at Rockwell then resumed:

RSD: Okay. He was just reporting on one of the levels. As he, as Charlie Stevenson, of NASA, moves up and down with the ice team, they're reporting on each level and on that particular level he was reporting a signficant amount of ice as the result of the overflow from the shower. You know, they left the water running in order to keep the pipes from freezing and then, I guess, some of the drains have frozen so then the water's overflowing and that's what's creating a lot of the big icicles. But at any rate, what I was getting around to, it just appears to me that when these icicles break off when they start the SSMEs some of them are very likely—in fact, I'll tell you, almost for sure—are gonna wind up on top of the MLP and then when we launch it seems to me it would be very difficult for anybody to predict where that debris would go and it appears to me that there would be a possibility of some of that debris impacting the Orbiter tiles and I don't know how our aerodynamists or analysts or anybody you know could really say that that wouldn't happen. They can predict what happens when you drop a piece of ice in the wind. They can also predict what happens due to aspiration when you start the Solid Rocket Motors and SSMEs. The real question is how do you predict what happens to ice chunks that are on top of the MLP at launch and where they go. So, at any rate, that's how we see it here at launch.

RTI. Well, one thing I guess we can see, from the view that we have, is ice on top of the MLP right now.

RSD. Yes, there is ice on top of the MLP right now.

RTI. That's unacceptable. Anything in the trough is unaccepatable and any ice that would impact the vehicle during ascent is unacceptable and we can't predict what's going to happen to all that massive ice on the towers, so I think we're in a critical situation. . . .

---

[104] OIS Channel 216, pp. 340-41.
[105] OIS Channel 245, pp. 600-601.
[106] OIS Channel 216, p. 345.
[107] OIS Channel 245, pp. 603-604.

RTP. Most of the ice on the tower is going to end up on an MLP, probably right before SRB ignition anyway—right, A1? You think?

RTI. Well, it's going to end up looking like snow, though, isn't it?

RTP. No, this is hard ice.

RTI. Once it hits that tower it's not going to be hard ice. What we're worried about is the aspiration effects of the motion of the ice into the vehicle.

RTP. You're still going to have large chunks of ice, ice cubes. Like an ice cube.

RTI. That's unacc . . . question is, how high is, is the highest elevation of ice, what was the. . . .

RTP. I think they're saying it's all the way to the top of the tower, like the 235 foot level has icicles forming and all the way down from there.

RTI. Okay.

RSD. Bob, would you say again what you were saying about the ice on the tower and the concern about that?

RTI. We really don't have a data base to know what's going to happen to the ice. We do have some information that we can get horizontal movement of the ice into the vehicle. Obviously, since it's very tenuous, it's going to be bouncing all over the place. It'll be bouncing off the J-boxes and everything else. So you're going to have some horizontal velocity of ice.

RTP. Hey, Bob, you're breaking up again.

RTI. Okay. Let me try once more. Our data base does not allow us to scientifically tell you what's going to happen to the ice. Therefore, we feel we're in a no-go situation right now.

RSD. Okay. That, Bob, is a consensus down here, too—that there's no way, that is for Rockwell, the consensus down here for Bill Frohoff and ourselves, that there's no way to predict what's going to happen, and I think that when we get into this next meeting, we need to state that as Rockwell's position and I think that's going to come up fairly soon, here. Now, I have told Dick Colonna that I suspect that's going to be the Rockwell position, I haven't told them officially. I've also told Horace that, but I haven't told him that officially, and I guess, or, uh, do you think we're ready now to, uh, for Rockwell to state that position and do you want to go back to the MER with that or how do you want to handle it?

RTI. Well, what I would like to do is get ahold of [Bob] Glaysher—we're not supposed to overrule him—and talk to him about it. Is he there?

RSD. We woke him up at 4:00 this morning. He called in about an hour ago. We understood he was on the way and he's not here yet.

RTI. Okay, I'd like to stonewall it until he gets here.[108]

At about this time, Mr. Bill Fleming, the senior representative of Rockwell International, reported from the Mission Evaluation Room at JSC that "ice on MLP, tower and trough not acceptable to MSR [Rockwell's Downey facility]."[109]

Just prior to attending the meeting called by Arnold Aldrich, the Vice President and Program Manager for Orbiter Operations Support at Rockwell held the following teleconference with their Chief Engineer at Downey.

RDE. Hey, we've gone over this again. Colonna called me and wanted to see if there is a way we could give it a go. But, when all the experts have looked at it, we still have concerns with three mechanisms. One, direct transport of falling ice into the vehicle at SSME ignition and the wind is adequate to make that happen. The ten-knot wind can move it laterally like twenty feet and a fifteen-knot gust could take it laterally forty feet. So even though you might be able to placard it, it's very close with the wind you've got. Secondly, you've got a rebound mechanism, where ice falls down into the lower part of the platform and goes out. Some pretty sizeable chunks and sometimes all it does is break an icicle in two, that's clearly enough to cause significant tile damage. And, finally, the ice ends up on the MLP and in the trough is all potential debris sources at SRB ignition and liftoff and the trajectory those things take are highly unpredictable and we just note in films tended to go in different directions. So we are not in the position to, uh. . . . So we've been through the three mechanisms, none of which we can completely clear. Dr. Petrone's here; we've discussed it with him. We still are of the position that it's still a bit of Rus-

---

[108] OIS Channel 216, pp. 345–48.

[109] NASA, Johnson Space Flight Center, "Presidential Commission Action Item (A-301) Response," DDATF-86-36, April 7, 1986, p. 1.

sian roulette; you'll probably make it. Five out of six times you do playing Russian roulette. But, there's a lot of debris. They could hit direct, they could be kicked up later by the SRBs, and we just don't know how to clear that.

RVP. Okay. Our position fundamentally hasn't changed. We'll just go in now, we got a 9:00, we'll go in and express it. I'll let you know what happens.

RDE. And obviously, uh, you know, it's their vehicle and they can take the risk, but our position is as stated.

RVP. Okay, you got it.[110]

No recording exists of the meeting at 9:00 a.m. on January 28. In testimony before the Commission, Mr. Robert Glaysher stated that he told Mr. Aldrich that "Rockwell cannot assure that it is safe to fly."[111]

Mr. Al Martin testified that:

> I also added that we do not have the data base from which to draw any conclusions for this particular situation with the icicles on the tower, and also, we had no real analytical techniques to predict where the icicles might go at lift-off. The other thing that I did was review the fact that prior to each launch there is great care taken to assure that there is no debris out on the launch pad. A day or two before launch a crew goes out and they walk down the entire tower and walk down the mobile launcher surface, and also the concrete apron around the launch pad for the purpose of removing any debris such as nuts, bolts, rocks or anything else that might be there. . . . So I was drawing a corollary between the care that is normally taken for debris and painting a picture that the icicles appeared to me to be in that same category.[112]

Mr. Marty Cioffoletti testified that "I felt that by telling them we did not have a sufficient data base and could not analyze the trajectory of the ice, I felt he understood that Rockwell was not giving a positive indication that we were for the launch."[113]

Mr. Aldrich, conversely, told the Commission that:

> Glaysher's statement to me as best as I can reconstruct it to report it to you at this time was that, while he did not disagree with the analysis that JSC and KSC had reported, that they would not give an unqualified go for launch as ice on the launch complex was a condition which had not previously been experienced, and thus posed a small additional, though unquantifiable, risk.[114]

Aldrich concluded the meeting by deciding to recommend that the countdown continue until the ice team could return to the pad just prior to launch and make a final assessment. Aldrich testified that he told Jesse Moore about Rockwell's reservations, explained his decision, and recommended that the launch proceed unless the ice team discovered that the situation had badly deteriorated.[115]

---

[110] OIS Channel 216, p. 353.
[111] Rogers Commission Report, Volume V, p. 1013.
[112] Ibid.
[113] Ibid., p. 1014.
[114] Ibid., p. 1025.
[115] Ibid.

The results of the meeting were reported back to Rockwell over the communications system:

RSR: MSR, this is MER.
UKN: Go ahead.
RSR: We just got a report in from Arnie and they're going to go ahead and go into the count. They're going to go out, sweep down the pad as best they can and remove as much ice as they can and go for the launch today.
UKN. We copied that.[116]

In their final report, the Ice Team found that ice on the MLP in direct sunlight had begun melting. They also found that icicles had begun to fall from the upper FSS levels. Ice cube sized pieces of these icicles were found within 10 feet of the left-hand SRB hole. The west MLP deck was swept clean of ice/icicles. The water troughs were checked and found to be forming ice, which was again removed using the "shrimp net."[117]

In analyzing launch films after the accident, NASA found that, contrary to expectations and analysis, ice from the Fixed Service Structure did reach and impact the Shuttle vehicle during liftoff. The report stated:

> Numerous launch films were viewed regarding FSS and RSS ice debris. A film (E-43) [Engineering Camera 43] looking directly in at the vehicle and FSS shows some ice falling straight down in the period between SSME ignition and vehicle ascent through approximately 20 feet. It shows that very many particles fell at approximately a 45° angle during the vehicle rise through 20 to 40 feet. This ice included sheet ice particles up to 6 in. x 6 in. and flowed down into the plumes at a point directly below the engine nozzles. Some of this struck the LH SRB. One downward looking camera (E-36) on the FSS clearly showed that a small amount of FSS ice debris reached the area of the LH SRB exhaust hole. Particles numbered 50-100 and were approximately ice-cube size. None of these or any other debris was observed to be ejected upward toward the Orbiter. Another film (E-18) looks upward from the SSME pit. This shows that after a vehicle rise of 10 ft. hundreds of ice particles flowed in below the main engine at a 45° angle. No Orbiter impacts are observed. Camera E-26 . . . reveals many small pieces of falling ice striking the LH2 TSM [liquid hydrogen tail service mast] in the period between SSME ignition and vehicle rise through approximately 25 feet. Due to aspiration, 50-100 small ice particles flowed into the LH SRB plume directly below the SRB nozzle as the vehicle rose through 4 to 25 ft. These films and others show fairly clearly that there was little or no debris damage to the oribiter [*sic*] during liftoff due to FSS/RSS icing for the conditions observed.[118]

In the summary and conclusions section of this report, the following statements appear:

---

[116] OIS Channel 216, p. 355.
[117] "Ice/Frost Team Evaluation Report," p. 5.
[118] NASA, "Hazard to Orbiter Tiles Posed by the Vertical Structure Ice: Mission 51-L," January 30, 1986, p. 3.

> On STS-33 (51L) the actual FSS/RSS ice movement, as proven by the photographic documentation did not conform to the predictions in two important respects:
>
> 1. The ice generally did not release until after SSME ignition.
> 2. The ice translated several times farther toward the vehicle than predicted.
>
> To do meaningful predictions of ice movement, the effects of aspiration must be considered. Similarly, the release time of the ice must be known.
>
> Until the above capability is available, it should be assumed that FSS/RSS ice would be released early and pulled by aspiration into contact with the vehicle. FSS/RSS ice thereby could be judged as a potential high risk to flight safety.[119]

The Committee has proceeded at some length to develop the conversations regarding ice that occurred on the morning of 28 January because they illuminate tendencies that are at variance with the careful attention to safety the Nation has come to expect from NASA. It is the Committee's view that the information developed by the discussions between members of the ice team and those that took place between Rockwell personnel on this subject should have led to the conclusion that "FSS/RSS ice . . . could be judged a potential high risk to flight safety."

The Committee also notes that, in his presentation to the STS Program Manager at the 9:00 a.m. Mission Management Team meeting, the ice team leader apparently did not inform Mr. Aldrich that he had earlier recommended that the launch be held due to the ice in the pad area. There is no indication in testimony to the Commission that Mr. Aldrich knew of the team leader's comment, "Well, I'd say the only choice you got today is not to go." Had it been presented to him in those terms, the later reluctance of Rockwell to recommend a launch might have been sufficient to cause Mr. Aldrich to recommend a launch scrub. In any event, the uncertainly present in connection with this discussion should have been sufficient to cause a delay in the lauch until the ice melted off the gantry. The unknown risk represented by the ice would then have been removed.

These conversations also indicate that the launch director was not operating in a manner the Committee would expect. Given his position as the senior official responsible for the preparation of the Shuttle for launch, the Committee would expect a healthy skepticism to underlie discussions he had with members of the launch crew. In contrast to these expectations these tapes demonstrate that the director was often reminding the engineering team about time, and spent much time questioning the ice team leader's analysis of the ice on the pad. There is also no indication that he took steps to see that the pad escape system was ready for use by the flight crew, if necessary.

---

[119] Ibid., p. 4.

Finally, Congressman Ron Packard discussed with witnesses at the Committee's hearings on July 25, 1986, ways to improve contractor participation in preparing the Shuttle for launch.

Mr. PACKARD. "Mr. Davis, you spoke regarding the companies having a voice in the decisionmaking, I presume, after the FRRs—that two week interim between launch and the readiness review system. Do you believe that the companies should have more voice, less voice, or have they had any voice in whether its a go or no-go?"

Mr. DAVIS.[120] "Well, I can tell you how it runs now. Up to and including the L-minus-one day review, there's no doubt that every company has a very strong voice; and, as a matter of fact, at the L-minus-one review, they are required to stand up and commit their hardware as go or no-go. And those are very unequivocal commitments, also. After that time, then the reviews are more mission management meetings that are held, and as you get down into the countdown, it turns into more of a real time polling of the people that are actually controlling the launch. In those latter meetings, we are not, I would say, formally involved in those unless there is some problem with the hardware itself, the External Tank hardware. We are in Firing Room 2 in a very significant presence; we are aware of what is happening in some of the consoles. We sit behind them; we do not operate them. We are polled by the Director of Engineering prior to the launch actually proceeding, so we are sort of polled in an informal manner. We are not asked at any time after the L-minus-one for a formal go or no-go. I believe it would probably be appropriate, in terms of the Commission's desires, that indeed we be more formally involved in the mission management meetings, and that at some appropriate late time in the launch count—and I would leave that to NASA to decide—that indeed the companies be asked to declare go or no-go."

Mr. PACKARD. "A quick answer, Mr. Murphy. Do you agree?"

Mr. MURPHY.[121] "Yes, I agree with what Rick has said. I think that we have found out that we commit ourselves, I guess, at 20 minutes and 9 minutes by the people who are manning the consoles, but it does not rise to the management level which it should, in accordance with what Mr. Davis has stated. We would like that opportunity also."

Mr. DAVIS. "I'd like to make one other comment on that. I have never felt that if I needed to stop a launch, I could not stop it. While I have not been asked for a positive go or no-go, the ability is always there if I decide no, to stop the launch."

Mr. PACKARD. "Mr. Jeffs, do you feel the same?"

---

[120] Mr. Richard Davis, President, Martin Marietta Michoud Aerospace.

[121] Mr. George Murphy, Executive Vice President and General Manager, United Technologies Booster Production Company.

Mr. JEFFS.[122] "Yes, I think the system should be formalized more. We have great visibility as to the problems and real times, being on the net and having CRTs [console displays] and people that are involved in depth, both at Downey and at Houston, who support it, even though it's at the Cape. But especially, when you have holds or delays and what have you, it needs to be—again—upgraded in real time with, I believe, the contractor's participation with NASA management right up to the launch decision point, and a little more formal process involved in the polling of the contractors."

Mr. PACKARD. "Mr. Murphy, if you'd had that system set up prior to the accident, would the flight—would it have still gone?"

Mr. MURPHY. "It would not have influenced our position at all. Our hardware—we had stipulations on what we required on the hardware during the whole period. They were met, and so we were in a 'go' posture as far as we were concerned. It would not have affected our position."[123]

The Committee believes that had the hardware contractors been required by NASA to formally declare their flight readiness, it would have removed the ambiguity in Rockwell's recommendations involving the ice on the Fixed Service Structure.

B. LAUNCH REDLINES

Section VI.B.1.b. of this report describes the rationale for development of certain criteria that serve to indicate when the Shuttle system is experiencing problems during the countdown. On the morning of January 27 and 28, during the countdown for STS 51-L, the launch crew in the Firing Room wrote waivers to certain of these criteria in order to permit launch of the Shuttle. Tapes and transcripts from the Operational Intercommunication System demonstrate that, at least in one instance, the technical analysis authorizing the use of a backup procedure did not account for ambient temperatures below the limits specified for this procedure. Thus, a waiver should not have been granted.

Revision C, Amendment 18 of the Launch Commit Criteria specified 45 degrees Fahrenheit as the minimum redline temperature for the External Tank nose cone.[124]

But the ambient temperatures during the countdown were well below that. On January 27, while the Shuttle waited for liftoff, conversations indicated that the nose cone heaters were not able to maintain proper temperatures. Excepts from the transcript of this discussion follow.

CF. Okay, go ahead, Fred.

FH. Okay, we may have a problem with propellant temperatures at that low level. We're about three degrees away from red line and losing ground right now.

CF. Because of the amount of heat that the ground system's able to put in there?

---

[122] Mr. George Jeffs, President, North American Space Operations, Rockwell International.

[123] Cmte. Hgs., Transcript, July 25, 1986, pp. 71-74.

[124] NASA, "Launch Commit Criteria and Background," Revision C, Amendment 18, JSC-16007, December 1, 1982, p. 5.1-4. See Appendix VIII-K.

FH. That's right, they're giving us all they can right now. . . .[125]

DIRECTOR. . . . What do you guys feel about all the temperatures we saw today, like nose cone, and all those? Think about that so we can talk about that a little bit.

M. Yeah, that's another thing we're not happy about. We think we could probably still get by with it, but we're marginal.

UKN. You know, nose cone temp. Horace, is probably gonna be down in the low 20's, maybe even below 20.

M. I think the intertank we ought to be able to keep it up high enough. . . .[126]

DIRECTOR. . . . Okay. You guys think that nose cone heater is putting out all we gonna be able to get out of it?

UKN. Yep. You got it full blast. You're gonna be down 18–20° tomorrow on the nose cone. And the waiver we wrote today said we're only good down to 28. That was today's. . . .

UKN. And Horace, the [intertank] heater was running full bore for quite awhile. And we were running at least 10° below the set point temperature.

DIRECTOR. What about Fred Heinrich? He on 245?

FH. I'm here, Horace.

DIRECTOR. Fred, what do you think about the flow rate, I mean the RCS temps?

FH. Okay, Grady said they can crank that thing up locally and get outside the OMRSD [Operations Maintenance Requirements and Specifications document] limit, which may be enough, but we need to get started as soon as we get in there. Otherwise, we can't get the tank warm enough. We're going to lose ground all throughut the cryo load.

DIRECTOR. Okay.

FH. We're about 3° away from red line right now. We lost some during this cryo load with full bore on the heaters.

DIRECTOR. Okay.

R. Horace, this is Robinson, We'll have to check with JSC about the upper limit on this temperature.

DIRECTOR. What do you mean?

R. —right now OMRSD is not in—it may be something else other than Fred Heinrich's temperatures.

DIRECTOR. Okay. Then you don't have no problem picking it up, though, that's the only requirement we got, right?

R. That's affirmative.[127]

The waiver referred to in this conversation offered the following technical rationale: "No visible ice buildup on the nose cap fairing exit area. Temperature is 12 deg F below redline."[128]

The waiver also read, "For STS-33 [51-L] Min LCC acceptable is 28 deg F (was 45 deg F). Ullage transducers are acceptable down to 28 deg F (was 40 deg F). Refer Note A, LCC 5.1-4."[129]

Note A read:

> "The following purge temps are backup measurements.
> GLOT 4104A PRI Nose Cone Heated Purge Temp.
> GLOT 4604A SEC Nose Cone Heated Purge Temp."[130]

These refer to telemetry data channels. As the temperatures are received via telemetry, they are to be interpreted by means of a curve shown on page 5.1-4B.[131]

It is important to note that Note A applies only if the ambient temperature is between 40 and 99 degrees F. Otherwise, the redundant procedure is invalid.

As for the effect of exceeding this redline, the launch commit criteria reads:

---

[125] OIS Channel 245, p. 217.
[126] Ibid., p. 218.
[127] OIS Channel 245, pp. 219–20.
[128] See Appendix VIII–L.
[129] Ibid.
[130] "Launch Commit Criteria and Background," p. 5.1-4.
[131] Ibid., p. 5.1-4B. See Appendix VIII–K.

"The minimum redline was established for two reasons.
A. L02 ullage pressure transducers calibrated to 40 deg F.
B. Avoid ice buildup at nose cone fairing exit."
"Consequences of exceeding redline;
1. Ice build up and possible impact to Orbiter.
2. Inaccurate ullage pressure readings." [132]

Engineering support communications continued with the following:

M. Horace, 245.
DIRECTOR. Yeah, go ahead.
M. Okay, one other thing's been brought to my attention. The, the LOX, the LOX ullage pressure transducers are calibrated to a minimum of 40 degrees and maximum of 140, which is what sets our minimum in the nose cone. Below that, we may get some variations in reading.
FH. No, that's not true, Mark.
M. Well, okay.
FH. (garbled) see I got some measure readings here from Mark was. . . .
M. That's what's on the LCC backup page.
UKN. Read the LCC backup page on the lower limit.
M. It says 40 degrees.
FH. It says for reasons of ice and frost at that the exit on the fairing.
M. Yeah.
MAC. Test on 245. (garbled) . . . 245.
DIRECTOR. Go ahead, Mac.
MAC. Hey, are you guys reading this LCC that the consequences of exceeding the nose cone temp redline? The sheet we have over here says that we will get inaccurate ullage pressure readings.
DIRECTOR. Okay, Mac, we understand, thank you.[133]

On the morning of January 28, the following discussions between engineering support personnel were also directed toward waiving the launch commit criteria on the External Tank nose cone.

UKN. Dave, they were, of course, expecting to violate those ET nose cone purge temps LCCs again. It's 20 degrees colder today than it was yesterday.
D. Well, I guess we'll be going down the line producing a waiver to the same effect that we produced yesterday.
UKN. How did we just define it yesterday for 51L? Or for this attempt for 51L?
D. Copy your whole question. Say again?
UNK. I thought we just annotated it yesterday as for 51L only.
D. That may be it, I'll check the waiver log. . . .
UNK. Dave? 161.
D. Go ahead.
UNK. Yeah, it is effectively 51L, I think we are in good shape.
D. OK.[134]

Later on the morning of the 28th, the following discussion occurred:

D. FR[Firing Room] 2, this is FR[Firing Room] 1.
UKN. Yeah Chris.
D. This is Dan. We need to send a waiver over for signatures. We're right now showing nose cone gas temps that we were discussing yesterday are down in the 12-16 degree range and the waiver that we wrote yesterday for 51L only gives us allowance down to 28 degrees F.
UKN. Yeah, we'll have to rewrite that waiver.
D. OK. You don't think we'll have any trouble getting that signed?
UKN. No, as long as our pressure transducers are OK.

---

[132] Ibid., p. 5.1-4.
[133] OIS Channel 245, pp. 220-21.
[134] OIS Channel 161, p. 289. Channel 161 is identified as an Engineering channel used for troubleshooting and systems integration.

D. OK. What number would you like to use on that Horace? [135]

Shortly after this conversation concludes, the discussion on preparing the ET nose cone temperature waiver continued. Told "[W]e're right now sitting at 10 [degrees]" KSC's Director of Engineering said:

DIRECTOR. Let's hold on a minute and then we'll write the waiver, we'll probably wanna go below 10.
UKN. Yeah, we may just want to say no low and put a note on there that based on the pressure transducers.
Director. OK, so you want us to stand by and wait on that?
UNK. Yeah.
DIRECTOR. OK. We'll be waiting. [136]

If engineers intended to apply the same rationale for waivers as that used on Janury 27, the rationale is invalid. Ambient temperatures were well below the 40 degree limit necessary for a valid backup procedure. Therefore, the backup procedure should not have been employed.

Later, in a discussion between the Launch Director and the Director of Engineering regarding countdown problems, the following discussion about the temperature waiver took place.

LD. OK. I understand we're in the process of writing a new waiver with a lower limit of 10?
DIRECTOR. We're still looking. We'll give you a low limit.
LD. What are they running today?
DIRECTOR. It's been down as low as 10 basically.
LD. Wow! [137]

It was in a subsequent discussion between the same principals that the new limit was established. Their conversation, however, does not reflect that the limit was chosen by rigorous technical analysis.

LD. We have nothing else, Horace, not unless you guys are working something.
DIRECTOR. No, just the ice.
LD. OK. The only outstanding item we have right now is the one waiver on the cone temps.
DIRECTOR. OK. It looks like we probably could say about 10° and be OK on that one.
LD. OK. We'll use 10° then.
DIRECTOR. OK. [138]

Completing preparations of the waiver, the Director had the following conversation with one of the technicians.

DIRECTOR. OK, Jackie, in writing this waiver for nose cone temps we want to put the words on here saying that we can use the ullage transducer as an alternative way of determine redline, but we believe that yesterday Warren Wiley was saying that they had looked at those and they're good down to approximately 11 degrees and we wanted to verify that.
J. I think they did to 10 degrees.
DIRECTOR. 10?
J. Yeah.
DIRECTOR. The ullage transducers.
J. OK. Thank you, Horace. [139]

---

[135] Ibid., p. 296.
[136] Ibid., p. 297.
[137] Ibid., p. 299.
[138] Ibid., p. 302.
[139] Ibid., p. 305.

There is no documentation to describe the method by which the ullage pressure transducers were qualified to "10 degrees." No alternative analysis is described on the launch commit criteria to support use of this low temperature as a rationale for approving a waiver.

This example indicates that NASA personnel do not necessarily employ a sufficiently rigorous engineering analysis to the waiver of launch commit criteria during countdown. There also appears to have been some confusion as to the effect of exceeding the redline temperature. Ullage pressure readings may be critical parameters if fed to the Main Engine controllers during flight. According to NASA:

> Following engine ignition at about T-4 seconds, the ullage pressure is supplemented using propellant gases vaporized in the engine heat exchangers and routed to the two ET propellant tanks. The tank pressure is maintained based on data inputs from ullage pressure sensors in each tank to control valves in the Orbiter. A combination of ullage and propellant pressure provides the necessary net positive suction pressure to start the engines. The net positive suction is the pressure needed at the main engine pump inlets to cause the pumps to work properly. The pumps, in turn, supply high-pressure liquid oxygen and liquid hydrogen to the thrust chamber. Acceleration pressure is added for operation. Fuel is forced to the engines primarily by tank pressures and, to a lesser degree, by gravity.[140]

Inaccurate readings from these sensors might cause the engines to operate improperly.

Also, according to the launch commit criteria, violations of launch redlines may also have occurred on the Auxiliary Power Unit (APU) gearbox lube oil (minimum redline temperature 42 deg F), and the fuel test lines (minimum temperature 41 deg F), if in fact the actual temperature were lower than minimum.[141]

There was also no mention of the 34.2 deg F minimum redline temperature for the SRB recovery batteries or what the temperature of the batteries was at launch.[142]

Under "Remarks," this criterion states, "Violation of this redline shall require an assessment to determine if a hazard exists which jeopardizes the Shuttle. . . ."[143]

Mr. Mulloy testified before the Commission that:

> Mr. MULLOY. "I had a discussion on my SRB loop with the SRB people dealing with the question of a 24-hour turnaround to attempt to launch again at 9:38 on the 28th and the effect that the predicted cold temperatures for the night of the 27th might have on that.
>
> The input was received back both to Mr. Reinartz and myself that we were looking at the Launch Commit Crite-

[140] NASA, "Space Shuttle News Reference," 1981, p. 2-40.
[141] "Lauch Commit Criteria and Background," Amendment 20, p. G-23.
[142] Ibid., p. G-1
[143] Ibid.

ria relative to temperatures. It was felt there was a need to look at the recovery battery temperatures that are in the forward skirt of the SRB and the fuel service module temperatures that are in the fuel service modules for the thrust vector control system in the aft skirt of the solid rocket booster.

The input received back by me was that they did not feel that would be of any concern. They were going to continue to look at it, and if any concern arose they would let me know.

I went to the 2:00 Mission Management Team and reported that there were no constraints to the solid rocket booster for a 24-hour turnaround, that we had taken a look at the recovery battery temperatures and the fuel service module. We did not feel at this time that there would be any Launch Commit Criteria for the low temperature limits that were established for those systems, but that we were continuing to assess that; should anything change in that regard, I would so report that."

Chairman ROGERS. "You referred to the Launch Commit Criteria. What were they as far as you knew in terms of weather conditions? Any?"

Mr. MULLOY. "In terms of weather conditions, yes, sir, I'm aware that there is a Launch Commit Criteria for the system for weather. There are a number of factors in that Launch Commit Criteria. One of them is the ambient temperature, which is established at 31 degrees.

Another is the sea state and winds in the SRB recovery area. Another is the cross-winds at the return to landing site runway at Kennedy Space Center. Another is the trans-Atlantic landing site weather, and another is severe weather, which is related to lightning and thunderstorms in the area."

Chairman ROGERS. "And when you say there were no constraints in the 2:00 meeting, does that mean that as far as you could see there were no problems in those areas?"

Mr. MULLOY. "No, sir, I did not evaluate those areas of the Launch Commit Criteria. What I was looking at was the specific Launch Commit Criteria items that are on the solid rocket booster and the effect that the low temperatures would have on that.

I would expect Mr. Aldrich would normally make the judgements on, and his people at the Johnson Space Center, would make the judgements on crosswinds and trans-Atlantic weather and the general ambient environment for launch."[144]

While there is no reason to believe that these waivers directly contributed to the cause of the accident, the low temperatures during the night of the 27th and morning of the 28th most probably did. Committee staff learned in discussions with Thiokol per-

[144] Rogers Commission Report, Volume V, pp. 827-28.

sonnel[145] that liquid hydrogen apparently remained in the External Tank throughout the night of January 27. This most likely played a role in the joint seal failure, since it permitted heat transfer through the ET/SRB aft attachment strut throughout the night. Of equal interest, however, is the fact that ET requires an eight-hour recovery period between tanking cycles, measured from the time the hydrogen tank low-level sensors are dry.[146]

If the hydrogen tank was never emptied during the turnaround procedure, this would represent a violation of those criteria. Had the criteria been observed, STS-51L would have required an afternoon window on January 28, or it might have been necessary to attempt the launch on January 29. This has not been independently confirmed, however.

[145] Discussion with Carver Kennedy, Thiokol Wasatch Operations, Brigham City, Utah, September 4, 1986. General Kutyna also noted this in the Commission's hearing on February 14, 1986 (Rogers Commission Report, Volume IV, p. 660).

[146] OIS Channel 245, p. 218.

IX. DEFINITIONS OF TERMS AND ACRONYMS

AA-SF—Associate Administrator for Space Flight.
AFPRO—Air Force Plant Representative Office.
APU—Auxiliary Power Unit.
BOC—Base Operations Contractor.
BTU—British Thermal Unit.
CAR—Configuration Acceptance Review.
CDR—Commander.
CDR—Critical Design Review.
CIL—Critical Items List.
CoFR—Certification of Flight Readiness.
CPIF—Cost-plus, Incentive-fee.
CTS—Call-to-stations.
DAR—Deviation Approval Request.
DCAS—Defense Contract Administration Service.
DCR—Design Certification Review.
DFRF—Dryden Flight Research Facility.
DR—Discrepancy Report.
EG&G—Edgerton, Germeshausen and Grier.
ESMC—Eastern Space and Missile Center.
EST—Eastern Standard Time.
ET—External Tank.
FEAT—Flight Element Assignment Table.
FDO—Flight Dynamics Officer.
FMEA—Failure Modes and Effects Analyses.
FRR—Flight Readiness Review.
FSS—Fixed Service Structure.
GOX—Gaseous oxygen.
GSE—Ground Support Equipment.
HA—Hazard Analyses.
HDP—Holddown Posts.
ILL—Impact Limit Line.
IPR—Interim Problem Report.
IR—Infra-red.
IUS—Inertial Upper Stage.
JSC—Johnson Space Center.
KSC—Kennedy Space Center.
ksi—thousands of pounds per square inch.
LCC—Launch Control Center.
LFC—Left Forward Center.
L/H—Left Hand.
LOS—Loss of signal.
LOX—Liquid oxygen.
LPS—Launch Processing System.
LRU—Line Replaceable Unit.
MEOP—Maximum Expected Operating Pressure.
MER—Mission Evaluation Room.

MLP—Mobile Launch Platform.
MMT—Mission Management Team.
MRB—Material Review Board.
ms—millisecond.
MSFC—Marshall Space Flight Center.
MST—Mountain Standard Time.
NASA—National Aeronautics and Space Administration.
NRC—National Research Council.
NRP—National Resource Protection.
NSTS—National Space Transportation System.
NTD—NASA Test Director.
OASCB—Orbiter Avionics Software Control Board.
OIS—Operational Intercom System.
OM—Operations Manual.
OMI—Operations Maintenance Instruction.
OMP—Operations and Maintenance Plan.
OMRSD—Operations Maintenance Requirements Specification Document.
OMS—Orbiter Maneuvering System.
OPF—Orbiter Processing Facility.
PAC—Problem Assessment Center.
PAS—Problem Assessment System.
PDR—Preliminary Design Review.
PGHM—Payload Ground Handling Mechanism.
PR—Problem Report.
PRCBD—Program Requirement Change Board Directive.
psig—pounds per square inch gage.
PSP—Processing Support Plan.
QC—Quality Control.
RCS—Reaction Control System.
R.F.—Radio Frequency.
RPSF—Rotation, Processing and Surge Facility.
RSO—Range Safety Officer.
RSS—Range Safety System.
RSS—Rotating Service Structure.
RTLS—Return to Launch Site.
SCA—Shuttle Carrier Aircraft.
SPC—Shuttle Processing Contractor.
SRB—Solid Rocket Booster.
SRM—Solid Rocket Motor.
SR&QA—Safety, Reliability and Quality Assurance.
SSME—Space Shuttle Main Engine.
STS—Space Transportation System.
TBD—to be determined.
TDRS—Tracking and Data Relay Satellite.
TM—Telemetry.
TSR—Technical Status Review.
TVC—Thrust Vector Control.
VAB—Vehicle Assembly Building.
VPF—Vertical Processing Facility.
WAD—Work Authorization Document.

X. APPENDICES

V-A

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546

REPLY TO ATTN OF: HO-1

MEMORANDUM

TO: Marshall Space Flight Center
Attn: Mr. Garland G. Buckner
~~Acting~~ Procurement Officer

THRU: Marshall Space Flight Center
Mr. Roy E. Godfrey
SEB Chairman

FROM: HO-1/Chief, Operations and
Review Division

SUBJECT: Selection of Contractor for Space Shuttle Program
Solid Rocket Motors

Subject statement, signed by the Administrator, is enclosed for inclusion in the official contract file. Also enclosed is a copy of the Administrator's Statement for the SEB Chairman.

William E. Stuckmeyer

Enclosure

copy to Roy Godfrey

* * * * * * *

THIOKOL CHEMICAL CORPORATION

Thiokol presented an approach to the SRM Program which clearly focused on maximum utilization of existing facilities and low early year funding. In-house production effort would be accomplished in the Wasatch Division, Utah facility. Increment III production would be accomplished by acquisition of portion of the adjacent Air Force Plant 78 as Air Force requirements phased out. AP requirements would be met by increasing the capability of existing facilities in nearby Henderson, Nevada. Use of an existing, skilled, stable work force in a low labor rate area would minimize new hires and provide low labor costs. Thiokol's decision to fabricate nozzles in-house provided cost savings and good control over this extremely critical component; however, the Board concluded that this introduced some early risk because of lack of experience in fabricating nozzles of this size. Facility location resulted in high transportation cost of the SRM's; however, these costs were more than offset by low facility investments. The Thiokol proposal received the second highest overall Mission Suitability score by the SEB, being tied with UTC. The SEB ranked Thiokol fourth under the Design, Development and Verification Factor, second under the Manufacturing, Refurbishment and Product Support Factor and first under the Management Factor.

Design, Development and Verification

The Thiokol case design met the general SRM requirements; however, the cylindrical segment was close to the upper limits of size capability of the case fabricator. The nozzle design

included ablative materials not currently developed or characterized. This offered potential savings in program cost, but with attendant technical and program risk. An expanded characterization and development program would be required. The thickness of the nozzle material was insufficient to meet required safety factors and thus degraded reliability. The amount of material required to correct the deficiency was substantial and the deficiency could require a redesign of the metal portions as well as the ablative portions. The design was complex and would contribute to difficulty in manufacturing. The Thiokol motor case joints utilized dual O-rings and test ports between seals, enabling a simple leak check without pressurizing the entire motor. This innovative design feature increased reliability and decreased operations at the launch site, indicating good attention to low cost DDT&E and production. The thickness of the internal insulation in the case aft dome was marginal and created a technical risk.

Thiokol provided comprehensive test plans and development verification objectives; however, they proposed to verify propellant burning characteristics by testing four to six full scale mixes which was excessive, and could be reduced by establishing correlations with smaller mix size data during DDT&E. Also, Thiokol proposed to hydroburst two motor case assembly specimens, whereas one test would be sufficient.

Manufacturing, Refurbishment and Product Support

Thiokol had extensive processing experience with their proposed propellant formulation, having processed over 150 million pounds of this general type of propellant. Thiokol's major weakness in this area of evaluation was in the area of case fabrication. The segment fabricator would be unable to fabricate the case segments strengthened with stiffening rings as proposed by Thiokol for alternate water entry load conditions, if required. This would probably require a case and grain redesign. Thiokol's manufacturing approach provided a good mechanized method of installing insulation, coupled with an innovative method of preparing the insulation surface for the liner by peeling off a dacron cloth from the inner surface of the insulation. A minor weakness in the manufacturing approach was the decision to fabricate nozzles in-house due to Thiokol's lack of experience in fabricating nozzles of this size.

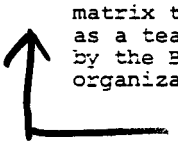
Thiokol proposed to utilize existing facilities which, with minor modifications, were totally adequate for all three increments. The one exception to this was a failure to meet Quantity Distance safety requirements between casting pits for Increment III, however, there are ways to adequately cure this problem. Thiokol maximized the refurbishment of components and the potential cost savings provided by refurbishment. Another less significant strength was the enhancement of segment assembly provided by three alignment pins thereby reducing the assembly hours on the launch pad. Thiokol failed to provide enough new cases and nozzles to meet the launch schedule. Eight additional cases and nozzles would be required to provide assurance that launch dates could be met.

Management

Thiokol structured the development program so that all major costs were deferred to the latest practicable date. This resulted in low early year funding, which is a key program objective. The availability of an operating plant, with ample experienced personnel and a proven organization which could be phased to the SRM effort with minimum modification added considerable maturity and confidence and proved to be cost effective. The Board considered this to be a major strength for all three increments. A strong matrix management was evident and key line organization supervisors were experienced and had worked together as a team on many successful development and production programs such as Minuteman and Poseidon. Strong management participation and visibility in variance analysis was another strong feature as was the approach to corrective actions and their effect on estimate-to-complete. Procurement Management was thorough and well planned. SRM commodity purchases would be consolidated with that of other programs at Wasatch, which should result in lower cost. The Procurement of major items was well matched to overall SRM schedule requirements. Thiokol proposed a strong Configuration Management System which included thorough identification and traceability during DDT&E, production and refurbishment. The tentative decision to make the molded and tape wrapped nozzle in-house was considered a strength in this area. It would contribute to the low cost-per-flight goal by using available resources, avoiding subcontract fees, lowering overhead rates, and taking advantage of lower cost labor. The inherent risk management aspects also were considered.

In the area of Key Personnel, the proposed Program Director was considered exceptionally strong and had successfully performed as a Project Manager on other major programs. He is widely known for his excellent performance. The proposed Deputy Program Director would also be the Chief Project Engineer. He had important and successful engineering management roles in previous major motor programs and has an excellent reputation in the trade.

Although adequately qualified for their proposed assignments, the proposed Functional Managers and their Team Members in the Project Organization did not reflect the depth of experience available in the Functional Departments of the Thiokol matrix type organization and had not previously performed as a team. This was not considered a significant weakness by the Board because of the strong experienced matrix organization at Thiokol.



* * * * * * *

V-B

TWR- 14359

PROGRAM PLAN

PROTECTION OF SPACE SHUTTLE SRM PRIMARY MOTOR SEALS

4 MAY 1984

DR NO. 5-6

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA 35812

CONTRACT NAS8-30490

WBS 1.1.1.2.3

Thiokol/WASATCH DIVISION
A DIVISION OF THIOKOL CORPORATION
P O Box 524, Brigham City, Utah 84302 801/863-3811

FC 0553-

FORM TC 4577 (REV 3-79)

PROGRAM PLAN

PROTECTION OF SPACE SHUTTLE SRM
PRIMARY MOTOR SEALS

Prepared by:

B. C. Russell, Manager
SRM Ignition System

Approved by:

R. V. Ebeling, Manager
Igniter, Final Assy,
& Static Test

A. J. McDonald,
Director
Solid Rocket Motor

Released by:

J. C. Tibbitts
Data Management

PROGRAM PLAN

PROTECTION OF SPACE SHUTTLE SRM
PRIMARY MOTOR SEALS

1.0 INTRODUCTION

There have been incidents on SRM flight and static test motors where a primary o-ring has been slightly charred by hot gases which penetrated through the vacuum putty barrier. Motors affected thus far are STS-2A aft field joint, QM-4 nozzle joint, STS-11A forward field joint, STS-11B nozzle joint, and STS-13A nozzle joint. This program plan will result in defining the solution to this o-ring char.

2.0 OBJECTIVE

The program objectives are to systematically isolate the problem and to eliminate damage to SRM seals.

3.0 APPROACH

The program approach will consist of analysis, subscale (hot) tests, full-scale joint tests, and final verification in motor static testing.

The analysis will attempt to identify the cause of o-ring erosion, its acceptability, and justification. It will identify specific design or process changes which will eliminate further o-ring charring. Studies will be performed showing the effects of material variation characteristics, putty layup configurations, and fresh materials versus environmentally exposed putty. In conjunction with these analyses, a thorough study will be performed on material from alternate sources.

Testing will include laboratory material characterization, small motor hot tests simulating effects on cavity volume variations, flow patterns, exposure of o-ring and lubrication effects. The burn time of small hot motors will be in the range of 3 ~ 30 seconds depending on the results of previous small scale motor test results. Morton Thiokol also recommends that actual full-scale segment joint tests be used to evaluate pressurization effects on putty layup arrangements and flow changes due to final assembly of the joints. It is further suggested that a group of experienced people from MSFC and MTI be selected to witness the entire joint preparation, assembly, leak testing, and postfire teardown at KSC and MTI/Clearfield facilities. This team will also review all analyses, laboratory tests, subscale hot test, and support team reviews.

3.1 Analysis

The following tests shall be performed on the vacuum putty as a minimum. If further testing is required, it shall be performed and documented. Some of the tests are currently being performed on the existing putty, but are listed to assure that presently available data are summarized in the ensuing report. These tests will have to be repeated on any potential new putty. (See section 3.1.5)

3.1.1 Chemical Composition

An analysis will be performed on the putty to determine solids content, asbestos fiber (or other filler) content, chromate content, binder makeup, and all other applicable tests described in STW4-2847, Putty, Vacuum Seal.

3.1.2 Physical Properties

Tests shall be developed and conducted to determine adhesive strength of the putty (tackiness), strain capability, compressibility, and resistance to heat, erosion, and pressure shock (at SRM ignition).

3.1.2.1 Environmental Effects

The putty will be conditioned in controlled temperature and humidity environments, including ambient conditions at Utah and Florida, then tested for all physical properties and appropriate chemical properties such as non-volatile content and water solubility.

3.1.3 Aging

An aging program will be conducted on putty in Utah and Florida. The program should run for five years with particular emphasis during the first year at 3, 6, 9, and 12 months. Chemical and physical tests shall be performed at each stage of aging. The putty will also be checked for shrinkage and silicone migration from the paper backing used as a separator in the roll form.

3.1.4 Compatibility of Putty With Other Materials

The putty will be tested to determine effects of its mixing with Conoco HD-2 grease, cured NBR rubber, and both fresh and saltwater. If the materials react, properties of the resultant material will be established. Tests shall also determine whether the resultant material is corrosive to the D6AC case.

3.1.5 Second Source for Putty

A second (or third) source of putty is desirable to prevent further supply problems, which could seriously impact the Space Shuttle program. A development program will be implemented to test alternate putty candidates per the preceding requirements as a minimum. Subscale firing tests shall use the alternate putty to establish confidence to install the new putty in FWC-SRM static firing DM-6 for putty qualification.

There are three alternate source candidates at this time: Plastic Sealer 579.6 from Inmont Corporation of Georgetown, Ontario, with asbestos and Plastic Sealer 579.6 from Inmont Corporation, St. Louis, without asbestos. General Sealants is developing a high temperature putty that will also be screened.

3.1.6 Viton Characteristics

To aid the accuracy of the hot gas jet analysis, tests will be run to determine the erosion rate of Viton. These data, along with results from other tests described in this plan, will be used when the analysis is redone.

3.2 Subscale Firing Tests

3.2.1 Five Inch CP Motor

To verify that hot gas jets through the putty openings is correct, tests will be conducted which induce a gas jet impingement on an o-ring using five inch CP motors as test beds. Under tightly controlled conditions (environmental and mechanical), this data will be assessed to more fully understand what is happening in the SRM applications.

If a meaningful subscale joint test can be devised with putty in it, it will also be performed using the five inch CP as the hot gas source.

3.2.2 40 Lb. Char Motor

Depending on the results of the five inch CP hot testing, it may be desirable to include larger scale test motors having putty installed. Morton Thiokol is investigating the 40 lb. char motors. If required, such tests shall be performed to further verify the change in putty layup, type, and/or other filler materials.

3.3 Full-Scale Joint Tests

Tests shall be performed using a full-scale SRM field joint to verify the subscale results of the candidate putty layup configurations as affected by the actual joint assembly and leak test procedures. The "short stack" hardware is preferred for use instead of SRM case segments for ease of assembly and inspection. The following questions shall be answered as minimum requirements of this test sequence:

a. What is post assembly pressure in the cavity between the putty and the primary o-ring?

b. What is the minimum pressure required to blow through the putty?

c. What is the maximum acceptable leakage rate at 200 psig which meets the 50 psig leak test criterion?

d. How is o-ring seating affected by pressure and time?

e. What is the proper amount of HD-2 grease to be applied to the joint metal surfaces to minimize the free volume between the vacuum putty and the primary o-ring?

f. What is the effect of case eccentricity during segment mating on the flow of vacuum putty in the joint?

g. What dimension and weight controls are required to assure the vacuum putty layup is consistent and adequate?

Potential fixes will be investigated such as inducing paths through the vacuum putty at regular circumferential intervals to prevent localized o-ring damage caused by small, supersonic gas jets. The concept of a soft rubber barrier between the putty and primary o-ring will also be investigated. In addition, leak check procedures, particularly those employing the use of a flow meter, will be examined with acceptable and non-acceptable (leaking) o-rings. In all instances the behavior of the putty shall be closely monitored.

Results of the above described tests will be extrapolated to the nozzle to case joint and tests using a full-scale nozzle fixed housing and aft dome will be conducted, if necessary, to verify the adequacy of any change resulting from the field joint tests.

A plan, TWR-13983, has been prepared to check the putty configuration of the igniter to case joint. These tests will also be conducted and the results will be summarized in the final report.

3.4 Full-Scale Static Test

All potential design changes will be adequately tested on the subscale level and shall be incorporated into the SRM-FWC static firing DM-6 for qualification. A critical postfire inspection will be performed on the new configuration as well as the baselined portions of the DM-6 joints.

An analysis will be performed to assess the results of the FWC-SRM field joints as they compare to the HPM-SRM field joint. The field joints are shown in figures 1 and 2 for HPM-SRM and FWC-SRM, respectively.

3.5 Review and Witness Team

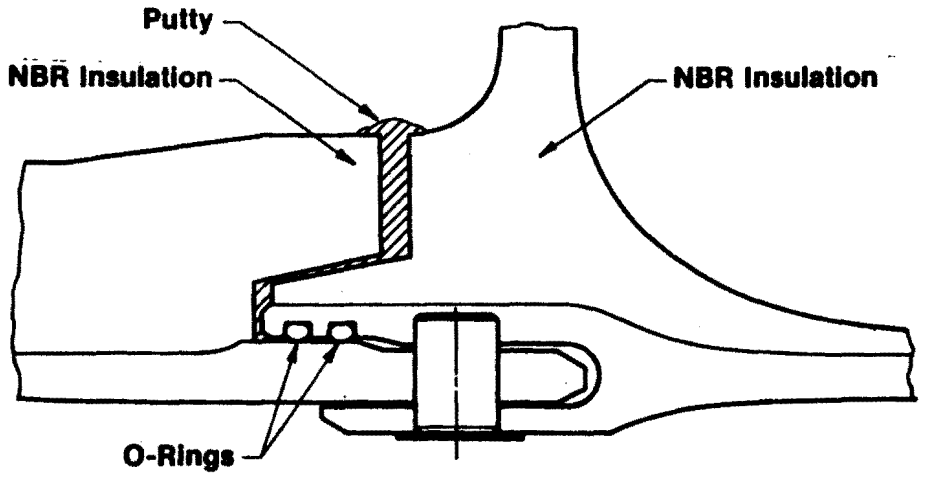
A review and witness team shall be established consisting of experienced engineers from Morton Thiokol and NASA to inspect and assess all test results. Anomalous conditions of joints from flown motors shall be critically inspected by members of this team. The team will determine the course of action to be taken as intermediate and final results become available.

4.0 SCHEDULE

The attached schedule reflects the time available to complete the testing and qualification of the preceding items.

5.0 REPORT

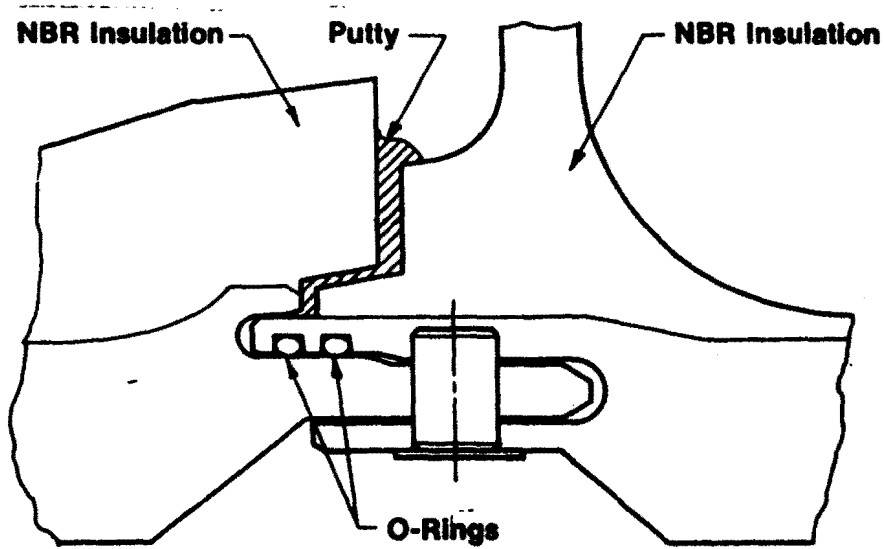
A comprehensive report shall be prepared by Engineering documenting the tests and results. The original shall be released after the development testing, but prior to DM-6. A revision shall be released after DM-6 and a second and final revision shall be published after the aging program.



SRM-HPM Field Joint

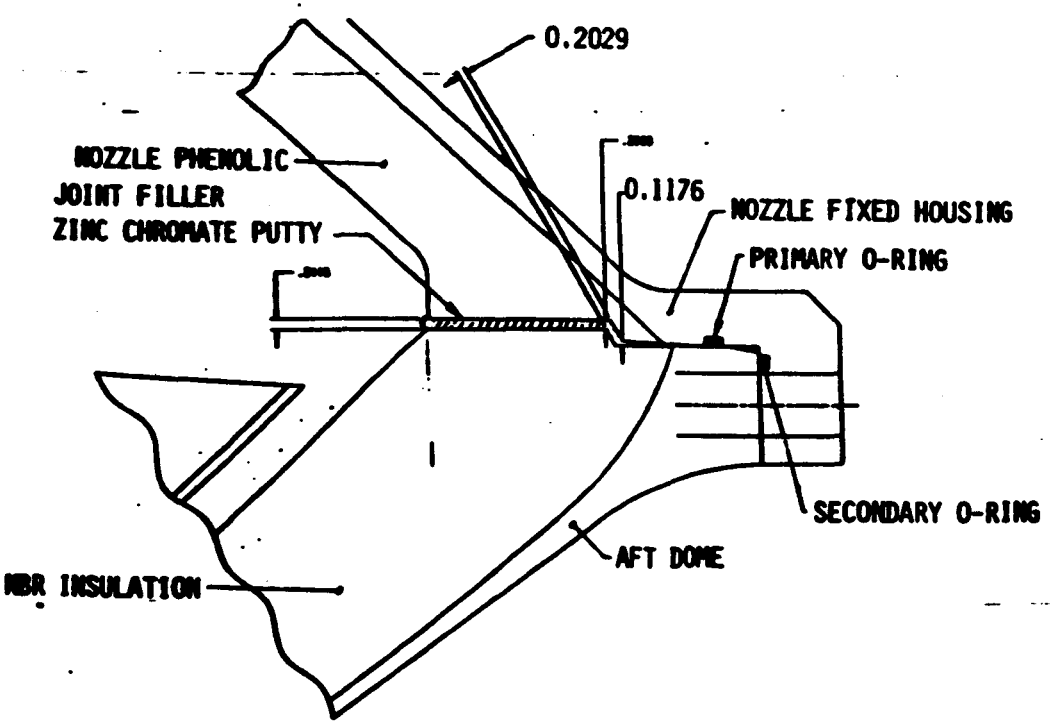
Figure 1

Thiokol Corporation, a Subsidiary of
MORTON THIOKOL, INC.
Wasatch Division



FWC-SRM Field Joint

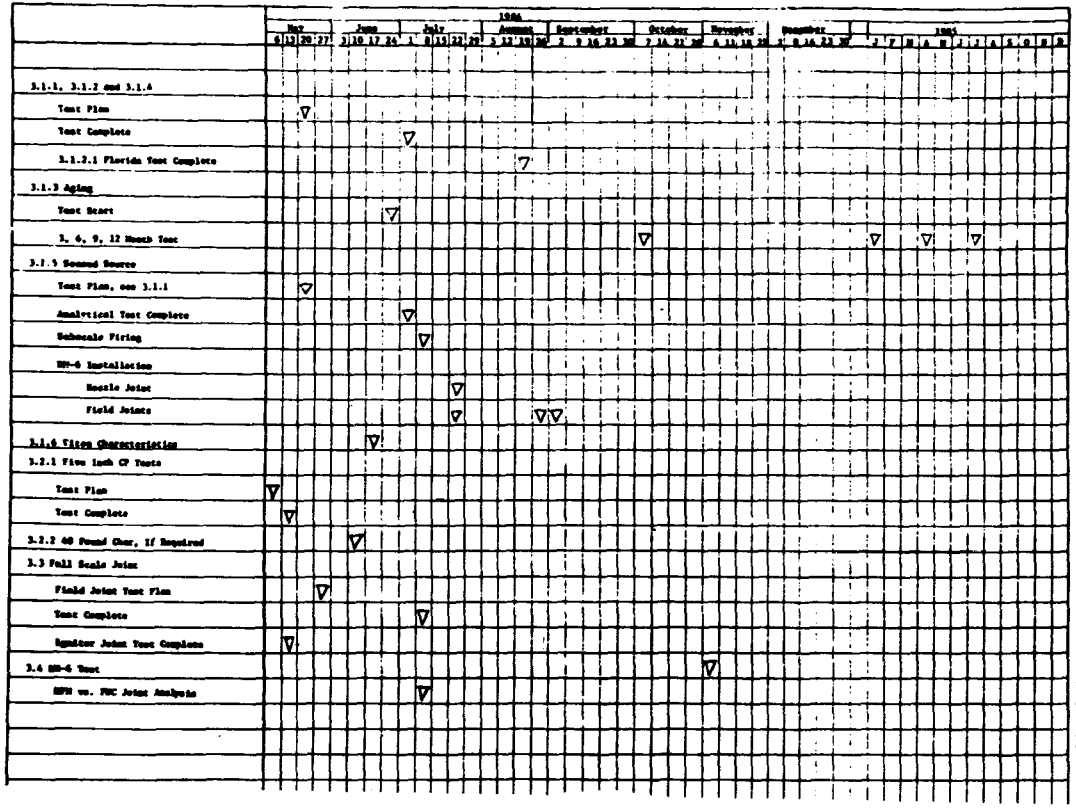
Figure 2



SRM NOZZLE-TO-CASE JOINT

SRM IGNITER-TO-CASE JOINT

PROGRAM SCHEDULE



DISTRIBUTION

NAME	QUANTITY	MAIL STOP
J. C. Kilminster	1	E00
A. J. McDonald	1	E00
R. V. Ebeling	1	E11
L. G. Bailey	1	E11
H. H. McIntosh	1	E03
J. R. Elwell	1	E12
B. G. Russell	1	E11
J. H. Keller	1	E12
F. C. Brasfield	1	E12
B. C. Brinton	1	250
P. C. Petty	1	250
R. M. Boiejoly	1	250
S. B. Pendleton	1	281
D. J. Hammon	1	280
L. H. Sayer	1	281
K. R. Eckhardt	1	281
R. C. Bunker	1	281
K. E. Wollenhaupt	1	281
G. G. Gorman	1	281
P. S. Shadleskey	1	281
A. R. Thompson	1	287
E. D. Brown	1	243
D. B. Davis	1	243
P. G. Russell	1	242B
F. E. Bares	1	240
S. H. Cardall	1	242B
W. B. Johnson	1	574
D. M. Fullmer	1	574
J. W. Galbraith	1	913B
D. M. Puskedra	1	913B
A. J. Burke	1	168
J. Trenkle	1	168
W. Coiner	1	168
F. W. Call	25	E14
M. E. Parker	1	MSFC
D. H. Harper	1	MSFC
W. L. Hankins	1	221
G. H. Schick	1	221

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

EP25 (84-49) May 23, 1984

TO: EE11/Mr. Horton

FROM: EP25/Mr. Miller

SUBJECT: Evaluation of TWR-14359, "Program Plan, Protection of Space Shuttle SRM Primary Motor Seals"

The subject Program Plan has been evaluated as requested and the following comments are submitted for your consideration:

a. Page 3, Paragraph 3.1.2, **Physical Properties** - This Program Plan mentions compressibility testing of the zinc chromate putty, however no laboratory tests are proposed which will determine the extrusion characteristics (displacement in a free volume under compression load) of various candidates. This should be accomplished to provide a better understanding as to why various types of putty exhibit unlike extrusion patterns with identical layups.

b. Page 4, Paragraph 3.1.5, **Second Source for Putty** - A second source for zinc chromate putty is desired and needed, but due to the poor performance of the Randolph putty, a more immediate need exists for development of a replacement for the present Randolph putty. Recommendation to this effect was made in Memorandum EP25 (84-35).

c. Page 4, Paragraph 3.2.2, **40-Pound Char Motor** - The 40-pound char motor should be made a definite part of the Test Program. It is vital that the test article be capable of simulating the total joint configuration as close as possible, which includes zinc chromate putty. Provisions for the installation of the putty together with extended burn time and increased volumes capabilities are achievable with the larger motor and should be included in the total program.

d. Page 5, Paragraph 3.3, **Full-Scale Joint Tests**, Reference: first paragraph following "g". Please explain how potential fixes such as inducing paths through the putty at regular circumferential intervals and use of a soft rubber barrier between the putty and primary O-ring will be verified by hot firing prior to installation on DM-6.

e. Page 5, Paragraph 3.3, **Full-Scale Joint Tests**, Reference: Second paragraph following "g". The Test Plan should specify a hard requirement to verify all potential nozzle/aft dome joint changes on full-scale hardware. The case joint and nozzle joint configuration differences warrant separate full-scale nozzle/aft segment assembly tests.

25

PC 026425

f. **General** - Design changes to the insulation interfaces which will prevent degradation of the thermal barrier due to joint rounding under pressure should be investigated as a part of this effort. The present design of the case joint and nozzle interfaces where the zinc chromate putty is installed are oriented such that the joint gaps can vary from minimum to maximum dimensions around the circumference during assembly due to out of roundness and eccentricity. This condition which is present to some degree during every joint assembly operation, guarantees that some, or almost all of the zinc chromate putty in certain areas will be wiped off when the mating surfaces move parallel to each other during mating. This results in open insulation gaps with insufficient zinc chromate putty during motor operation because the joints tend to become round and concentric when the case is presssurized internally.

Questions concerning this memorandum should be referred to Mr. William L. Ray, 3-3809.

for John Q. Miller
Chief, Solid Motor Branch

cc:
EA01/Mr. Hardy
EE01/Dr. Littles
SA41/Mr. Mulloy
SA42/Mr. Wear
SA42/Mr. McIntosh
EE11/Mr. Coates
EE11/Mr. Jones
EP01/Mr. McCool
EP21/Mr. McCarty
EP25/Mr. Powers
EP25/Mr. Ray

PC 026426

V-D

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

Reply to Attn of EP25 (83-119) December 6, 1983

TO: EE51/Mr. Horton

FROM: EP25/Mr. Miller

SUBJECT: Request for Tests by the Contractor to Obtain Space Shuttle SRM Clevis Joint, Fixed Housing/Aft Segment Joint and Igniter Adapter/ Forward Segment Joint Leak Check Data

It is requested that you take formal action to assure that the following tests are performed in a timely manner by the contractor, Morton/Thiokol, on SRM Hardware:

a. Case Clevis Joint Dual O-Ring Seal Leak Detection - Perform tests with full scale clevis joint hardware (short joints) to obtain the following data as a minimum:

(1) Post assembly pressure in the zinc chromate sealant cavity.

(2) Minimum and maximum volume of the zinc chromate sealant cavity, post assembly.

(3) Minimum pressure required to effect zinc chromate sealant blow through.

(4) Bleedback capability of the primary seal (from sealant cavity to cavity between the primary and secondary seals) at a variety of pressure values ranging from 10 psig up to a value which has been determined to effect sealant blowthrough. Various types of primary seal leakage conditions at predetermined leakage rates should be simulated.

(5) Determine maximum acceptable leakage rates at 200 psig which meets the 50 psig leak test criteria.

(6) Determine minimum pressure and time required to position O-rings for 50 psig leak check.

(7) Determine the volume of the cavity between the primary and secondary O-rings by analysis and flow test prior to and following the 200 psi O-ring positioning cycle.

(8) KSC and MTI GSE volumes should be simulated and the required temperature range should be duplicated as close as possible. Type II zinc chromate sealant should be used for all tests.

PC 026431

25 Years
1958 1983

b. Nozzle Fixed Housing/Aft Segment Boss Joint - Perform tests with full scale hardware to accomplish the objectives in item a. above. The test designed to determine O-ring bleedback rate need not be repeated.

c. Igniter Adapter/Case Forward Segment Boss Joint - Perform tests with full scale hardware to accomplish the objectives in item a. above. Tests to determine pressure value required to position the seal is not required.

It is highly desirable to complete these tests prior to stacking of STS-12.

Questions concerning this memorandum should be referred to Mr. Leon Ray, 3-3809.

John Q. Miller
Chief, Solid Motor Branch

cc:
SA42/Mr. Wear
SA42/Mr. Denton
EE11/Mr. Coates
EP01/Mr. McCool
EP21/Mr. McCarty
EP25/Mr. Powers
EP25/Mr. Ray

PC 026432

V-E

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

85-18

EP01 (85-48) March 7, 1985

TO: SA41/Mr. Mulloy
THRU: EE11/Mr. Horton
FROM: EP01/Mr. McCool

SUBJECT: Request for Initiation of Testing to Provide Data for Resolving the Burned O-Ring Seal Problem on the Space Shuttle SRM

Letter EP25 (83-119), from Mr. Miller (EP25) to Mr. Horton (EE11), subject: "Request for Tests by the Contractor to Obtain Space Shuttle SRM Clevis Joint, Field Housing/Aft Segment Joint and Ignition Adapter/Forward Segment Joint Leak Check Data" is referenced.

On December 6, 1983, this office requested via the referenced letter that the contractor obtain available full scale diameter, short stack hardware and conduct tests to provide data on zinc chromate putty behavior as related to affect on joint leak checks. Fourteen months have elapsed and no visible action has been taken to obtain and equip the short stack hardware although agreement was made to perform the test at the time of request. The only positive response by the contractor was the submittal of TWR-14359 on May 4, 1983, which contained a program plan followed by 5-inch CP motor tests, which were not designed to provide a solution to the burned O-ring problem. The acquisition of joint putty layup and leak check data on a high priority basis has become very important in view of the need to resolve the burned O-ring problems; accordingly, it is requested that you take the necessary action to direct that the following tasks be expeditiously performed by the contractor:

a. Subscale and full scale tests to determine effects of asbestos filled, cotton and talc filled, and non-filled zinc chromate putty on O-ring sealing integrity.

b. Full scale tests:

(1) Putty layup tests using current layup design.

(2) Putty layup tests using the attached figure 1 layup concept.

(3) Putty layup tests using the attached figure 2 layup concept.

PC 026413

25

(4) Repeat tests (1) and (2) except with vent slots located at 120-degree interval around the circumference as shown by attached figures 3 and 4. The slots are designed to prevent air entrapment and resulting volcanoes. Evaluation of layup effectiveness should be performed with flow meters to determine cavity volumes.

The above tasks are intended to complement TWR-14359 rather than replace the tests defined therein. We will be happy to assist the contractor in working out the details for the above proposals.

A.A. McCool
Director
Structures and Propulsion Laboratory

Enclosures:
As stated

cc:
SA42/Messrs. McIntosh/Denton
EE11/Mr. Coates
EE11/Mr. Jones
EP21/Mr. McCarty
EP25/Messrs. Miller/Powers/Ray
EE01/Dr. Littles

PC 026414

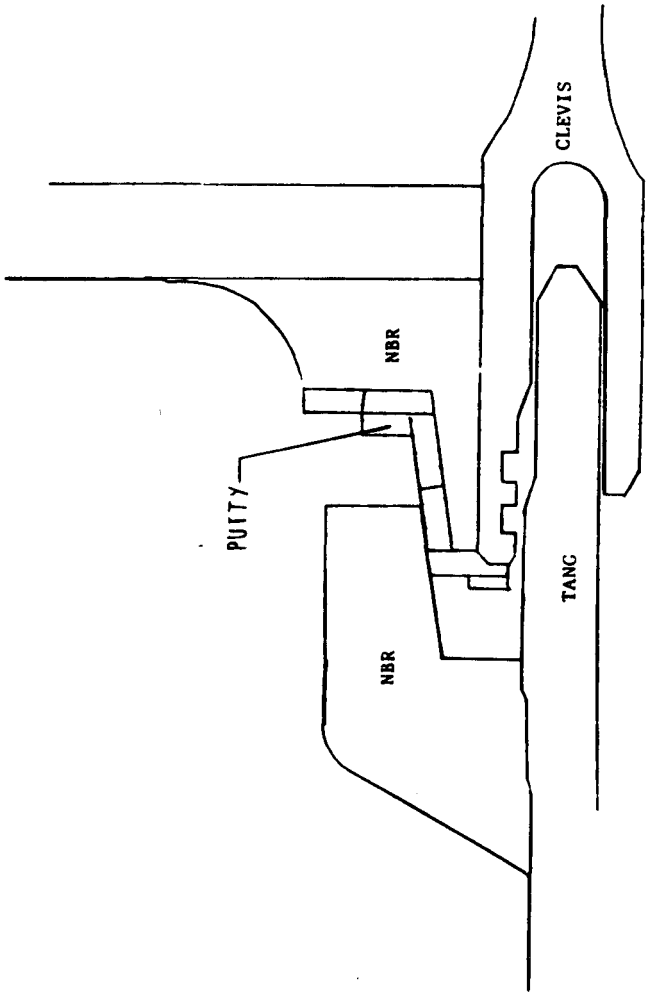


Figure 1

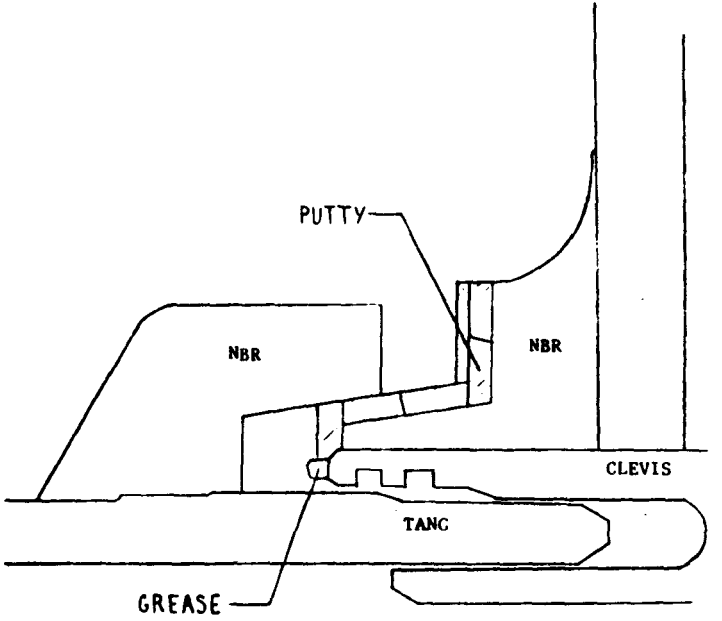
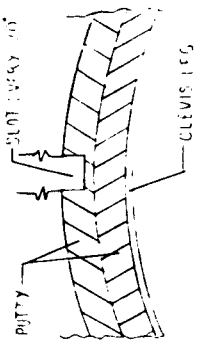


Figure 2

PC 026416

PC 026417



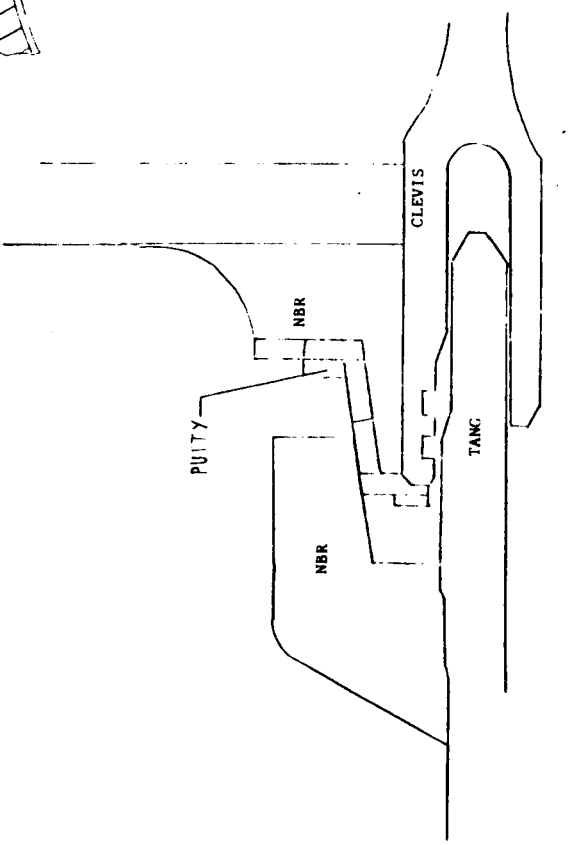
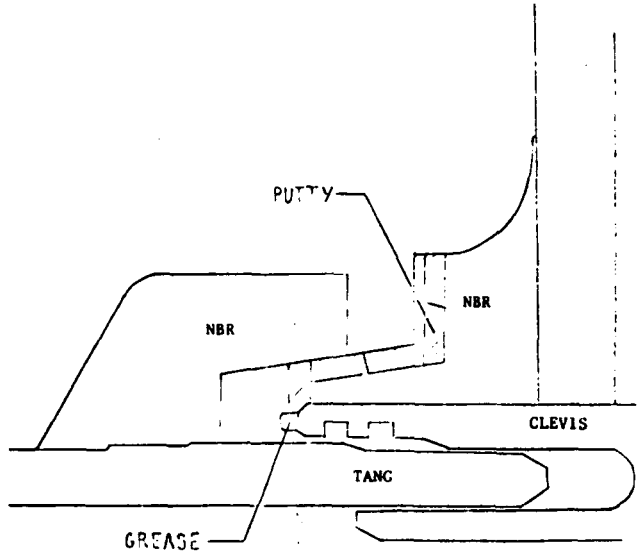


Figure 3



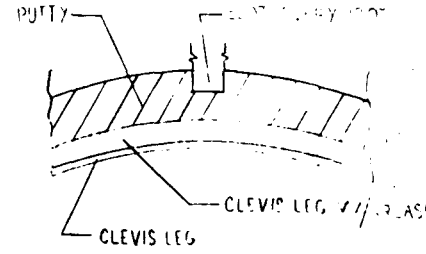


Figure 4

PC 026418

V-F

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

Reply to Attn of EP25 (84-15)

February 22, 1984

TO: EE11/Mr. Horton

FROM: EP25/Mr. Miller

SUBJECT: Inspection of Fired SRM Pressure Joint During Disassembly

Please take the necessary action to reinstate detail post flight and post static firing inspection of specific pressure joints on the SRM which incorporate the thermal barrier and O-ring seal design concept. The inspection must be conducted at the time of disassembly to preclude destruction of data. The task should be performed by experienced, qualified engineering personnel and should be continued until the burned O-ring problem is understood and resolved.

The incidence of heat damaged O-rings on STS-2, QM-4 and on the recent flight of STS-11 warrants close surveillance of these areas to ensure that suspected anomalies are detected and properly recorded for assessment purposes. Recent discovery that the new type II zinc chromate sealant (thermal barrier material) would not adhere to the nozzle surface to which it was being applied, has opened up several unanswered questions, the most important being adhesion life of the sealant after installation on the SRM. Type II zinc chromate sealant was installed on all SRM's beginning with STS-8.

Areas of concern which warrant inspection are:

a. SRM case field joints.

b. SRM case nozzle boss to nozzle fixed housing joint.

c. SRM igniter to SRM case igniter boss.

d. Nozzle field splice joint.

for John Q. Miller
Chief, Solid Motor Branch

PC 026428

cc:
EE01/Mr. Hardy
SA41/Mr. Mulloy
SA42/Mr. Wear
SA42/Mr. McIntosh
EP01/Mr. McCool
EP21/Mr. McCarty
EP25/Mr. Powers
EP25/Mr. Ray

PC 026429

V-G

22 July 1985

PROGRESS REPORT

APPLIED MECHANICS DEPARTMENT
Cost Center 287x

SPACE SHUTTLE

SRM O-Ring Erosion Problem

This problem has escalated so badly in the eyes of everyone, especially our customer, NASA, that NASA has gone to our competitors on a propriety basis and solicited their experiences on their joint configurations.

This whole week has been spent concepting ideas on how to eliminate the O-ring erosion problem. The new ground rule is to present every idea regardless of impact to cost weight schedule or whatever. Eleven hours of just group meetings has been spent discussing the problem and potential solutions. This does not include the many hours of informal meetings held with smaller groups.

One thing is increasingly obvious as time passes on this problem. If the company and/or Engineering does not assign specific people to this task (with no other work allowed - this being an absolute requirement) to secure a timely solution, then we stand in danger of having one of our competitors solve our problem via an unsolicited proposal. This thought is almost as horrifying as having a flight failure before a solution is implemented to prevent O-ring erosion.

Roger M Boisjoly
7/22/85

MORTON THIOKOL, INC.
Wasatch Division

Interoffice Memo

1 October 1985
E150/RVE-86-47

TO: A. J. McDonald, Director
Solid Rocket Motor Project

FROM: Manager, SRM Ignition System, Final Assembly, Special Projects and Ground Test

CC: B. McDougall, B. Russell, J. McCluskey, D. Cooper, J. Kilminster, B. Brinton, T. O'Grady, B. MacBeth, J. Sutton, J. Elwell, I. Adams, F. Call, J. Lamere, P. Ross, D. Fullmer, E. Bailey, D. Smith, L. Bailey, B. Kuchek, Q. Eskelsen, P. Petty, J. McCall

SUBJECT: Weekly Activity Report
1 October 1985

EXECUTIVE SUMMARY

HELP! The seal task force is constantly being delayed by every possible means. People are quoting policy and systems without work-around. MSFC is correct in stating that we do not know how to run a development program.

GROUND TEST

1. The two (2) GTM center segments were received at T-24 last week. Optical measurements are being taken. Significant work has to be done to clean up the joints. It should be noted that when necessary SICBM takes priority.

2. The DM-6 test report less composite section was released last week.

ELECTRICAL

As a result of the latest engineering analysis of the V-1 case it appears that high stress risers to the case are created by the phenolic OFI housings and fairings. As it presently stands, these will probably have to be modified or removed and if removed will have to be replaced. This could have an impact on the launch schedule.

TC 3018 (REV 7-84)

PC 102554

FINAL ASSEMBLY

One SRM 25 and two SRM 26 segments along with two SRM 24 exit cones were completed during this period. Only three segments are presently in work. Availability of igniter components, nozzles and systems tunnel tooling are the present constraining factors in the final assembly area.

IGNITION SYSTEM

1. Engineering is currently rewriting igniter gask-o-seal coating requirements to allow minor flaws and scratches. Bare metal areas will be coated with a thin film of HD-2 grease. Approval is expected within the week.

2. Safe and Arm Device component deliveries is beginning to cause concern. There are five S&A's at KSC on the shelf. Procurement, Program Office representatives visited Consolidated Controls to discuss accelerating scheduled deliveries. CCC has promised 10 A&M's and 30 B-B's no later than 31 October 1985.

O-RINGS AND PUTTY

1. The short stack finally went together after repeated attempts, but one of the o-rings was cut. Efforts to separate the joint were stopped because some do not think they will work. Engineering is designing tools to separate the pieces. The prints should be released tomorrow.

2. The inert segments are at T-24 and are undergoing inspection.

3. The hot flow test rig is in design, which is proving to be difficult. Engineering is planning release of these prints Wednesday or Thursday.

4. Various potential filler materials are on order such as carbon, graphite, quartz, and silica fiber braids; and different putties. They will all be tried in hot flow tests and full scale assembly tests.

5. The allegiance to the o-ring investigation task force is very limited to a group of engineers numbering 8-10. Our assigned people in manufacturing and quality have the desire, but are encumbered with other significant work. Others in manufacturing, quality, procurement who are not involved directly, but whose help we need, are generating plenty of resistance. We are creating more instructional paper than engineering data. We wish we could get action by verbal request but such is not the case. This is a red flag.

R. V. Ebeling

NASA-CR-175355

V-I

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

Reply to Attn of: EP25 (79-23)

February 6, 1979

TO: Distribution

FROM: EP25/Mr. Ray

SUBJECT: Visit to Precision Rubber Products Corporation and Parker Seal Company

The purpose of this memorandum is to document the results of a visit to Precision Rubber Products Corporation, Lebanon, TN, by Mr. Eudy, EE51 and Mr. Ray, EP25, on February 1, 1979 and also to inform you of the visit made to Parker Seal Company, Lexington, KY on February 2, 1979 by Mr. Ray. The purpose of the visits was to present the O-ring seal manufacturers with data concerning the large O-ring extrusion gaps being experienced on the Space Shuttle Solid Rocket Motor clevis joints and to seek opinions regarding potential risks involved.

The visit on February 1, 1979, to Precision Rubber Products Corporation by Mr. Eudy and Mr. Ray was very well received. Company officials, Mr. Howard Gillette, Vice President for Technical Direction, Mr. John Hoover, Vice President for Engineering, and Mr. Gene Hale, Design Engineer attended the meeting and were presented with the SRM clevis joint seal test data by Mr. Eudy and Mr. Ray. After considerable discussion, company representatives declined to make immediate recommendations because of the need for more time to study the data. They did; however, voice concern for the design, stating that the SRM O-ring extrusion gap was larger than that covered by their experience. They also stated that more tests should be performed with the present design. Mr. Hoover promised to contact MSFC for further discussions within a few days. Mr. Gillette provided Mr. Eudy and Mr. Ray with the names of two consultants who may be able to help. We are indebted to the Precision Rubber Products Corporation for the time and effort being expended by their people in support of this problem, especially since they have no connection with the project.

The visit to the Parker Seal Company on February 2, 1979, by Mr. Ray, EP25, was also well received; Parker Seal Company supplies the O-rings used in the SRM clevis joint design. Parker representatives, Mr. Bill Collins, Vice President for Sales, Mr. W. B. Green, Manager for Technical Services, Mr. J. W. Kosty, Chief Development Engineer for R&D, Mr. D. P. Thalman, Territory Manager and Mr. Dutch Haddock, Technical Services, met with Mr. Ray, EP25, and were provided with the identical

PC 026477

SRM clevis joint data as was presented to the Precision Rubber Products Company on February 1, 1979. Reaction to the data by Parker officials was essentially the same as that by Precision; the SRM O-ring extrusion gap is larger than they have previously experienced. They also expressed surprise that the seal had performed so well in the present application. Parker experts would make no official statements concerning reliability and potential risk factors associated with the present design; however, their first thought was that the O-ring was being asked to perform beyond its intended design and that a different type of seal should be considered. The need for additional testing of the present design was also discussed and it was agreed that tests which more closely simulate actual conditions should be done. Parker officials will study the data in more detail with other Company experts and contact MSFC for further discussions in approximately one week. Parker Seal has shown a serious interest in assisting MSFC with this problem and their efforts are very much appreciated.

William L. Ray

William L. Ray
Solid Motor Branch, EP25

Distribution:
SA41/Messrs. Hardy/Rice
EE51/Mr. Eudy
EP01/Mr. McCool

PC 026478

V-J

COMPANY PRIVATE

MORTON THIOKOL INC.
Wasatch Division

Interoffice Memo

31 July 1985
2870:FY86:073

TO: R. K. Lund
Vice President, Engineering

CC: B. C. Brinton, A. J. McDonald, L. H. Sayer, J. R. Kapp

FROM: R. M. Boisjoly
Applied Mechanics - Ext. 3525

SUBJECT: SRM O-Ring Erosion/Potential Failure Criticality

This letter is written to insure that management is fully aware of the seriousness of the current O-Ring erosion problem in the SRM joints from an engineering standpoint.

The mistakenly accepted position on the joint problem was to fly without fear of failure and to run a series of design evaluations which would ultimately lead to a solution or at least a significant reduction of the erosion problem. This position is now drastically changed as a result of the SRM 16A nozzle joint erosion which eroded a secondary O-Ring with the primary O-Ring never sealing.

If the same scenario should occur in a field joint (and it could), then it is a jump ball as to the success or failure of the joint because the secondary O-Ring cannot respond to the clevis opening rate and may not be capable of pressurization. The result would be a catastrophe of the highest order - loss of human life.

An unofficial team (a memo defining the team and its purpose was never published) with leader was formed on 19 July 1985 and was tasked with solving the problem for both the short and long term. This unofficial team is essentially nonexistent at this time. In my opinion, the team must be officially given the responsibility and the authority to execute the work that needs to be done on a non-interference basis (full time assignment until completed).

PC 122534

COMPANY PRIVATE

It is my honest and very real fear that if we do not take immediate action to dedicate a team to solve the problem, with the field joint having the number one priority, then we stand in jeopardy of losing a flight along with all the launch pad facilities.

Roger M Boisjoly

R. M. Boisjoly

Concurred by:

Jack R Kapp

J. R. Kapp, Manager
Applied Mechanics

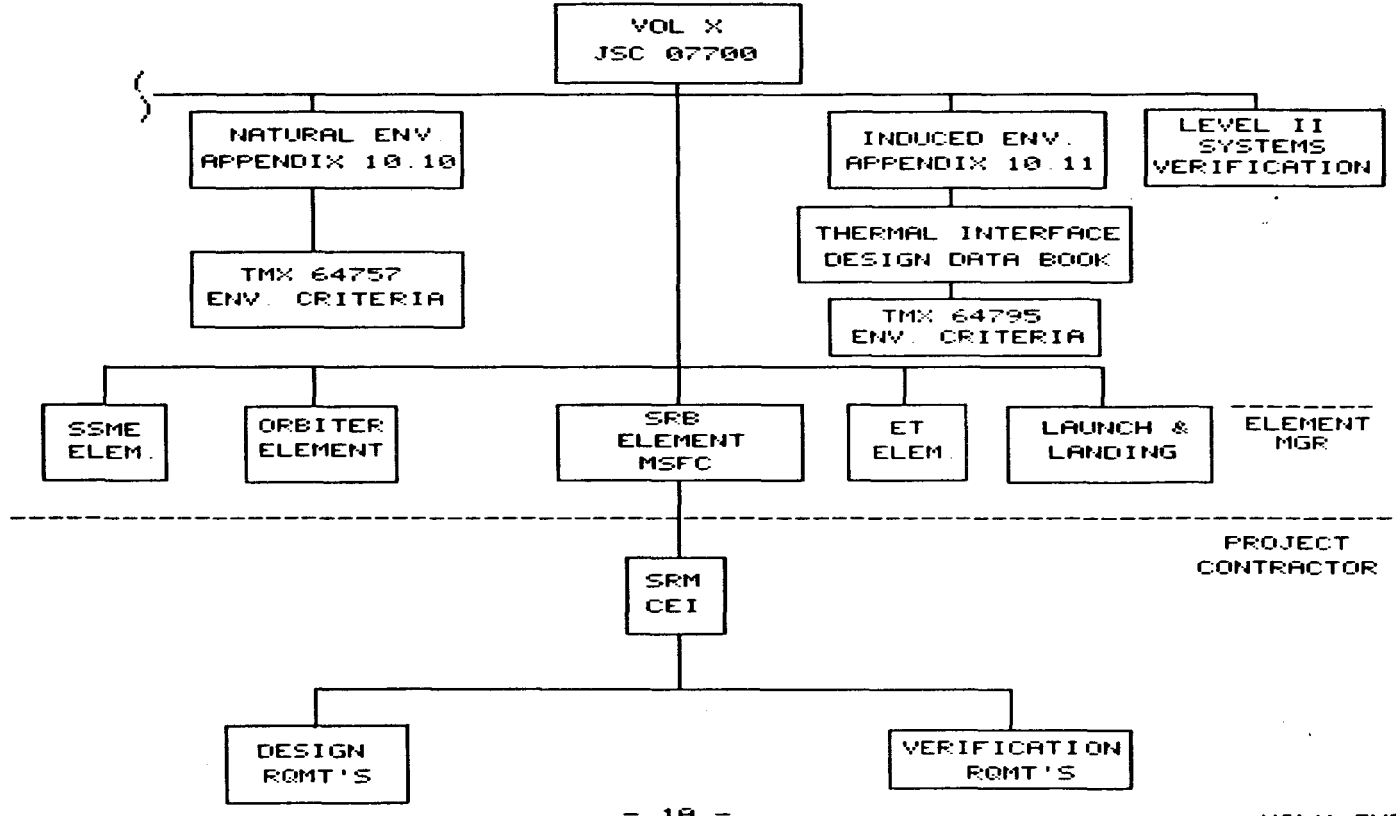
COMPANY PRIVATE

PC 102537

NATURAL/INDUCED ENVIRONMENT

REQUIREMENTS

R. H. KOHRS
MAY 19, 1986


VOL X
JSC 07700
NATURAL ENV.
APPENDIX 10.10
TMX 64757
ENV. CRITERIA
INDUCED ENV.
APPENDIX 10.11
THERMAL INTERFACE
DESIGN DATA BOOK
TMX 64795
ENV. CRITERIA
LEVEL II
SYSTEMS
VERIFICATION
SSME
ELEM.
ORBITER
ELEMENT
SRB
ELEMENT
MSFC
ET
ELEM.
LAUNCH &
LANDING
ELEMENT
MGR
PROJECT
CONTRACTOR
SRM
CEI
DESIGN
RQMT'S
VERIFICATION
RQMT'S

VOLX.SKE

JSC 07700
VOLUME X
REVISION D

National Aeronautics and
Space Administration

Lyndon B. Johnson Space Center
Houston, Texas 77058

NATIONAL
SPACE TRANSPORTATION SYSTEMS
PROGRAM

SPACE SHUTTLE FLIGHT AND GROUND SYSTEM SPECIFICATION

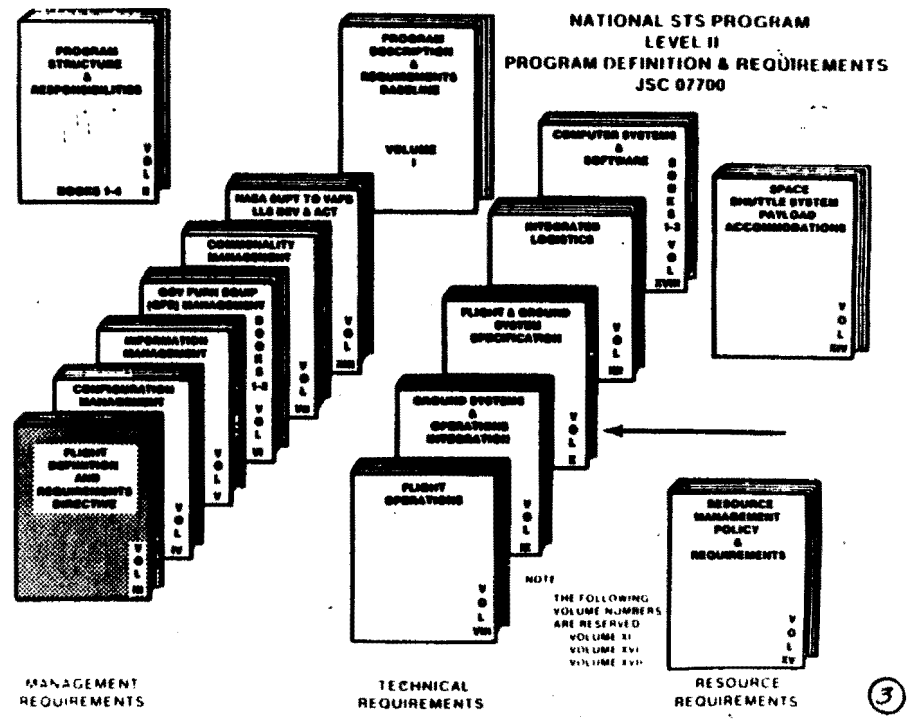
LEVEL II PROGRAM DEFINITION
AND REQUIREMENTS

SEPTEMBER 30, 1983

(2)

NASA National Aeronautics and Space Administration

Lyndon B. Johnson Space Center
Houston, Texas 77058



shall not exceed 0.5 psig maximum at a boil-off rate of 2.9 lb/sec. The H_2 vent system shall not interface with the Orbiter but shall vent directly to atmosphere in flight. In addition to providing RT relief protection, the vent valves shall be capable of being actuated open, prior to launch, by ground command. The electrical command and pneumatic supply will be provided by GSE. Capability shall be provided to monitor the main propulsion LH_2 system pressure when vehicle or ground power is not applied to the flight instruments.

3.2.2.1.16 LO$_2$ Compatibility. Any material used internally in the liquid oxygen system of the Space Shuttle System main propulsion subsystems shall be compatible as determined by NHB 8060.1.

3.2.2.1.17 Design Environments.

3.2.2.1.17.1 Natural Environment. The Shuttle Flight Vehicle design shall satisfy the natural environment design requirements specified in Appendix 10.10.

3.2.2.1.17.2 Induced Environment. Each element of the Shuttle Flight Vehicle shall be capable of withstanding the induced environments imposed during transportation, ground operations, handling and flight operations as defined in Appendix 10.11. Each interface between elements shall be designed to withstand the induced environments defined in the applicable ICD.

3.2.2.1.17.2.1 Ascent Heating Design Criteria. In general, all elements of the Space Shuttle System shall be designed to withstand limiting induced ascent aerodynamic and plume heating environments, encompassing all baseline reference missions. The Orbiter vehicle for which limit ascent aerodynamic heating environments coupled with reuse criteria would result in unnecessary weight and cost penalties, shall be designed to meet reuse requirements considering the frequency of occurance of the ascent heating environments resulting from statistical treatment of the baseline reference missions and shall be shown to have single-mission survivability for limit ascent aerodynamic heating case encountered on any mission during the lifetime of the vehicle. The applicable environments are defined in Appendix 10.11.

3.2.2.1.17.3 TPS Absorption. All TPS material and installation design shall minimize absorption and entrapment of liquids or gases which would degrade thermal or physical performance or present a fire hazard (wicking), and shall not require draining, drying or any dedicated purge system from refurbishment through launch.

(83) 3.2.2.1.18 Flow Induced Vibration. All flexible hoses and bellows shall be designed to exclude or minimize flow induced

(83) Refer to the Deviation/Waiver Page in front of the document.

(4)

8.0 METEOROID. The Space Shuttle shall be designed [illegible] a 0.95 probabilty of no penetration during the maximum total time for 500 missions in orbit, using the meteoroid model defined in Section 2.5.1 of TMI-64627.

8.1 METEOROID IMPACT. Space Shuttle meteoroid impact requirements shall be specified below:

a. Pressure Loss. The Space Shuttle manned volume shall be protected from meteoroid impact damage which would result in pressure loss when subjected to the meteoroid flux model as defined in TMI-64627.

b. Functional Capability. The Space Shuttle shall provide protection against loss of functional capability of selected critical items when subjected to the meteoroid flux model as defined in TMI-64627. The probability of no penetration shall be assessed on each item dependent upon function criticality.

9.0 AERODYNAMIC CONSTANTS. The values given in Sections 1.6 and 2.7 of TMI-64627 shall be used.

10.0 THERMAL.

10.1 GROUND THERMAL ENVIRONMENT. The ground thermal environment, including air temperature, solar radiation, and sky temperature limits, are specified in Table 10.1-1 and Figure 10-1. Also, see Sections 2.5 and 2.6 of TMI-64757.

10.1.1 Ambient Temperature for SRB Propellant Temperature Predictions. The appropriate KSC and VAFB monthly mean and extreme ambient temperatures listed below shall be used to establish SRB propellant temperatures and thrust performance. The low and high extremes are the 2.5 and 97.5 percentile average monthly temperatures.

(5)

NASA TECHNICAL MEMORANDUM

NASA TM X-64757

TERRESTRIAL ENVIRONMENT (CLIMATIC) CRITERIA GUIDELINES FOR USE IN AEROSPACE VEHICLE DEVELOPMENT, 1973 REVISION

Glenn E. Daniels, Editor
Aero-Astrodynamics Laboratory

July 5, 1973

NASA

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

⑥

MSFC - Form 3190 (Rev June 1971)

Table 10.1-1

Ground Thermal Environment

Thermal Environment Factor	Ferry Sites	Vertical Flight
Air Temperature (Degrees F)		
Design High	103	99
Low	20	31
Solar Radiation (Btu/ft²-hr)		
Design High	See Figure 10-1	
Low (Diffuse)		
Sky Temperature (Degrees F)		
Design High	50	
Low	-22	5

Local Standard Time - Hour

Time of Day	Design High Solar Radiation		Design Low Solar Radiation	
Hour	BTU/ft²/hr	gm-cal/cm²/min	BTU/ft²/hr	gm-cal/cm²/min
0500	0	0.00	0	0.00
1100	363	1.64	70	0.32
1300			80	0.36
1400	363	1.64		
2000	0	0.00	0	0.00

⑦

2. 5. 3. 4 Solar Radiation during Extreme Conditions

When ground winds occur exceeding the 95, 99, or 99. 9 percentile design winds given in this document in Section V, the associated weather normally is such that clouds, rain, or dust are generally present; therefore, the intensity of the incoming solar radiation will be less than the maximum values given in Tables 2. 3 and 2. 4. Maximum values of solar radiation intensity to use with corresponding wind speeds are given in Table 2. 5.

TABLE 2. 5 SOLAR RADIATION MAXIMUM VALUES ASSOCIATED WITH EXTREME WIND VALUES

Steady-State Ground Wind Speed at 10 m Height	Maximum Solar Radiation (Normal Incident) — Huntsville, New Orleans River Transportation, Gulf Transportation, Eastern Test Range, Western Test Range, Sacramento, West Coast Transportation and Wallops Test Range			White Sands Missile Range		
($m\ sec^{-1}$)	($kJm^{-2}\ sec^{-1}$)	($g\text{-}cal\ cm^{-2}\ min^{-1}$)	($BTU\ ft^{-2}\ hr^{-1}$)	($kJm^{-2}\ sec^{-1}$)	($g\text{-}cal\ cm^{-2}\ min^{-1}$)	($BTU\ ft^{-2}\ hr^{-1}$)
10	0.84	1.20	266	1.05	1.50	332
15	0.56	0.80	177	0.70	1.00	221
≥20	0.35	0.50	111	0.56	0.80	177

2. 6 Temperature

Several types of temperatures at the earth's boundary layer may be considered in design. These are as follows:

a. Air temperature normally measured at 1. 22 meters (4 ft) above a grass surface.

b. Changes of air temperature (Usually the rapid changes which occur in less than 24 hours are considered.)

c. Surface or skin temperature measured of a surface exposed to radiation.

d. Temperatures within a closed compartment.

All of the above will be discussed in the following subsections.

(F)

2.6.1 Air Temperature Near the Surface

Surface air temperature extremes (maximum, minimum, and the 95 percentile values) and the extreme minimum sky radiation (equal to the outgoing radiation) are given in Table 2.6 for various geographical areas. Maximum and minimum temperature values should be expected to last only a few hours during a daily period. Generally, the maximum temperature is reached after 12 noon and before 5 p.m., while the minimum temperature is reached just before sunrise. Table 2.7A shows the maximum and minimum air temperatures which have occurred on each hour at Kennedy Space Center, but not necessarily on the same day, although these curves represent a cold and hot extreme day. The method of sampling the day (frequency of occurrence of observations) will result in the same extreme values if the same period of time for the data is used, but the 95 percentile values will be different for hourly, daily, and monthly data reference periods. Selection of the reference period depends on engineering application. Table 2.7B gives month mean temperatures, standard deviations and 2.5 and 97.5 percentiles of values of temperature for Kennedy Space Center, Florida and Vandenberg AFB, California.

2.6.2 Extreme Air Temperature Change

a. For all areas the design values of extreme air temperature changes (thermal shock) are:

(1) An increase of air temperature of 10°C (18°F) with a simultaneous increase of solar radiation (measured on a normal surface) from 0.50 g-cal cm^{-2} min^{-1} (110 BTU ft^{-2} hr^{-1}) to 1.85 g-cal cm^{-2} min^{-1} (410 BTU ft^{-2} hr^{-1}) may occur in a 1-hour period. Likewise, the reverse change of the same magnitude may occur for decreasing air temperature and solar radiation.

(2) A 24-hour change may occur with an increase of 27.7° C (50°F) in air temperature in a 8-hour period, followed by 4 hours of constant air temperature, then a decrease of 27.7°C (50°F) in a 8-hour period, followed by 10 hours of constant air temperature.

b. For Eastern Test Range (Kennedy Space Center), the 99.9 percentile air temperature changes are as follows:

(1) An increase of air temperature of 5.6°C (11°F) with a simultaneous increase of solar radiation (measured on a normal surface) from 0.50 g-cal cm^{-2} min^{-1} (110 BTU ft^{-2} hr^{-1}) to 1.60 g-cal cm^{-2} min^{-1} (354 BTU ft^{-2} hr^{-1}), or a decrease of air temperature of 9.4°C (17°F) with a simultaneous decrease of solar radiation from 1.60 g-cal cm^{-2} min^{-1} (354 BTU ft^{-2} hr^{-1}) to 0.50 g-cal cm^{-2} min^{-1} (110 BTU ft^{-2} hr^{-1}) may occur in a 1-hour period. (9)

2.20

TABLE 2.7 MAXIMUM AND MINIMUM SURFACE AIR TEMPERATURES AT EACH HOUR FOR EASTERN TEST RANGE[4]

Time	Annual Maximum		Annual Minimum	
	°C	°F	°C	°F
1 a.m.	28.9	84	1.1	34
2	28.9	84	0.6	33
3	29.4	85	-1.1	30
4	28.3	83	-0.6	29
5	28.3	83	-1.1	28
6	29.4	85	-1.1	27
7	30.6	87	-1.7	26
8	30.6	87	-2.2	25
9	31.7	89	-0.6	28
10	33.9	93	1.1	30
11	35.0	95	2.2	35
12 noon	35.6	96	5.0	41
1 p.m.	37.2	99	5.6	42
2	35.6	97	5.0	41
3	35.6	97	5.6	42
4	35.6	97	5.6	42
5	35.6	97	5.6	42
6	35.0	95	3.9	39
7	33.3	92	2.2	36
8	31.7	89	2.2	36
9	30.0	86	1.7	35
10	30.0	86	1.7	35
11	30.0	86	1.1	34
12 mid	30.0	86	1.1	34

4. Based on 10 years of record for Patrick Air Force Base and Kennedy Space Center. (16)

2.0 APPLICABLE DOCUMENTS. The below listed documents form a part of this appendix to the extent specified herein. These documents shall be individually approved as baseline requirements. The "Current Issue" of each document may be determined from JSC 08102, Space Shuttle Program Level II Baseline Description and Status Report.

Contractor Handbooks

SD73-SH-0069-1 (Current Issue)	Structural Design Loads Data Book, Baseline Vehicle and Missions Ref. Para. 3.3, 3.3.1, Table 10.11.1
SD73-SH-0069-2 (Current Issue)	Orbiter Structural Design Loads Data Book Ref. Para. 3.3, 3.3.2, Table 10.11.1
SD73-SH-0069-3 (Current Issue)	Structural Design Loads Data Book, External Tank Ref. Para. 3.3, 3.3.3, Table 10.11.1
SD73-SH-0069-4 (Current Issue)	Structural Design Loads Data Book, Solid Rocket Boosters Structural Loads Ref. Para. 3.3, 3.3.4, Table 10.11.1
SD73-SH-0181-1 (Current Issue)	Aerodynamic Heating Data Book, Orbiter - Ascent (Books I, II and III) Ref. Para. 3.1, 3.1.1, Table 10.11.1
SD73-SH-0181-2 (Current Issue)	Aerodynamic Heating Data Book External Tank - Ascent (Books I and II) Ref. Para. 3.1, 3.1.1, Table 10.11.1
SD73-SH-0181-3 (Current Issue)	Aerodynamic Heating Data Book, Shuttle Vehicle Booster - Ascent (Books I and II) Ref. Para. 3.1, 3.1.1, Table 10.11.1

//

SD73-SH-0181-4 (Current Issue) — Aerodynamic Heating Data Book, Space Shuttle Main Engine

Ref. Para. 3.1, 3.1.1, Table 10.11.1

SD73-SH-0181-5 (Current Issue) — Aerodynamic Heating Data Book, Shuttle Launch Facility

Ref. Para. 3.1, 3.1.1, Table 10.11.1

SD73-SH-0181-7 (Current Issue) — Aerodynamic Heating Data Book, Lightweight External Tank-Ascent

Ref. Para. 3.1

SD74-SH-0082 (Current Issue) — Acoustics and Shock Data Book, Space Shuttle System

Ref. Para. 3.4, 3.5, Table 10.11.1

SD74-SH-0144 (Current Issue) — Thermal Interfaces Design Data Book ←

Ref. Para. 3.2, Table 10.11.1

(12)

SD 74-SH-0144D

SPACE SHUTTLE PROGRAM
THERMAL INTERFACES DESIGN DATA BOOK

DECEMBER 1977 SDM BASELINE

Contract NAS9-14000
IRD No. SE-699T2
WBS 2.2.1

Prepared by

W.G. Casey, Supervisor
RCC and Penetrations Analysis

C.A. Laren
RCC and Penetrations Analysis

Approved by

P.C. Marhoff, Manager
Thermal Analysis

N.F. Witte, Director
Shuttle Aero Sciences

Rockwell International
Space Division

(13)

1.0 INTRODUCTION

1.1 SCOPE

Contained herein are the thermal design data required to complete the thermal interface definitions for the following interface control documents (ICD's):

ICD 2-12001	Orbiter Vehicle/External Tank
ICD 2-14001	Orbiter Vehicle/Solid Rocket Booster
ICD 2-24001	External Tank/Solid Rocket Booster
ICD 13M-15000	Space Shuttle Orbiter Vehicle/Main Engine

1.2 APPLICABLE DOCUMENTS

The natural environments that have an effect on the thermal design are defined in the following documents:

JSC 07700, Space Shuttle Flight and Ground Systems Specification, Volume X, Appendix 10.10, Natural Environment Design Requirements.

NASA TM X-64795, Distribution of Eight Meteorological Variables at Cape Kennedy, Florida, and Vandenberg Air Force Base, California; Marshall Space Flight Center (dated November 19, 1973).

The induced environments applicable to the Shuttle system elements performing as part of the integrated flight vehicle are defined in the following data books:

SD 73-SH-0181-1A, Space Shuttle Aerodynamic Heating Data Book - Orbiter Ascent, Volume I (dated February 1975)

SD 73-SH-0181-2, Space Shuttle Aerodynamic Heating Data Book - External Tank Ascent, Volume II (dated June 1976).

SD 73-SH-0183-3, Space Shuttle Aerodynamic Heating Data Book - Shuttle Vehicle Booster Ascent, Volume III (dated September 1976).

SD 73-SH-0181-4, Space Shuttle Aerodynamic Heating Data Book - Space Shuttle Main Engine Ascent, On-Orbit and Entry, Volume IV (dated September 1977)

These books form part of the Space Shuttle Flight and Ground Systems Specification, JSC 07700, Volume X, Appendix 10.11, Induced Environment Design Requirements. The induced environments data are available on magnetic tape records as specified in Reference 1 for the orbiter, Reference 2 for the external tank, and Reference 3 for the Shuttle vehicle booster.

(14)

2.4 SURFACE TEMPERATURE RESPONSES

Surface temperature responses are provided to establish the radiant sources and sinks for the Shuttle vehicle from prelaunch through ascent. These data are not to be construed as material temperature limits. They are intended as thermal interface data required for element thermal design analysis.

2.4.1 Prelaunch

The prelaunch, post-fill surface temperature histories are presented herein for the hot and cold day environments specified in the Space Shuttle Flight and Ground Systems Specification, JSC 07700, Volume X, Appendix 10.10, Natural Environment Design Requirements. The 95th percentile day for July at Cape Kennedy was used for the extreme hot day environment. For the extreme cold day environment, the 5th percentile day for January at Vandenberg Air Force Base was used. External tank fill was assumed at 1100 hours for all cases.

2.4.1.1 Orbiter Vehicle

The prelaunch environmental temperatures for the fuselage lower external surfaces (Zones 01 through 05, Figure 2.1-1) are presented in Figures 2.4.1.1-1 through 2.4.1.1-4. For the wing leading edge (Zones 06 through 09, Figure 2.1-1), the external surface temperature histories for prelaunch are given in Figures 2.4.1.1-5 and 2.4.1.1-6. The temperature variations for the wing lower external surfaces (Zones 010 through 012, Figure 2.1-1) during prelaunch are provided in Figures 2.4.1.1-7 through 2.4.1.1-9.

2.4.1.2 External Tank (ET)

Figures 2.4.1.2-1 through 2.4.1.2-3 provide the temperature response of the external tank surfaces defined in Figure 2.1-2 when subjected to a hot day prelaunch environment. The cold day prelaunch environment temperature variations for the external tank surfaces (Figure 2.1-2) are presented in Figures 2.4.1.2-4 through 2.4.1.2-6. The temperatures to be used for the ET crossbeam surfaces (Figure 2.1-3) are 98°F for hot day and 29°F for cold day prelaunch environments.

2.4.1.3 Solid Rocket Booster (SRB)

Temperature profiles for the SRB external surfaces defined in Figure 2.1-4 are provided in Figure 2.4.1.3-1 for the hot day prelaunch environment and Figure 2.4.1.3-2 for the cold day prelaunch environment.

2.4.1.4 Space Shuttle Main Engine (SSME)

Hot day prelaunch environmental temperature histories for the Space Shuttle main engine interface envelope, defined in Figure 2.4.1.4-3, are presented in Figure 2.4.1.4-1. The SSME compartment surface temperatures are given in Figure 2.4.1.4-2 for a cold day prelaunch environment.

NASA TECHNICAL
MEMORANDUM

NASA TM X-64795

DISTRIBUTIONS OF EIGHT METEOROLOGICAL VARIABLES AT CAPE KENNEDY, FLORIDA AND VANDENBERG AIR FORCE BASE, CALIFORNIA

By M. E. Graves, R. L. King, and S. Clark Brown
Aero-Astrodynamics Laboratory

November 19, 1973

NASA

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

MSFC - Form 3190 (Rev June 1971)

Rockwell International
Space Division

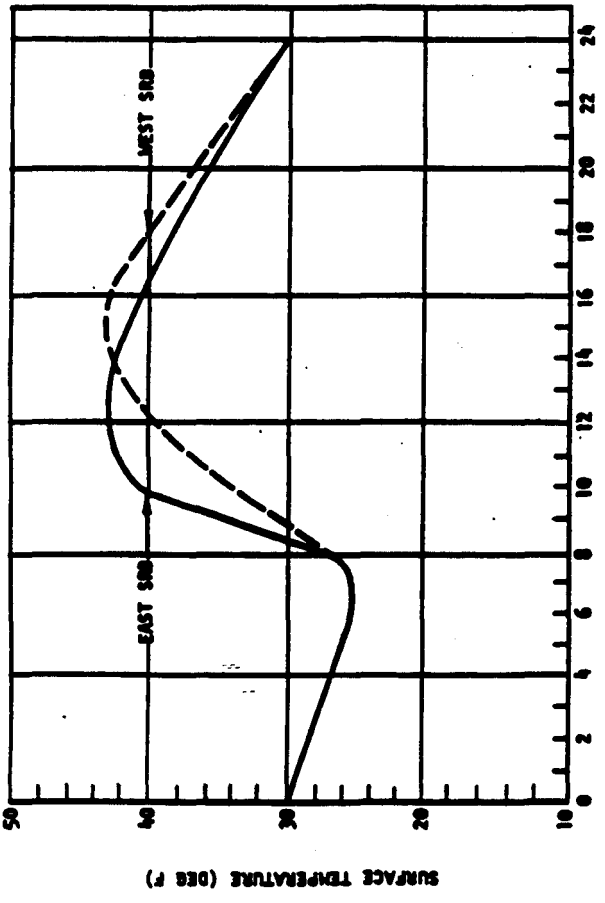


Figure 2.4.1.3-2. Solid Rocket Booster Surface Temperature, Zones B1 Through B10 - Prelaunch Environment, Cold Day

*ET FILL COMPLETE AT 1100 HRS

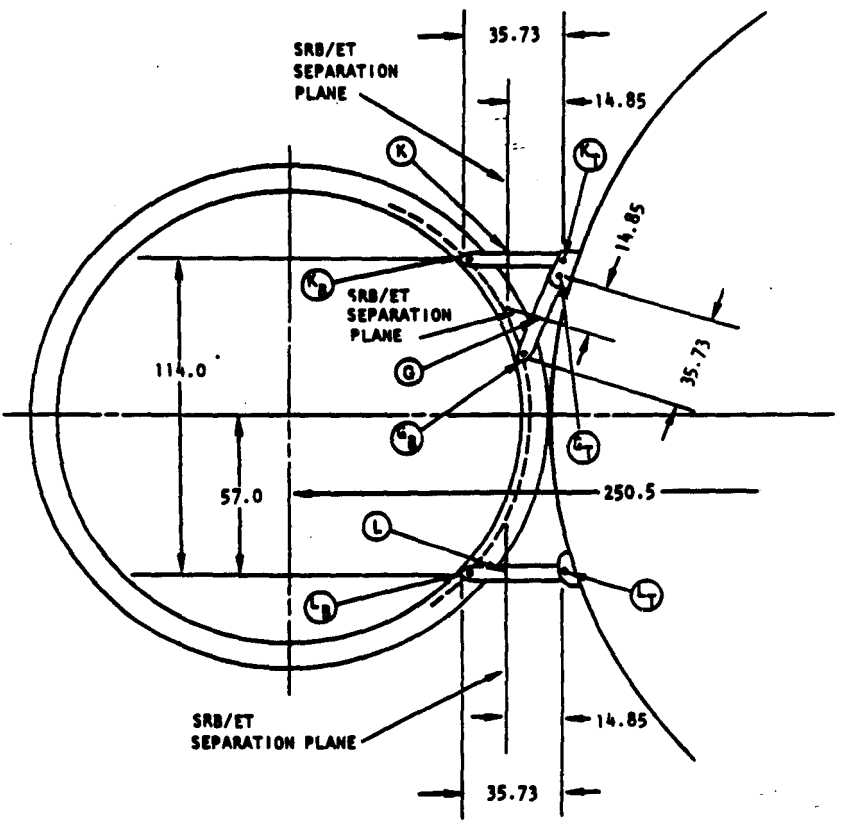


Figure 3.1.2-4. ET/SRB Aft Attachment

3.2.3 Solid Rocket Booster

Temperature extremes for the SRB side of the following conduction interfaces:

ET/SRB forward attachment (Figure 3.1.2-3)

ET/SRB aft attachment (Figure 3.1.2-4)

are presented in Table 3.2.3-1 for hot day (maximum) and cold day (minimum).

Table 3.2.3-1. Solid Rocket Booster/External Tank Conduction Interface Temperatures—Prelaunch Environment

Location	Temperature (deg F)	
	Hot Day (maximum)	Cold Day (minimum)
Forward attachment, Location F_B	89	39
Forward attachment, Location F	90	26
Aft attachment, Locations G_B, K_B, and L_B	96	21

3.2.4 Space Shuttle Main Engine

Temperature histories for the SSME side of the following interfaces:

LPFTP flange (Figure 3.1.1-3)

LPOTP flange (Figure 3.1.1-4)

Gimbal bearing (Figure 3.1.1-5)

Heat shield attachment (Figure 3.1.1-6)

TVC actuator attachment (Figure 3.1.1-7)

are presented in Figures 3.2.4-1 through 3.2.4-5.

3.2.5 Mobile Launch Platform

Maximum temperatures for the MLP side of the following condition interfaces:

SRB/MLP hold-down support hardware (Figure 3.1.3-1)

are presented in Table 3.2.5-1

(19)

Specification No. CPW1-3300

PRIME EQUIPMENT CONTRACT END ITEM
DETAIL SPECIFICATION
PART I OF TWO PARTS

PERFORMANCE, DESIGN AND VERIFICATION REQUIREMENTS SPACE SHUTTLE HIGH PERFORMANCE, SOLID ROCKET MOTOR LIGHTWEIGHT CPW1-3300 FOR SPACE SHUTTLE SOLID ROCKET MOTOR PROJECT OPERATIONAL FLIGHT (STS-8, 12 & Subs)

17 February 1984

PENDING NASA APPROVAL

DR NO. 2-2

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA 35812

BY

Morton Thiokol, Inc.
Wasatch Division
P.O. Box 524, Brigham City, Utah 84302 801/863-3511

maintenance and refurbishment operations.

3.2.6.3 Personnel Safety. Provisions for personnel safety shall be in accordance with the following:

a. Safety Devices. Known hazards which cannot be eliminated through design selection shall be reduced to an acceptable level through the use of appropriate safety devices as part of the system, subsystem, or equipment.

b. Warning Devices. Where it is not possible to preclude the existence or occurrence of a known hazard, devices shall be employed for the timely detection of the condition and the generation of an adequate warning signal. Warning signals and their application shall be designed to minimize the probability of wrong signals or of improper personnel reaction to the signal.

3.2.6.4 Explosive and/or Ordnance Safety. The propellants for the HPML and the igniter shall meet the requirements of hazard classification 2 as defined in the Army Material Command Regulation Safety Manual AMCR 385-100, or DoD Contractor's Safety Manual for Ammunition, Explosives, and Related Dangerous Materials, DoD 4145.26. The HPML segments and ignition system less initiators shall have a DoT explosive classification of Class B.

3.2.7 Environment.

3.2.7.1 Natural Environment. The HPML shall withstand the natural environments defined in JSC 07700, Volume X, Appendix 10.10 and the air and sea temperature environments and salinity of SE-019-043-2H. 31°

3.2.7.2 Induced Environment. The HPML shall withstand the induced environmental conditions as defined in the following documents:

Thermal
Base Heating - SD73-SH-0181-3
Launch & Ascent - SD73-SH-0181-3
Re-entry - SE-019-053-2H
Interface - SD74-SH-0144, ICD 3-44003 25° + 21°

Loads
Vibration, Acoustic & Shock SE-019-049-2H and SE-019-067-2H (as changed by approved Deviation RDW-0012R4) In case of conflict, SE-019-049-2H shall take precedence over SE-019-067-2H.
Prelaunch through Separation - SE-019-057-2H, Book 1 (21)

TABLE V

VERIFICATION CROSS REFERENCE INDEX	CPW1-3300 Dated 17 February 1984 Page I-92

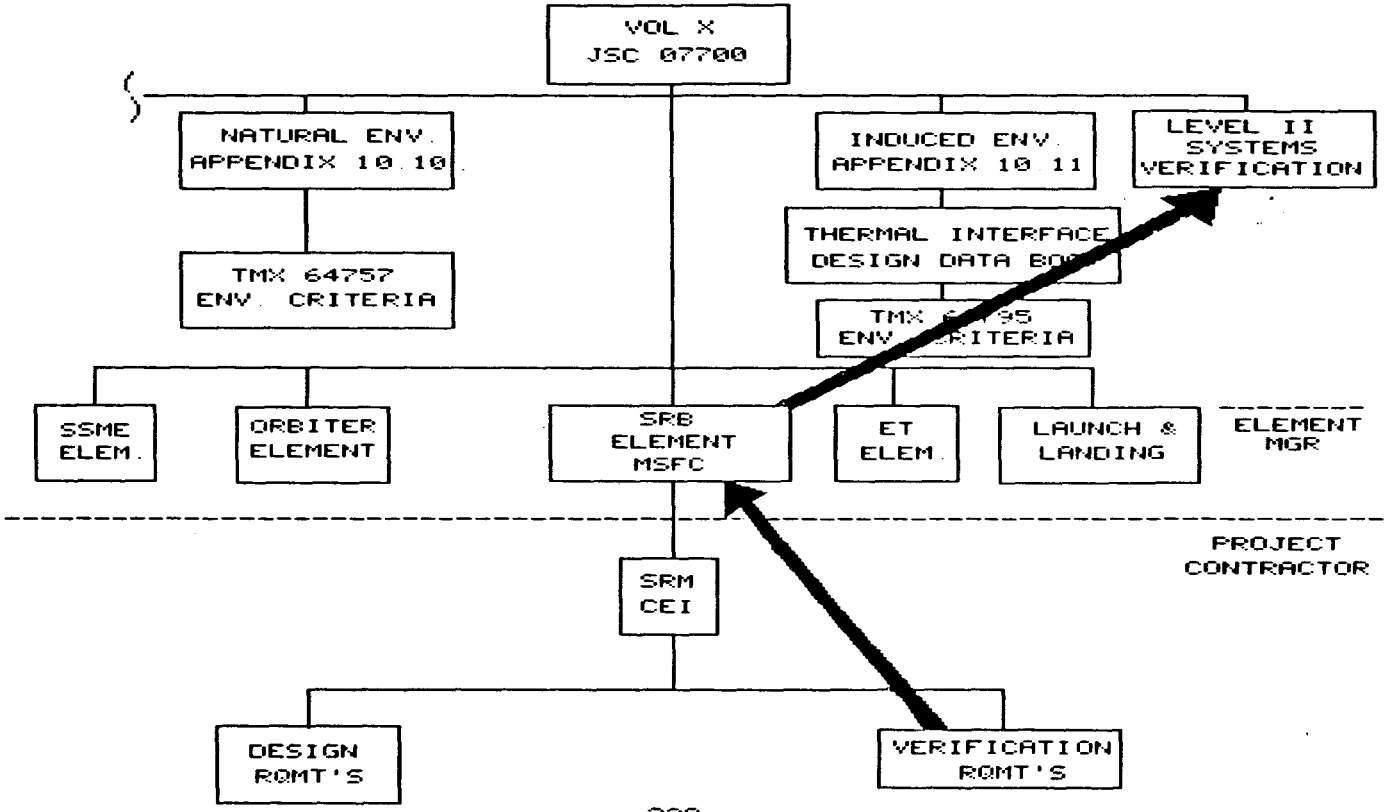
REQUIREMENTS FOR VERIFICATION

VERIFICATION METHOD:
1. Similarity
2. Analysis
3. Inspection
4. Demonstration
5. Test

N/A - Not Applicable

VERIFICATION PHASE:
A. Development
B. Qualification
C. Acceptance
D. Preflight
E. Flight
F. Postflight

Section 3.0 Performance/ Design Requirement Reference	Verification Methods							Section 4.0 Verification Requirement Reference
	N/A	A	B	C	D	E	F	
3.2.6.3 Personnel Safety		2	2					4.2.1.2, 4.2.2.2
3.2.6.4 Explosive and/or Ordnance Safety		5	5					4.2.1.5, 4.2.2.5
3.2.7 Environment	X							
3.2.7.1 Natural Environment		2	2					4.2.1.2, 4.2.2.2,
3.2.7.2 Induced Environment		2	2					4.2.1.2, 4.2.2.2
3.2.8 Transportability/ Transportation		1	1		4		4	4.2.1.1, 4.2.2.1, 4.2.5.4, 4.2.7
3.2.9 Storage	X							
3.2.9.1 Post Acceptance Requirements				2				4.2.3.2
3.2.9.2 Storage/ Age Control		3	3		3			4.2.1.3, 4.2.2.3, 4.2.5.3


VOL X
JSC 07700
NATURAL ENV.
APPENDIX 10.10
TMX 64757
ENV. CRITERIA
INDUCED ENV.
APPENDIX 10 11
LEVEL II
SYSTEMS
VERIFICATION
THERMAL INTERFACE
DESIGN DATA BO
TMX
ENV
SSME
ELEM.
ORBITER
ELEMENT
SRB
ELEMENT
MSFC
ET
ELEM.
LAUNCH &
LANDING
ELEMENT
MGR
PROJECT
CONTRACTOR
SRM
CEI
DESIGN
RQMT'S
VERIFICATION
RQMT'S

VOLX1.SKE

SPECIFICATION VERIFICATION

- PROJECTS REQUIRED TO SHOW THAT EACH REQUIREMENT IN THE CONTRACT SPECIFICATION HAS BEEN VERIFIED
- ELEMENT MANAGERS REQUIRED TO VERIFY COMPLIANCE WITH LEVEL II REQUIREMENTS THRU VERIFICATION COMPLETION NOTICES

STS-1

Rockwell International
Space Systems Group

SPACE SHUTTLE SYSTEM
INTEGRATED ELEMENTS

NASA

VERIFICATION COMPLETION NOTICE
(FRF/FMOF)

IVLN NO.: 12 PAGE 1 OF 3

IVLN TITLE: BASELINE MISSION CAPABILITY VCN NO.: 12A11

DATE:

CONSTRAINT: FMOF

SCOPE OF IVLN

The Baseline Mission Capability IVLN, as applicable to this VCN, verifies those activities and interrelationships that apply to FMOF constraints. STS-1 flight performance has been evaluated by trajectory simulations utilizing propulsion tag values and specified SRB performance. In addition to flight performance reserves and intact abort, yaw steering, insertion point accuracy, ET disposal and flight personnel loads have been evaluated and found satisfactory for STS-1 flight. (See Continuation Sheet)

JSC – 07700 VOL X VERIFICATION REQUIREMENTS COMPLETED

3.2.1.1.2	3.2.1.1.3.4	3.2.1.1.17
3.2.1.1.3	3.2.1.1.4	3.2.1.2.14
3.2.1.1.3.1	3.2.1.1.4.1	3.2.2.1.17.2 ←
3.2.1.1.3.2	3.2.1.1.5	3.2.2.1.17.2.1
3.2.1.1.3.3.1	3.2.1.1.6	3.3.1.2.2.1
3.2.1.1.3.3.2	3.2.1.1.11	3.3.1.3.3.10
		3.3.1.2.5.1

(SEE THE BACK SIDE OF THIS SHEET FOR INCOMPLETE VERIFICATION OR EXCEPTIONS)

APPROVALS

Except as noted on the back side of this sheet:

(1) Endorsement of this VCN by the Project Manager shall signify completion of element verification to the extent identified on the above cited IVLN and as further defined in the applicable Shuttle Master Verification Plans.

(2) Endorsement by the Systems Integration Technical Manager shall signify completion of the verification tasks and products defined in the attached V&I-2 data file segments.

(3) Systems Integration Manager approval signifies completion of the total integrated systems verification task for the identified networks.

	ROCKWELL		NASA		
	SIGNATURE AND ORGN	DATE	SIGNATURE AND ORGN	DATE	SIGNATURE AND ORGN / DATE
TECHNICAL MANAGEMENT	[illegible] - Ops	11/4/81			
	[illegible] - Avionics	11/4/81			
	[illegible]				
	[illegible] PRR SRB'S - MPAD	11-5-81			
	[illegible] Avionics	11-5-81			
	[illegible]	11/5/81			
	[illegible]	11/5/81			
	[illegible]	11/5/81			
	[illegible]	11/5/81			
	[illegible]	11/5/81			
	[illegible] EXT. LOADS	11/6/81			
	[illegible] CPE	11/9/81			
PROJECT			[illegible] for Aaron Cohen	1-20-81	
			[illegible]		
			[illegible]	1-9-81	
			[illegible]	1-19-81	
			[illegible]	1-19-81	
			[illegible]	1/16/81	
			[illegible]	1/15/81	
			[illegible]	1/16/81	
SYSTEM	[illegible] PROGRAM MANAGEMENT:	1/16/81	[illegible] PROGRAM MANAGEMENT:	3-17-81 SIGNATURE/DATE	(24)

Flight 5 - Scale

Rockwell International
Space Systems Group

SPACE SHUTTLE SYSTEM
INTEGRATED ELEMENTS

NASA

VERIFICATION COMPLETION NOTICE

IVLN NO.: 12

IVLN TITLE: BASELINE MISSION CAPABILITY

PAGE 1 OF 2

VCN NO.: 12A21

DATE: OCT 8 7 1982

CONSTRAINT: OPS

SCOPE OF IVLN

This Baseline Mission Capability IVLN identifies the activities and interrelationships occuring during the Orbital Flight Test phase of the program applicable to verifying the capability of the vehicle to perform the baseline missions specified in the applicable paragraphs of JSC-07700-10. The verification activities accomplished prior to the first flight are identified in VCN No. 12A11. (See continuation sheet)

JSC — 07700 VOL X VERIFICATION REQUIREMENTS COMPLETED

3.2.1.1.2	3.2.1.1.3.4	3.2.1.1.17
3.2.1.1.3	3.2.1.1.4	3.2.1.2.14
3.2.1.1.3.1	3.2.1.1.4.1	3.2.2.1.17.2 ←
3.2.1.1.3.2	3.2.1.1.5	3.2.2.1.17.2.1
3.2.1.1.3.3.1	3.2.1.1.6	3.3.1.2.2.1
3.2.1.1.3.3.2	3.2.1.1.11	3.3.1.3.3.10

(SEE THE BACK SIDE OF THIS SHEET FOR INCOMPLETE VERIFICATION OR EXCEPTIONS)

APPROVALS

Except as noted on the back side of this sheet

(1) Endorsement of this VCN by the Project Manager shall signify completion of element verification to the extent identified on the above cited IVLN and as further defined in the applicable Shuttle Master Verification Plan.

(2) Endorsement by the Systems Integration Technical Manager shall signify completion of the verification tests and products defined in the attached VIS-2 data file segments.

(3) Systems Integration Manager approval signifies completion of the total integrated systems verification task for the identified network.

	ROCKWELL		NASA			
	SIGNATURE AND ORGN	DATE		SIGNATURE AND ORGN	DATE	SIGNATURE AND ORGN	DATE
TECHNICAL MANAGERS	R.H.Lassen Acoustics	11-4-82	PROJECT			J. Lovingood	12-8-82
	R.H.Lassen Flutter	11-4-82				R. H. Gray	11-5-82
	R.A. Weaver Integ.	11-5-82		J.B. Odom	12-8-82		
	R.A.Burg MPS	11-5-82		L.B. Mulloy	12-8-82		
	W.T.Schleich AGN&C	11-15-82	TECHNICAL MGRS	M.K. Craig Sep	11-3-82	K.J. Cox GN&C	11-3-82
	W.T.Schleich Sep	11-15-82		B.B. Roberts Sep	11-3-82	H.J. Brasseaux MPS	12-14-
	J.W.Haney Aerothermo	11-22-82		J.G. Hondros Aero	11-16-82	M.C. Coody Acou	11-2-82
	M.R.Schall AscFlt Perf.	11-15-82		J.G. Hondros Asc Flt Perf.	11-16-82	G. Strouhal Therm	11-15-
	E.D.Schlessinger Therm	11-22-82		A.L. Mackey Ext Lds	11-10-82		
	W.E. Bornemann Aero	11-24-82		D.B. Lee Thermo	11-22-82		
	R.E. Gatto Ext lds	12-1-82		C.T. Modlin Flutter	11-29-82		
SYSTEM	*signatures on file B. A. Gerstner PROGRAM MANAGEMENT:		SYSTEM	PROGRAM MANAGEMENT:		SIGNATURE/DATE	

(25)

VI-B

SOLID ROCKET MOTOR TEAM OVERVIEW COMMITTEE

ALLAN NORTON, CHAIRMAN, MARTIN MARIETTA/ORLANDO
VICE PRESIDENT, ELECTRONIC SYSTEMS DIVISION

MICHAEL CARD, NASA LANGLEY RESEARCH CENTER
CHIEF, STRUCTURES AND DYNAMICS DIVISION

AARON COHEN, NASA JOHNSON SPACE CENTER
DIRECTOR, RESEARCH AND ENGINEERING

MAXIME FAGET, NASA JOHNSON SPACE CENTER (RETIRED)
SPACE INDUSTRIES

CHARLES FELTZ, ROCKWELL INTERNATIONAL (RETIRED)

LEONARD HARRIS, NASA HEADQUARTERS
DIRECTOR FOR SPACE

HORACE LAMBERTH, NASA KENNEDY SPACE CENTER
DIRECTOR, SHUTTLE ENGINEERING

ADRIAN O'NEAL, MCDONNELL DOUGLAS CORP., HUNTSVILLE

DOMINIC SANCHINI , ROCKWELL INTERNATIONAL

SAMUEL TENNANT, AEROSPACE CORPORATION

DAVID L. WINTERHALTER, NASA HEADQUARTERS
ACTING DIRECTOR, PROPULSION DIVISION

JOHN YOUNG, NASA JOHNSON SPACE CENTER
CHIEF, ASTRONAUT OFFICE

June 19, 1986

NATIONAL RESEARCH COUNCIL
COMMISSION ON ENGINEERING AND TECHNICAL SYSTEMS
Committee on NASA Scientific and Technological Program Reviews

PANEL ON TECHNICAL EVALUATION OF NASA'S PROPOSED REDESIGN OF THE SPACE SHUTTLE SOLID ROCKET BOOSTER

NAE Dr. H. Guyford Stever, CHAIRMAN
NAS Foreign Secretary
National Academy of Engineering
2101 Constitution Avenue, N.W (NAS 307)
Washington, DC 20418

Mr. Laurence J. Adams
(Retired Former President,
Martin Marietta Corporation)
13401 Beall Creek Court
Potomac, MD 20854

Dr. David Altman
(Retired Manager of Engineering
United Technologies Company)
1670 Oak Avenue
Menlo Park, CA 94025

Mr. Robert C. Anderson
(Retired Vice President,
TRW Energy Development Group,
Electronics & Defense Sector)
Engineering and Management Consultant
840 55th Avenue, N.W.
Salem, Oregon 97304

Dr. Jack L. Blumenthal
Chief Engineer
TRW
Materials and Chemistry Applications
1 Space Park, Building 01, Room 2010
Redondo Beach, CA 90278

Dr. Robert C. Forney
Executive Vice President
E.I. duPont de Nemours & Co., Inc.
Administration Department
DuPont Building, Room 9000
Wilmington, DE 19898

Dr. Alan N. Gent
Professor of Polymer Physics
The Institute of Polymer Science
University of Akron
Auburn Science Building, Room 325
Akron, Ohio 44325

Mr. Dean K. Hanink
(Retired Manager of Engineering
Operations, Detroit Diesel)
145 Maple Crest Drive
Carmel, IN 46032

NAE Dr. James W. Mar
Jerome C. Hunsaker Professor of
Aerospace Education
Dept. of Aeronautics & Astronautics
Massachusetts Institute of Technology
One Surf Way, Apt. 119
Monterey, CA 93940

MIT Office:

Professor Edward W. Price
Regents' Professor
School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, GA 30332

Mr. Robert D. Watt
(Retired Group Leader,
Stanford Linear Accelerator Center)
11117 Palos Verde Drive
Cupertino, CA 95014

NRC Staff

National Research Council, JH 415
2101 Constitution Avenue, N.W.
Washington, DC 20418

Dr. Robert H. Korkegi (Co-Director)

Dr. Myron F. Uman (Co-Director)

Ms. Viviane Scott (Adm. Assistant)

VI-C

INFORMATION REQUIREMENT DESCRIPTION

Title:	Type	No.	Date Rev.
Failure Mode Effects Analysis	2	RA-2678B	

Submittal Schedule:

Submit updates at 6-month intervals with submittal linked to nearest schedule CIL FRR submittal.

Contract SOW Reference:

Exhibit A, 3.3.3 - Engineering Support

Use:

To identify critical failure modes to be used as a basis for support of: (1) Additional Design Action; (2) Safety Analysis; and (3) Mission Contingency Planning.

Interrelationship:

Scope-Contents-Format-Maintenance-Government Furnished Data:

1. Scope/Contents - Failure mode effects analysis will be prepared for each Orbiter Vehicle Subsystem, including the following:

   a. System/Subsystem/Assembly/Item - Identification of item for which the FMEA is being conducted.

   b. Prepared/Approved By - Identification of analyst who performed the FMEA and individuals responsible for overall FMEA effort.

   c. Revision - Date individual pages are revised.

   d. Item Identification:

      (1) Name

      (2) Identification Number - Drawing number by which the Contractor identifies and describes each component or module.

      (3) Drawing Reference Designation - Identification of the component or module on the schematic.

      (4) Quantity - Total number of items in the subsystem.

   e. FMEA Number - A number that uniquely identifies the subsystem, component, and failure mode.

   f. Function - Concise statement of the function performed.

INFORMATION REQUIREMENT DESCRIPTION

Title:	No.	Date Rev.
Failure Mode Effects Analysis	RA-267EB	

2. Scope/Contents (Continued)

g. Failure Mode and Cause - Identification of the specific failure mode after considering the four basic failure conditions:

(1) Premature operation.
(2) Failure to operate at a prescribed time.
(3) Failure to cease operation at a prescribed time.
(4) Failure during operation.

For each applicable failure mode, describe the major cause(s) including operational and environmental stress factors, if known.

h. Mission/Phase - Phase of mission in which failure occurs, e.g., Prelaunch: checkout, countdown; Flight: boost phase, earth orbit, etc.

i. Failure Effect on - Subsystem, interfacing subsystem, mission/crew, element and/or vehicle as required.

j. Failure Detection Method - A description of the methods by which the failure could be detected.

k. Corrective Action - An identification of corrective action, automatic or manual, which would be taken to circumvent the failure. Include statement of alternate means of operation and redundancy available after failure.

l. Failure Mode Criticality Category Designation - Categorize the failure mode criticality in relation to crew safety and mission effect. Include an identification of all items not meeting redundancy requirements during intact aborts.

Equipment other than criticality 1 shall be further evaluated in accordance with the redundancy hardware screens described below. A notation will be made identifying each screen the hardware does not pass.

(1) The redundant elements are not capable of checkout during the normal mission turnaround sequence, or

(2) Loss of a redundant element if not readily detectable by the flight crew, or

(3) All redundant elements can be lost by a single credible cause or event such as contamination or explosion.

m. Ground Rules and Assumption - Statement of all ground rules and assumptions followed during the performance of FMEA.

INFORMATION REQUIREMENT DESCRIPTION

Title:	No.	Date Rev.
Failure Mode Effects Analysis	RA-267E3	

2. Scope/Contents (Continued)

   n. Remarks/Hazards - Statement of any remarks, recommendations, and potential hazards as required.

   o. Vehicle Effectivity - Identification of the vehicle effectivity for the failure mode identified.

3. Format - To be prepared in Contractor's format.

4. Maintenance - To be maintained by page revision/total reissuance, as applicable.

5. Government Furnished Data - Not applicable.

INFORMATION REQUIREMENT DESCRIPTION

Title:	Type	No.	Date Rev.
Failure Mode Effects Analysis	2	RA-267E3	

Submittal Schedule:

Submit updates at 6-month intervals with submittal linked to nearest schedule CIL FRR submittal.

Contract SOW Reference:

Exhibit A, 3.3.3 - Engineering Support

Use:

To identify critical failure modes to be used as a basis for support of: (1) Additional Design Action; (2) Safety Analysis; and (3) Mission Contingency Planning.

Interrelationship:

Scope-Contents-Format-Maintenance-Government Furnished Data:

1. Scope/Contents - Failure mode effects analysis will be prepared for each Orbiter Vehicle Subsystem, including the following:

   a. System/Subsystem/Assembly/Item - Identification of item for which the FMEA is being conducted.

   b. Prepared/Approved By - Identification of analyst who performed the FMEA and individuals responsible for overall FMEA effort.

   c. Revision - Date individual pages are revised.

   d. Item Identification:

      (1) Name

      (2) Identification Number - Drawing number by which the Contractor identifies and describes each component or module.

      (3) Drawing Reference Designation - Identification of the component or module on the schematic.

      (4) Quantity - Total number of items in the subsystem.

   e. FMEA Number - A number that uniquely identifies the subsystem, component, and failure mode.

   f. Function - Concise statement of the function performed.

INFORMATION REQUIREMENT DESCRIPTION

Title:	No.	Date Rev.
Failure Mode Effects Analysis	RA-26723	

2. Scope/Contents (Continued)

g. Failure Mode and Cause - Identification of the specific failure mode after considering the four basic failure conditions:

(1) Premature operation.
(2) Failure to operate at a prescribed time.
(3) Failure to cease operation at a prescribed time.
(4) Failure during operation.

For each applicable failure mode, describe the major cause(s) including operational and environmental stress factors, if known.

h. Mission/Phase - Phase of mission in which failure occurs, e.g., Prelaunch: checkout, countdown; Flight: boost phase, earth orbit, etc.

i. Failure Effect on - Subsystem, interfacing subsystem, mission/crew, element and/or vehicle as required.

j. Failure Detection Method - A description of the methods by which the failure could be detected.

k. Corrective Action - An identification of corrective action, automatic or manual, which would be taken to circumvent the failure. Include statement of alternate means of operation and redundancy available after failure.

l. Failure Mode Criticality Category Designation - Categorize the failure mode criticality in relation to crew safety and mission effect. Include an identification of all items not meeting redundancy requirements during intact aborts.

Equipment other than criticality 1 shall be further evaluated in accordance with the redundancy hardware screens described below. A notation will be made identifying each screen the hardware does not pass.

(1) The redundant elements are not capable of checkout during the normal mission turnaround sequence, or

(2) Loss of a redundant element if not readily detectable by the flight crew, or

(3) All redundant elements can be lost by a single credible cause or event such as contamination or explosion.

m. Ground Rules and Assumption - Statement of all ground rules and assumptions followed during the performance of FMEA.

INFORMATION REQUIREMENT DESCRIPTION

Title:	No.	Date Rev.
Failure Mode Effects Analysis	RA-267E3	

2. Scope/Contents (Continued)

   n. Remarks/Hazards - Statement of any remarks, recommendations, and potential hazards as required.

   o. Vehicle Effectivity - Identification of the vehicle effectivity for the failure mode identified.

3. Format - To be prepared in Contractor's format.

4. Maintenance - To be maintained by page revision/total reissuance, as applicable.

5. Government Furnished Data - Not applicable.

VI-D

Date: January 31, 1984
Page 1 of 34
Approved: V.P. Ostrander, Mgr., Reliability Space Shuttle Program

RELIABILITY

DESK INSTRUCTION

No. 100-2G

FLIGHT HARDWARE

FAILURE MODE EFFECTS ANALYSIS (FMEA) & CRITICAL ITEMS LIST (CIL)

References: (1) EOM 70 1-5.1.1.1
70 2-6.3.1

(2) Reliability Desk Instruction No. 100-1 - Reliability Evaluation

(3) Reliability Desk Instruction No. 100-12 - Shuttle Element Interface

INTRODUCTION

Reliability of the design is the ultimate responsibility of Design. However, it is incumbent on other Engineering functions, including Reliability, to support the design engineer in discharging his responsibilities. The Failure Mode Effect Analysis (FMEA) is primary reliability technique for providing design and program support and constitutes a documented record of the design status and coordinated decisions.

1.0 PURPOSE

This desk instruction defines the procedures for generating, documenting and maintaining Failure Mode Effects Analyses (FMEA) and Critical Items Lists (CIL) for the Space Shuttle Orbiter subsystems in order to verify design adequacy with respect to inherent reliability.

2.0 DEFINITIONS

1. Failure - is the inability of a system, subsystem, component, or part to perform its required function within specified limits under specified conditions for a specified duration.

2. Failure Mode - a description of the manner in which an item can fail.

3. Hazard - is the presence of a potential risk situation caused by an unsafe act or condition.

4. Redundancy (depth of) - describes the available (number of) ways of performing a function.

5. Backup Mode of Operation - describes the available ways of performing a function utilizing "like" (identical) hardware.

6. Alternate Mode of Operation - describes any additional ways of performing a function utilizing "unlike" hardware.

7. Criticality - is the categorization of a hardware item by the worst case potential direct effect of failure of that item. In assigning hardware criticality, the availability of redundancy (backup or alternate) modes of operation is considered. Assignment of functional criticality, however, assumes the loss of all redundant (backup or alternate) hardware elements. The definition of criticality is shown in Table 2.0.

Table 2.0 - Criticality Definition

CRITICALITY	POTENTIAL EFFECT OF FAILURE
1	Loss of life or vehicle.
2	Loss of mission.
3	All others.
1R	Redundant hardware element, all of which if failed, could cause loss of life or vehicle.
2R	Redundant hardware element, all of which if failed, could cause loss of mission.

NOTE: See Appendix B, paragraph 3.1.1, Ground Rules, sub-paragraphs 1 and 2.

8. Single Failure Point (SFP) - is a single item of hardware, the failure of which would lead directly to loss of life, vehicle, or mission. Where safety considerations dictate that abort be initiated when a redundant item fails, that item is also considered a single failure point.

9. Functional Mode - identifies each function to be performed by the item being analyzed.

10. Multiple Order Failure - describes the failure due to a single cause or event of all units which perform a necessary (critical) function.

11. Critical Item - a single failure point and/or a redundant element in a life or mission-essential application where:
    a. Redundant elements are not capable of checkout during the normal ground turnaround sequence.
    b. Loss of a redundant element is not readily detectable in flight.
    c. All redundant elements can be lost by a single credible cause or event such as contamination or explosion.

12. Kit - For the purposes of this desk instruction, a kit is defined as a temporary addition or modification to the Orbiter or its subsystems to satisfy unique requirements for a specific mission.

13. Post Landing Safing Operations - For the purposes of this desk instruction, post landing safing operations are defined as those activities performed after landing to prepare the Orbiter for hangar operations. This includes the deservice and draining of all hazardous fluids, safing of unused ordnance, application of ground power and cooling, removal of potentially hazardous components, pods and payloads, purging and venting of gases and the installation of protective covers.

14. Prelaunch Operations - Prelaunch operations for propulsion subsystems is defined as beginning with propellant loading for each specific subsystem. For all other subsystems prelaunch operations commence with start of main engine conditioning.

3.0 FMEA/CIL PREPARATION

FMEA's will be prepared jointly by the responsible designer and the assigned Reliability Subsystem Analyst (RSA) in accordance with the attached format, Appendix D (Ground Rules and Criteria) and as shown in FIGURE 1. Safety, other engineering disciplines, and technical support functions (see EOM Directive

70 2-6.3.1) will provide support as required. Where the FMEA deviates from the instructions and ground rules contained herein, appropriate notation will be included within the "Ground Rules and Criteria" of the FMEA preface.

3.1 SCHEDULE

Reliability, in coordination with Design, will define the schedule and depth of detail for each FMEA to be prepared for the Orbiter in support of contractual requirements, and issue an FMEA schedule.

3.2 CONTENT

Each subsystem FMEA/CIL shall be prefaced by the following information, sequenced as indicated:

1.0 INTRODUCTION
2.0 QUALITATIVE RELIABILTY SUMMARY
  2.1 SUBSYSTEM DESCRIPTION AND EFFECTIVE DATE
  2.2 SIGNIFICANT UNDEFINED DESIGN AREAS
  2.3 CRITICAL ITEMS SUMMARY
3.0 GROUND RULES AND CRITERIA
4.0 DISPLAYS AND CONTROLS INDEX
5.0 LIST OF REFERENCE DOCUMENTS
6.0 SCHEMATICS

The backup information, including rationale and anaylses involved in assessing failure modes and their effects, generally is not included in the final FMEA and CIL package. Where such information exists in the form of notes, calculations, IL's, references and other similar material, it will be retained by the responsible RSA. Should the RSA be reassigned, he will turn over the material to his supervisor.

3.3 ANALYSIS REQUIREMENTS

1. FMEA's will be performed for each functional mode of a subsystem or functional kit. Electrical FMEA's will be conducted to the "black box" level and within the "black box" to pursue functions which have single failure point potential effect on the orbiter safety or mission success. The level of detail required in mechanical FMEA's below the component level in pursuit of critical failure modes will vary. Standard design, such as check valves, relief valves, isolation valves, etc., require only common types of failure causes to be listed.

EXAMPLE: Failure Mode - internal/external leakage.
Cause - poppet/seat damage, contamination, structural failure.

When a component is a non-standard type of design or is unique in application or contains unusual/unique failure modes of a critical nature, a more detailed analysis is required. Piece parts and their failure modes and effects that could result in component critical failure modes must be identified and included in the "CAUSE" section of the component FMEA for each component failure mode of concern.

EXAMPLE: Spring - fracture, structural failure - poppet fails to seat.

2. FMEA's for mechanical systems and avionics will interface at the connector. (See section 4.3.4, Mechanical/Electrical Interface.)

3. All identified failure modes will be assigned two criticalities (functional and hardware) based on the definitions in section 2.0, Definitions, and procedures contained in sections 4.3.1 and 4.3.2, Hardware and Functional Criticality Determination.

4. The criticality assigned to pressure carriers (pressure lines and vessels) shall reflect the worst case failure effect. These include potential shrapnel damage to the vehicle/subsystems resulting from rupture of non-filament wound tanks, potential overpressurization caused by releasing substantial quantities of fluids from ruptured lines or tanks, or depletion of consumables. Where released fluids are flammable or oxidizers and the possibility of an ignition source exists, appropriate notation will be entered under "HAZARDS" for Safety action. (See Appendix B, paragraph 3.1.1, Ground Rules, subparagraphs 13, 14, and 15.)

5. Failures which could occur during all mission phases from prelaunch through deactiviation (including safing & purging) of subsystems subsequent to landing and during ferry flights shall be considered, regardless of occurrence probability. Documentation of prelaunch analysis is required only for items classified as criticality 1/1.

6. All ordnance/pyrotechnic items will be listed in the CIL according to the most severe effect (criticality 1 or 2) of a premature operation.

7. Each hardware or function critical item summary will include a count of the total number of critical failure modes per item, by criticality, classified either structural or functional (see paragraph 4.1.11).

8. Critical item summaries for kits will be included, but identified separately.

9. FMEA's will not be required on structures, wire harnesses, cables and electrical connectors. For all critical circuits where a short between adjacent connector contacts could reslut in loss of crew (MSC D&P Standard No. 32), the design schematics shall be reviewed to verify that this condition does not exist. The incorporation of a switch on the ground side that precludes an adjacent contact short to result in crew loss is considered acceptable for meeting the MSC D&P Standard No. 32 requirement.

   For all other critical circuits, separation of redundant functions will be verified by selective review of design schematics to insure that the requirements for separation have been incorporated and complied with.

10. Logic diagrams (ref. Desk Instruction 100-1, Reliability Evaluation) will be developed only where required to provide proper correlation between schematics and FMEA's.

11. Those components that are criticality 3 (functional and hardware) in the electrical circuits by "black box" criticality may be listed on one FMEA for for that circuit. Those components that are hardware criticality 1 or 2 will have individual FMEA's. Those components that are criticality 1R or 2R, and appear in the CIL, will have individual FMEA's.

4.0 IMPLEMENTATION

A program has been developed to provide computer printout of FMEA and CIL data. Format examples of these printouts are shown in FIGURES 2 and 3. The following section contains instructions for documenting the FMEA. Data entry

sheets (FIGURES 4 and 5) will be completed by the RSA as information becomes available. The information will be entered into the computer and the RSA will receive a copy of the resultant data printout (FIGURES 6 and 7) which will comprise a working document of the information stored in the computer and a baseline for additional inputs or revisions.

4.1 DATA ELEMENTS

The following procedure describes the information to be filled out on Data Sheets 1 and 2 (FIGURES 4 and 5). Each data descriptor is preceded by the entry code for that item (e.g., LV1, Subsystem ID). These codes also are shown on the examples of the FMEA and CIL formats, FIGURES 2 and 3, for information.

DATA SHEET NO. 1

4.1.1 (DI, LV1, LV2) DATA IDENTIFIER: This line uniquely identifies the component being analyzed and the "update" information to be taken.

a. Circle "A", "R" or "D" to indicate appropriate action --
   A - Add a new record (component or assembly).
   R - Review an existing record by adding, deleting, or revising an element(s) of that record.
   D - Delete an entire record and all information in that record.

b. SUBSYSTEM ID (LV1): Enter the last two digits of the applicable designator and dash number. (See TABLE 4.0).

c. COMPONENT ID (LV2): Enter a number which uniquely identifies the particular component being available. If an existing schematic identifier is available, it may be used. For computer printout purposes, the first digit(s) of the number shall be selected to indicate the assembly. The use of special characters such as periods or dashes will be avoided.

4.1.2 (C1) ASSEMBLY NAME: Enter the name of the assembly.

4.1.3 (C1, J1) ITEM NOMENCLATURE: Enter the nomenclature for the component. In the first block (C2), give the basic identifying noun. Enter any additional modifiers or description on the J1 line. A typical example is "Valve, Solenoid", where "valve" is the basic identifier.

Table 4.0 - IDENTIFIERS & SUBSYSTEM NAMES

01-5 PURGE, VENT & DRAIN

02-1 LANDING DECELERATION

02-2 DOCKING MECHANISM

02-3 SEPARATION MECHANISM

ACTUATION MECHANISMS

02-4 DOORS

ET Umbil door
Star Tracker
Air Data Sensor

02-4A HATCHES

02-4B PAYLOAD BAY DOORS

02-4C RUDDER/SPEEDBRAKE, BODY FLAP

02-5 PAYLOAD RETENTION/DEPLOYMENT MECHANISMS

02-6 HYDRAULICS

03-1 MAIN PROPULSION

03-2 REACTION CONTROL
03-2A AFT

03-2B FORWARD

03-3 ORBITAL MANEUVER

04-1 ELECTRICAL POWER - CYRO

04-1A ELECTRICAL POWER - FUEL CELL

04-2 AUXILIARY POWER (APU)

05-1 GUIDANCE, NAVIGATION & CONTROL

COMMUNICATIONS & TRACKING

05-2A AUDIO
05-2B UHF
05-2C TACAN
05-2D ALTIMETER
05-2F MICROWAVE SCAN BEAM LANDING (MSBLS)
05-2G S-BAND
05-2J PAYLOAD INTERRAGATOR
05-2K CLOSED CIRCUIT TV (TV)
05-2R KU-BAND COMM & RADAR

05-3 DISPLAYS & CONTROLS

05-4 INSTRUMENTATION

05-5 DATA PROCESSING & SOFTWARE & COMPUTERS

*05-6 ELECTRICAL POWER DISTRIBUTION & CONTROL

05-8 BACKUP FLIGHT CONTROL

06-1 ATMOSPHERIC REVITALIZATION (ARS, ARPCS, Airlock)

06-2 LIFE SUPPORT

06-3 ACTIVE THERMAL CONTROL & WATER SPRAY BOILER

07-1 CREW PROVISIONS, ACCOMMODATIONS & EMERGENCY EGRESS

07-2 CREW ESCAPE - 102 PRE-AAMOD ONLY

07-3 TUNNEL ADAPTER

*See TABLE 5.0 for EPD&C/INTERFACING SUBSYSTEM IDENTIFIERS

Table 5.0 - EPD&C/INTERFACING SUBSYSTEM IDENTIFIERS

ELECTRICAL	INTERFACE	MECHANICAL SUBSYSTEMS
05-6AA	01-5	Purge, Vent & Drain
05-6AB	01-5	Vent Doors
05-6B	02-1	Landing Deceleration
05-6BA	02-1	Landing Gear Control
05-6BB	02-1	Brake & Anti-Skid
05-6BC	02-1	Nosewheel Steering
05-6C	02-2	Docking Mechanism
05-6D	02-3	Separation
05-6DA	02-3	Carrier A/C Separation
		ACTUATION MECHANISMS SUBSYSTEMS
05-6EA	02-4A	Hatches
05-6EB	02-4B	Payload Bay Door
05-6EC	02-4C	Rudder/Speedbrake, Body Flap
05-6ED	02-4	ET Umbilical Doors
05-6EE	02-4	ADP Deploy & Htr
05-6EF	02-4	Star Tracker Doors
05-6EG	02-4	Freon Radiator Deploy
05-6EH	02-4	Rendezvous Radar & Comm. Antenna Deploy
05-6F	02-5	Payload Retention, Manipulator Positioning
05-6G	02-6	Hydraulics
05-6IA	02-5	Remote Manipulator Arm
05-6IB	02-5	Manipulator Deploy Control
05-6IC	02-5	Manipulator Latch Control
05-6ID	02-5	Manipulator Arm Shoulder Jettison & Retention Arm Jettison
05-6IE	02-5	DAC Camera-PLB OPS
		PROPULSION SUBSYSTEMS
05-6J	03-1	Main Propulsion
05-6KA	03-2A	Reaction Control-Aft
05-6KF	03-2F	Reaction Control-Fwd
05-6L	03-3	Orbital Maneuvering
05-6LA	03-3	OMS Auxiliary Kit
		POWER GENERATION SUBSYSTEMS
05-6MA	04-1A	Electrical Power Generation - Fuel Cell
05-6MB	04-1	Electrical Power Generation - Cyro
05-6N	04-2	Auxiliary Power Unit

Table 5.0 - (Cont'd.)

ELECTRICAL	INTERFACE	AVIONICS SUBSYSTEMS
05-6O	05-1	Guidance, Navigation & Control Communications & Tracking:
05-6PA	05-2A	Audio
05-6PB	05-2B	UHF
05-6PC	05-2C	TACAN
05-6PD	05-2D	Altimeter
05-6PF	05-2F	Microwave Scan Beam Landing (MSBLS)
05-6PG	05-2G	S-Band
05-6PJ	05-2J	Payload Interragator
05-6PK	05-2K	Closed Circuit TV (TV)
05-6PH		Ground Command Interface Logic (GCIL)
05-6PR	05-2R	Ku-Band Comm. & Radar
05-6Q	05-3	Displays & Controls
05-6R	05-4	Instrumentation
05-6S	05-5	Data Processing & Software
05-6T	05-8	Backup Flight Control
05-6	05-6	Electrical Power Distribution & Control
		ECLSS SUBSYSTEM
05-6U	06-1	Atmospheric Revitalizatin - ARS, ARPCS
05-6UA	06-1	Airlock Environmental Control
05-6V	06-2	Smoke Detection, Fire Suppression
05-6VA	06-1	ARPCS
05-6VB	06-2	Galley
05-6VC	06-2	Waste Management
05-6VD	06-2	Water Management
05-6W	06-3	Active Thermal Control
05-6Y	07-1	Crew Station & Equipment
05-6Z	07-2	Crew Escape

4.1.4 (J10) FUNCTION: Describe the function performed by the component. Also, enter the component designator(s) as identified on the design schematic.

4.1.5 (E4, C5, C6) QUANTITY: Enter the total number of items having identical part numbers performing the same function in the subsystem. The E4 field will reflect the total quantity in Arabic numerals. The C5 and C6 fields will reflect written quantities.

4.1.6 (C7, C8) PART NUMBER:

a. C7) ROCKWELL PART NUMBER: Enter the appropriate Rockwell part number in accordance with the following DRM/SRM example, starting at the most left-hand-position -

(1) V070-XXXXXX (Airborne, In-House)
(2) MEXXX-XXXX (SCD)
(3) MCXXX-XXXX (Procurement Spec)

Note: Dash numbers to basic part numbers are required when the basic part number has dash numbers having differences in the failure mode and effects.

b. (C8) SUPPLIER PART NUMBER & SUPPLIER NAME: Enter supplier part number and supplier name (abbreviate if required) when available for ME and MC part numbers.

4.1.7 (C11-14) REFERENCE DOCUMENTS: Enter the referenced schematic diagram first, followed by the related block diagrams, logic diagrams, etc.

4.1.8 (C9, C10) FMEA PREPARED BY: Enter the initials and last name of the Reliability Subsystem Analyst and Design Engineer who prepared the subsystem FMEA.

DATA SHEET NO. 2

4.1.9 (DI, LV1, LV1) DATA IDENTIFIERS: Indicate appropriate action and specify the subsystem and component ID as described for Data Sheet No. 1.

4.1.10 (LV3) FAILURE MODE SEQUENCE: Assign different sequence numbers (e.g., 1, 2 - 5) for various failure modes of the specified component.
Do not use leading zero's.
Example: (LV3) _ _ _ 1, NOT (LV3) 0 0 0 1.

4.1.11 (C31, J130, C32) FAILURE MODE: Enter first the basic failure mode (keyword) (C31), then any additional modifiers (J130) necessary to fully describe the specific failure mode - the exact manner in which the item fails. Failure mode keyword identifiers are listed below. Selection should include but not be limited to those listed.

FAILURE MODE KEYWORD IDENTIFIERS

- STRUCTURAL FAILURE (RUPTURE)
- PHYSICAL BINDING/JAMMING
- FAILS TO REMAIN OPEN/CLOSED
- FAILS MID-TRAVEL
- FAILS TO OPEN/CLOSE
- INTERNAL/EXTERNAL LEAKAGE
- FAILS OUT OF TOLERANCE
- INADVERTENT OPERATION
- INTERMITTENT OPERATION
- ERRATIC OPERATION
- ERRONEOUS INDICATION
- RESTRICTED FLOW
- FAILS TO START/STOP
- FAILS TO SWITCH
- PREMATURE OPERATION
- DELAYED OPERATION
- ERRONEOUS OUTPUT
- LOSS OF OR PARTIAL OUTPUT
- SHORTED
- OPEN (ELECTRICAL)
- LEAKAGE (ELECTRICAL)

Appendix B, paragraph 3.1.1, sub-paragraph 13, reflects the ground rule to be used for external leakage. For OV-102 pre-AA mod only, those failure modes which result in a criticality classification of 1 and 2, or 1R and 2R, and appear in the CIL (item 4.1.22) shall be classified further as structural or functional failures by circling "S" or "F" in the C32 field. The following guidelines apply:

STRUCTURAL (S) - A failure mode involving structural failure of a pressure vessel, component housing, fluid lines, attach fittings, or load-carrying members such as cranks or rods.

FUNCTIONAL (F) - A failure mode, generally within a component, which negates the described component function. This type of failure would include binding, leakage, failure to open or close, or loss of output. The failure cause could be improper installation of parts or structural failure of power transmitting parts such as gear teeth, shafts or springs; however, in such instances the mode is still classified as functional. Electrical and electronic component failures would normally fall in this category.

4.1.12 OV-102 PRE-AA MOD ONLY:

(C62-66) APPLICABLE MISSIONS: Enter an "X" in the block of the mission to which the FMEA applies.

Horizontal Flight Test	C62
Vertical Flight Test	C63
Ferry Flight	C64
Operational Flights	C65
Specific Orbital Mission	C66

Note: "Operational Flights" and "Specific Orbital Mission" are not to be used for the duration of OV-101 and OV-102 flight test programs.

OPERATIONAL VEHICLE(S):

(C83 - C86) VEHICLE EFFECTIVITY: Enter an "X" in the appropriate block(s) to which the FMEA appplies.

Orbiter Vehicle 102	C83
Orbiter Vehicle 099	C84
Orbiter Vehicle 103	C85
Orbiter Vehicle 104 & SUBS	C86

4.1.13 (C33-37) MISSION PHASE(S): Enter an "X" in appropriate box(es) to indicate when the specified effects would be manifested. If the failure occurs at discrete points in time within a given mission phase, and different effects may be observed, it may be necessary to define the subphase or event under "EFFECTS".

4.1.14 (C38, C58) ABORT CRITICAL COMPONENTS:

a. For those items whose criticality is increased to 1/1 during an abort resulting from unrelated failures, enter the word "Abort" (C38 - six spaces only), followed by the appropriate acronym(s); i.e.,

(C58) RTLS - Return to Landing Site
(C58) AOA - Abort Once Around
(C58) ATO - Abort to Orbit

b. For non-redundant modes where normal mission effect is criticality 3 but are hardware criticality 1 unique to intact abort, classify these modes

as hardware criticality 1 and functional criticality 1. Add in J10 (FUNCTION) the notation, "Unique to Intact Abort". Add appropriate intact abort notation in a. above.

Additional information must also be entered under J240, EFFECT(S) - see paragraph 4.1.16. NOTE: For SSME induced aborts, maximum two engine burn time is approximately twelve minutes. If "TIME TO EFFECT" is equal to or greater than twelve minutes, there is no change in criticality.

4.1.15 (J380) CAUSE(S): Enter causes including but not limited to those listed below and amplify as necessary. See paragraph 4.4.2 for instructions on supplier furnished piece parts.

CAUSES

• CONTAMINATION	• TEMPERATURE (HIGH/LOW)	• INADVERTENT OPERATION/ACTIVATION
• MECHANICAL SHOCK	• THERMAL SHOCK	• VIBRATION
• VACUUM	• PRESSURE (HIGH/LOW)	• PROCEDURAL ERROR
• ACOUSTICS	• IONIZING RADIATION	• CHEMICAL REACTION
• OVERLOAD	• ACCELERATION	• LOSS OF/IMPROPER INPUT
• MISHANDLING OR ABUSE	• ELECTROMAGNETIC FIELDS	• PIECE-PART STRUCTURAL FAILURE

4.1.16 (J240) EFFECT(S): Enter the letters (A), (B), (C) or (D) as defined in the headings of Appendix A, together with the words under each heading describing the effects on the subsystem, interfaces, mission, and crew/vehicle, respectively, and explain. If the identified effect is not listed, describe briefly. Where the effect is the same for two or more of the above, consolidate entries. Specify if there is no effect on a specific category or categories and provide a brief explanation. In those instances when time to abort requires automatic operation or immediate dependence on a parallel subsystem and such is provided, the effect on mission is "None" with explanation for each mission phase as appropriate. See section 4.1.21d. for screening of functional criticality 3 failure modes. For those items identified as abort critical (see paragraph 4.1.14) enter, subsequent to the (A), (B), (C) and (D) entries, the criticality and effects per the following example:

"Crit 1 for RTLS - Loss of additional engine-vehicle loss"

or

"Crit 1 for RTLS - Incomplete propellant dump, stability problem, probable vehicle loss."

Where functional criticality is 1R or 2R per paragraph 4.3.2 and hardware criticality is 3, the appropriate entry for "FUNCTIONAL" effects should be included. The "FUNCTONAL" effects entry relative to the loss of all functional redundancy will be entered per the following example:

(E) FUNCTIONAL CRITICALITY EFFECT:
Possible loss of crew/vehicle (specify) or probable loss of crew/vehicle (describe) or loss of crew/vehicle.

4.1.17 (C39) TIME TO EFFECT:

Immediate	-	less than 1 second
Seconds	-	1 to 50 seconds
Minutes	-	50 seconds to 50 minutes
Hours	-	50 minutes to 20 hours
Days	-	20 hours to mission completion

Enter the descriptor which indicates shortest credible time or time range available to correct the situation before the effect is manifested.

4.1.18 (C40-45) FAILURE DETECTABLE: Enter "YES" or "NO" in the block following "IN FLIGHT" and "GROUND TURNAROUND". If either answer is "YES", indicate how it can be detectable -- symptoms, instrumentation, etc. Include measurement number from MML (Master Measurements List) where applicable and available. (See section 4.3.5, Instrumentation FMEA's.) Development flight instrumentation (DFI) measurements will not be used as a means of detectability.

4.1.19 (J490) CORRECTING ACTION: Describe any action, automatic or manual, which may be taken to circumvent the specified failure. Also identify any alternate means (utilizing "unlike" hardware) of accomplishing the function performed by the item or its assembly. If none, so indicate. For instruments (sensors, transducers, etc.) that provide measurements assessed

as critical to vehicle/crew safety or mission continuation, the FMEA shall identify the redundant or alternate measurements by Measurement List identification number.

4.1.20 (C77) NUMBER OF SUCCESS PATHS REMAINING AFTER FIRST FAILURE: With respect to the item being evaluated, indicate the number of ways remaining to perform the function after the first failure. You may leave the block blank for non-critical functions.

4.1.21 (C53, C55-57) REDUNDANCY SCREEN: For all criticality 1R, 2 and 2R failure modes (see FIGURE 8 and paragraphs 4.3.1 and 4.3.2) circle "P" (PASS), "F" (FAIL), or "NA" (NOT APPLICABLE) for each of the following tests:

a. Redundant elements are capable of checkout during normal ground turnaround with no vehicle design modification. Where a subsystem is characterized by redundant strings and the status of each string can be verified during ground turnaround, no individual component(s) in any one string should be shown as failing this screen.

NOTE: This screen is not applicable under the following conditions:

(1) Pyrotechnic devices, excluding electrical control circuitry.

(2) Non-redundant item.

b. Loss of a redundant element is readily detectable during flight. Where a subsystem is characterized by redundant strings and the status of each string can be verified in flight, no individual component(s) in any one string should be shown as failing this screen.

NOTE: This screen is not applicable under the following conditions:

(1) Standby redundancy (redundant paths were only one path is operational at any given time).

(2) All functional paths of any subsystem which is inoperative (during such inoperative periods). This groundrule does not apply if the redundant elements are operative during any normal mission phase; i.e., the screen is considered applicable if the element is operative during any normal mission phase.

(3) Pyrotechnic devices.

(4) Mechanical linkage.

(5) Non-redundant item.

(6) Subtier level redundant functional path(s) (power/control circuits, etc., failures where the primary functional path (LRU, etc.) is

Criticality 1R3 or 2R3 and the primary redundancy would not be degraded (i.e., loss of two of the subtier functional paths would not result in an abort decision).

c. Failure of an element to pass this screen should be in direct relation to the noted failure mode under normally expected environmental conditions. Consideration of environmental extremes as caused which could induce "multiple order failure" is limited to abnormal conditions generally resulting from some other failure. Where multiple failures must first occur to result in environmental extremes, such events may be considered non-credible. As a ground rule, it may be assumed that hardware items will be qualified and properly installed to withstand the "design-to" environmental envelope. The following are typical questions to be answered in this phase of the analysis:

(1) Contamination:
   (a) Are the items being evaluated susceptible to contamination?
   (b) Is contamination a credible event or does the design (including filters) result in this failure mode being categorized as non-credible?

(2) Explosion:
   (a) Is there a credible source?
   (b) Must other multiple failures occur first to result in the explosion?
   (c) Is the explosion catastrophic to crew or vehicle?
   (d) Is the container frangible?
   (e) Are the items being considered susceptible to this type of damage in view of their physical characteristics and location; i.e., shielding?

3. Temperature:
   (a) Are components susceptible to damage or failure from high temperature?
   (b) Other than as a result of multiple failures, is such exposure credible? This implies temperature peaks or sustained levels sufficient to cause catastrophic effects on the component in a short time. For example, temperature increases to certain levels merely increase electronic parts failure rates - the actual failure and time of occurrence are still probabilistic.

(4) Vibration, Shock, Acceleration, Acoustics, etc.:

(a) Assuming that components are qualified and properly installed to withstand design environments, can a credible cause be identified which would cause these levels to be exceeded?

(b) Are vibration/shock/acceleration-sensitive redundant units physically oriented or separated to reduce the chance of multiple failure from the same cause(s) and is there sufficient analysis and test data to verify the failure as non-credible?

(5) Fire:

Do not consider fire as one of the single events or causes in failing screen "C". NASA has edicted that fire not be considered one of the events (NB/83-L 216).

If none of the redundancy screens are applicable, enter "NA" in the C53 field and briefly explain reason for REMARKS/HAZARDS.

d. Screening of Functional Criticality 3 Failure Modes

(1) Where the failure modes have been identified as non-critical for loss of all redundancy (Criticality 3), enter "NA" in the C53 field. Enter under "REMARKS" the notation, "Criticality 3 failure mode - loss of all redundancy would have no effect on the mission or crew/vehicle safety". In such cases, minimum entries on Data Sheet No. 2 consist of DI, LV1, LV2, LV3, C31, C53 and J240. For functional Criticality 3 items, J240 must contain a brief explanation regarding the assigned criticality.

(2) Where a component has an identified failure mode in the Criticality 1 of 2 category, and additional functional Criticality 3 failure modes are identified, these Criticality 3 modes will be treated as described in para. (1) above.

4.1.22 CRITICALITY:

a. (C54 - HARDWARE) - Enter 1, 2 or 3 based on the definitions in section 2.0 and the ground rules contained in section 4.3.1 and Appendix B, paragraph 3.1.1, sub-paragraph 1.

b. (C67 - FUNCTIONAL) - Enter 1, 2, 1R, 2R or 3 based on the definitions in section 2.0 and the ground rules contained in section 4.3.2 and Appendix B, paragraph 3.1.1, sub-paragraph 1.

4.1.23 (J500) REMARKS/HAZARDS: Identify potential hazards resulting from the specified failure. Enter the words "Hazard Potential" followed by appropriate explanation and any other comments or recommendations that might prove useful in evaluating the system. Indicate requirements for additional instrumentation, and any other special consideration.

4.1.24 (J600) DISPOSITION AND RATIONALE: For criticality 1 and 2 items, and/or 1R, 2R items that fail a redundancy screen and/or hardware criticality 2 items where the screen is NA, in all of the following categories to describe the retention criteria. Each category must reflect a description of rationale for retention of the item:

a. Design - Identification of design features which minimize the occurrence of the failure mode and causes.
b. Test - identification of specific tests accomplished to detect failure mode and causes during acceptance tests, certification tests, and checkout tests.
c. Inspection - Statement that specific inspection points are included to determine that specific failure mode causes are not inadvertently manufactured into the hardware.
d. Failure History - Provide an indication that the hardware or similar hardware has been used successfully and that a history of generic failures does not exist. If the hardware is new to this program, so state.

4.1.25 (C9, C10) APPROVAL: Responsible Reliability and Design approval signatures as follows:

a. Subsystem FMEA package: Design/Reliability Manager
b. Figure 2 (FMEA) - PREPARED BY: Design_____ Responsible
Reliability_____ Analyst's Name
APPROVED BY: Design_____ Signature
Reliability (Analyst) Signature
c. Figure 3 (CIL) - APPROVED BY: Design (Supervisor) Signature
Reliability (Supervisor) Signature

NOTE: The initial issue of a CIL sheet will be signed by the Reliability Supervisor. Signatures will not be required on subsequent issues unless the CIL sheet is revised.

02461 0 0

4.2 REVISIONS & SUBMITTALS

1. Revisions to the FMEA will be made as follows:

a. New Data:

(1) To identify new components or failure modes, use the data entry sheets and follow the instruction given in section 4.1.

(2) To add information to a component or failure mode record, either a blank data sheet or the appropriate page of the data printout working copy may be used.

(a) Data Entry Sheets - Using a blank data sheet, circle "R" (Revise) on the "Data Identifier" line (DI, FIGURE 4 or 5) and enter the correct subsystem/component/(failure mode) ID number to identify the record to which the information is to be added. Fill in complete blocks of information to be added (e.g., Disposition block), and submit for keypunching.

(b) Data Printout - Circle "R" (Revise) on the "Data Identifier" line (DI, FIGURES 6 and 7) of the record to which new information is to be added. Using a colored pen or pencil, enter the information in the appropriate blocks and submit for keypunch.

b. Data Entry Change:

Circle the "R" (Revise) on the "Data Idenifier" line (DI, FIGURE 4 or 5) and either "red-line" the appropriate sheet of the data printout or re-enter the data as it should appear, using the appropriate data entry sheet as described in section 4.2, paragraph

(a) Data Entry Sheets. To clear the "J" field of any remaining unwanted information, asterisk (*) the blank lines within the block on the master record and supporting record work sheets.

c. Data Deletion:

(1) To delete data, circle the "R" on the "Data Identifier" line (DI, FIGURE 6 or 7) of the appropriate data printout sheet, cross out the entry to be deleted with a colored pen or pencil

and submit for keypunching. If a blank data sheet is used, enter an asterisk (*) in the block which corresponds to the entry to be deleted.

(2) To delete the entire record (i.e., all data pertaining to a particular failure mode or component) and all related entries, circle the "D" (Delete) on the appropriate sheets of the data printout. Again, filling out the data identifier line of a blank data sheet will accomplish the same purpose. All information pertaining to a particular component or failure mode will be deleted.

d. Data Identifier Change:
To change a data identifier (LVl, 2 or 3), it is necessary to delete the entire record under the old number and re-entrer (add) under a new number. The B999/revision data on computer reports is automated and prints the date of the latest update or revision.

e. Identification of Revisions/Changes:
Identify each line changed with a vertical black bar on the left-hand margin of the page.

2. FMEA/CIL Submittal

a. Critical Items List (CIL)
Updates will include the following:
(1) Any new CIL items
(2) Updates to existing items having technical changes affecting the following sections:
(a) function
(b) failure mode
(c) failure effects
(d) criticality
(e) abort critical components
(f) failure detectability (redundancy screen)
Other changes will be incorporated when pages are submitted for the above reasons.

b. FMEA's
Updates of the FMEA's will be at six month intervals linked to nearest scheduled CIL FRR publication. Only those changes related to the CIL submittals (a above) and other technical changes will be submitted.

4.3 IMPLEMENTATION GROUND RULES (See also Appendix B - Ground Rules and Criteria)

4.3.1 HARDWARE CRITICALITY DETERMINATION

Hardware criticality will be determined by the categorization of the singular effect of the identified failure mode on the subsystem/vehicle (See FIGURE 10). FIGURE 8 illustrates the analytical logic for criticality determination of all functional hardware.

1. Reliability Engineering identifies hardware where if redundancy fails the effect would be critical.

2. Reliability and Design Engineering jointly identify those equipments with (single point) criticality 1 or 2 failure modes. Those equipments that are not criticality 1 because they incorporate redundancy are then screened further, as described in paragraph 4.1.21, and appropriate entries made in the FMEA data sheet.

NOTE: The criticality of instrumentation and test ports will be assessed according to their function. Test ports, when capped, shall be treated as a structural part of the component and not be considered further. Where instrumentation (e,g., pressure transducer) penetrates the wall of a component or line and structural failure of the joint would result in gross leakage, the failure mode shall be considered as a failure of the component or line. The criticality of the instrumentation, therefore, would not be affected in such instances.

3. The criticality of those systems which are to be used only in the event of an emergency shall be established strictly on the basis of direct failure effect on crew, vehicle, or mission, regardless of the number of prior failures which must occur before the use of the system is required.

All other backup or standby equipment (e.g., relief valves, cross-feed valves, etc.) shall be assigned criticality in the normal manner.

4.3.2 FUNCTIONAL CRITICALITY DETERMINATION

Functional criticality will be determined by the categorization of the effect on the subsystem/vehicle of loss of all redundancy (like or unlike) for the identified failure mode (See FIGURE 10). FIGURE 8 illustrates the analytical logic for criticality determination of all functional hardware.

1. Reliability Engineering identifies hardware if all like or unlike redundancy fails the effect would be critical.

2. Reliability and Design Engineering jointly identify those equipments with criticality 1R or 2R failure modes.

4.3.3 CIL CONTENT CRITERIA

1. The following classification of failure modes will be entered in the CIL:
   a. All functional/hardware criticality category 1/1's, 2/2's, and 1R2's.
   b. All criticality category 1R3's and 2R3's that fail one or more redundancy screens.
   c. All failure modes that become criticality category 1/1 during intact abort.

2. CIL Section 12.0 - Critical Items List orbiter modifications to support special missions:

   This section of the Critical Items List contains those critical items associated with Orbiter subsystems that have been added to or modified by Orbiter Mission Kits to support special missions. These CIL items will only apply to specific vehicle missions as noted in this specific CIL subsection.

   This CIL section contains the single failure points and criticality 1R and 2R CIL items identified by the Failure Mode Effects Analysis (FMEA) conducted on the Orbiter subsystems that have been added to or modified

to support special mission application. These vehicle changes are identified by individual Mission Kits which are installed specifically for these special missions and would be removed when the mission objectives have been achieved. Each vehicle change is identified by a Master Change Record (MCR) and is referenced in the applicable FMEA.

Each critical failure mode identified in the vehicle modification section is categorized on a separate Critical Items List form which includes the failure causes, effects, and rationale for retention. CIL dispositions and rationale are contained on individual CIL sheets and those that are generally applicable to all components are contained in Section 3.0.

A critical items list summary is included for each major vehicle modification. Additions will be made to this CIL section to maintain this document current with the vehicle flight configuration. CIL page revisions are indicated by revision date.

NOTE: Prior to each CIL submittal, notify the CIL coordinator of any input to CIL Section 12.0.

4.3.4 MECHANICAL/ELECTRICAL INTERFACE

For mechanical components having an electrical interface, the mechanical FMEA will consider only the effects of "black box" functional failure (e.g., loss of output, premature signal, etc.). Where it becomes necessary to conduct an FMEA within the "black box" because of the assigned criticality (1 or 2), the FMEA will be conducted by Avionics Reliability who will be provided with the following information in the mechanical FMEA regarding the failure effects on the mechanical system:

a. Enter under "CAUSE" (J380), each applicable failure mode of the electro-mechanical device reflecting the avionics malfunction causing the failure mode; i.e., loss of electrical power, premature electrical signal, etc.

b. Identify under "REFERENCE DOCUMENTS" (C11-14), the specific mechanical/avionics interface.

The electrical interface FMEA will be included in the appropriate section of the avionics FMEA. For criticality 1, 1R, 2 or 2R failure modes, the mechanical FMEA, which considers the effect of "black box" loss of function, will indicate in the "REMARKS" section the avionics FMEA number of the "black box". The avionics FMEA of the "black box" will contain a similar reference to the appropriate mechanical FMEA. At the earliest point in time when the mechanical analyst can ascertain that the "black box" is criticality 1, 1R, 2 or 2R, it shall be his responsibility to convey to the Avionics Reliability group copies of his worksheets to facilitate initiation of detailed avionics analysis effort.

4.3.5 INSTRUMENTATION FMEA'S

Instrumentation (e.g., sensors, signal conditioners, etc.) may be provided by either Avionics Instrumentation or by a specific Design group. In either case, instrumentation FMEA's will be included in the FMEA for the using subsystem. Criticality 3 instrumentation may be listed on one FMEA form by family or type. FMEA's for criticality 1 or 2 and criticality 1R and 2R instrumentation that fail a redundancy screen or the screen is "NA", will be completed in their entirety and included in the using subsystem CIL. Avionics Reliability will provide support as required to identify failure modes, retention rationale, etc.

A copy of each instrumentation FMEA completed by a Mechanical Reliability group will be provided to Avionics Reliability.

4.4 SUPPLIER FMEA UTILIZATION

4.4.1 GENERAL

In many instances, depending on the cost, complexity, and state of development of the design, suppliers will be required to develop and submit FMEA's reflecting their area of design responsibility. The submissions will precede the joint supplier/Rockwell PDR or CDR. (See the applicable PDRD for content requiremennts and submittal schedules.) FIGURE 9 shows the overall supplier FMEA flow as related to the in-house effort.

4.4.2 FMEA UTILIZATION

Upon receipt of a supplier FMEA, the responsible RSA will compare the identified failure modes with those called out on his corresponding subsystem FMEA and update his FMEA as required (see section 4.2) to include any failure modes not already identified relating to subsystem effect. Supplier FMEA's will be reviewed for single failure points below the black box level by Rockwell and analyzed for corrective action directly with the supplier as part of their design review. Where Rockwell does not concur with portions of the supplier analysis, telephone contact with the supplier Reliability Engineer normally should suffice to resolve any differences. If not, the matter shall be resolved through normal Rockwell data handling procedures. Supplier black boxes will be identified in the subsystem FMEA based on the supplier schematic or drawing part identification number. For criticality 1 or 2 and 1R or 2R (CIL only) electronic black boxes, the piece parts (or if all or many circuits, so state) identified by the supplier FMEA which are single point failures that have a direct critical effect on the vehicle will be described with reference to the supplier FMEA in the "CAUSE" section of the applicable failure mode identified at the subsystem level. Parts will be listed only in those cases where less than five parts are involved.

4.5 ELEMENT CONTRACTOR FMEA CORRELATION

Requirements and procedures for conducting interfacing analyses and for element integration tasks are contained in Reliability Desk Instruction 100-12 (Shuttle Element Interface).

4.6 GFE

For items identified as GFE hardware, NASA will identify those which require FMEA's and will perform FMEA's on the hardware identified to the level defined by their ground rules. Upon completion of the FMEA, NASA will provide Rockwell with a copy. In addition to the completed copy, a preliminary copy may be transmitted. Upon receipt of the GFE FMEA, Rockwell will evaluate the interface effects on the Orbiter defined by the GFE FMEA.

Appropriate comments shall be included to insure that this area is correct and complete. Rockwell will conduct FMEA's for all interfaces between CFE and GFE. The Rockwell FMEA will consider all failure modes consistent with this desk instruction. The analysis is to consider as a "CAUSE" any failure mode identified by the GFE FMEA which could produce a failure in the CFE interface. Where GFE failures are identified as a "CAUSE", the appropriate GFE FMEA and document number shall be identified as a part of the "CAUSE" section. In addition, those vehicle failures which could cause FMEA failure modes will be identified to NASA in the comments to the GFE FMEA. Where structural failures are identified, appropriate hazards analyses shall be included in available.

The accountability of CIL items for GFE will be NASA. Those CIL items resulting from interface failure modes will be a part of the Rockwell CIL.

Exceptions to this instruction will be identified and concurred in jointly by Rockwell and NASA and documented as a part of letters of agreement.

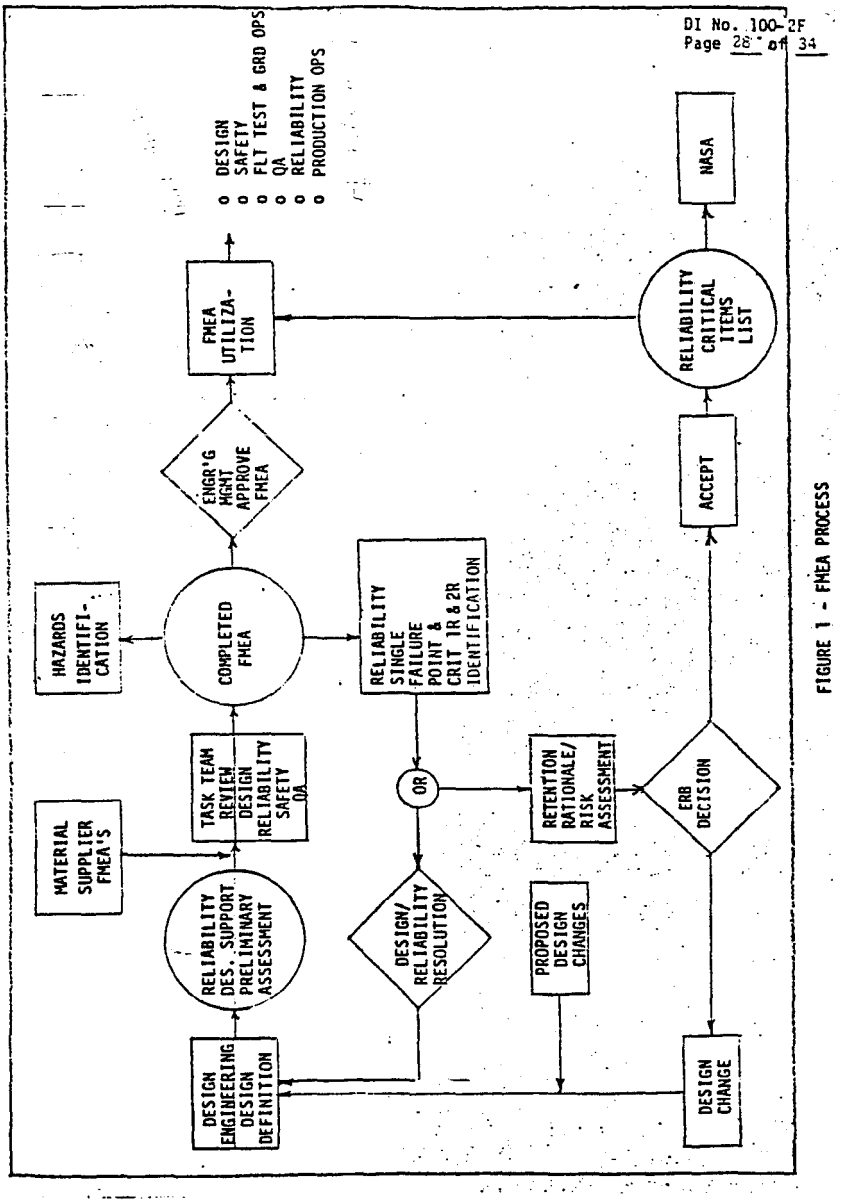


FIGURE 1 - FMEA PROCESS

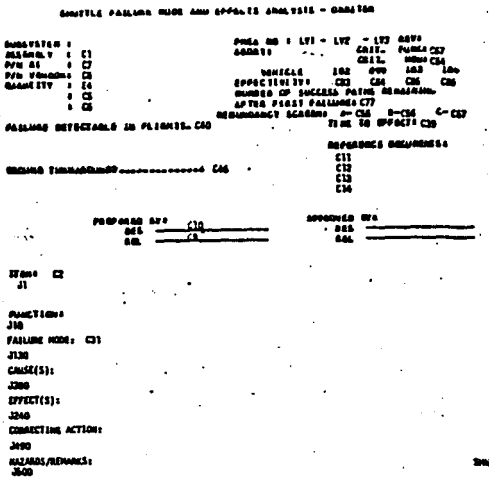


Figure 2. FMEA

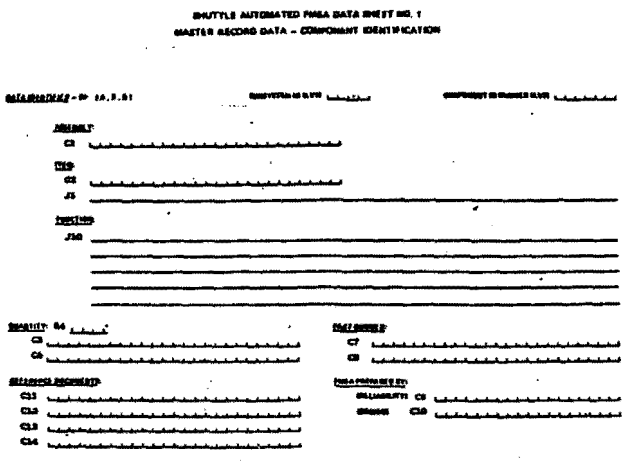


Figure 4. Data Sheet No. 1

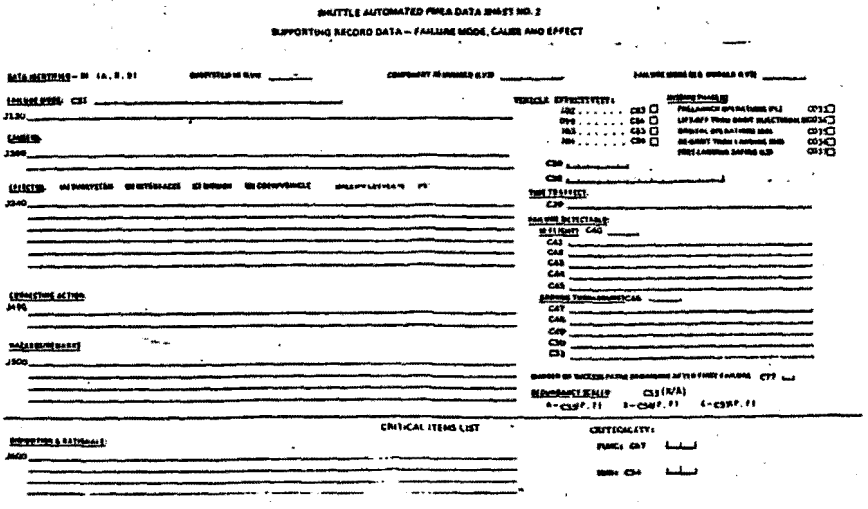


Figure 5. Data Sheet No. 2

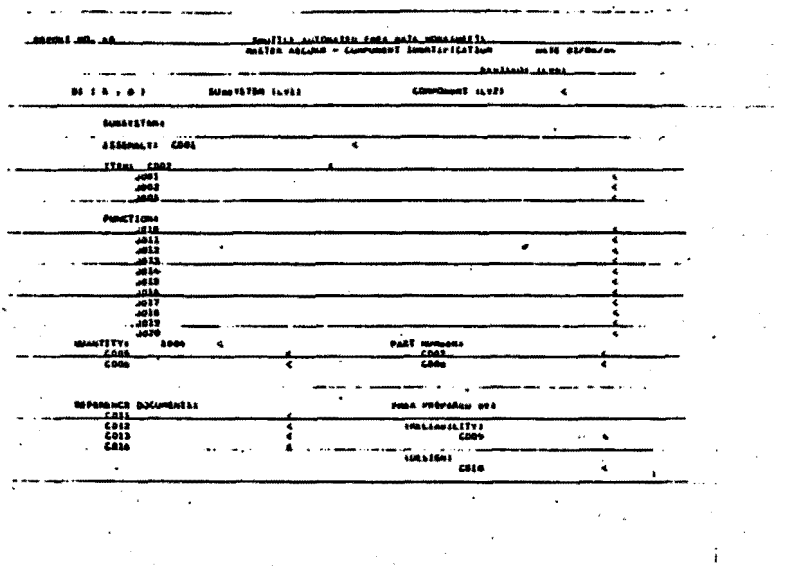


Figure 6. Data Printout, Component-Related Data

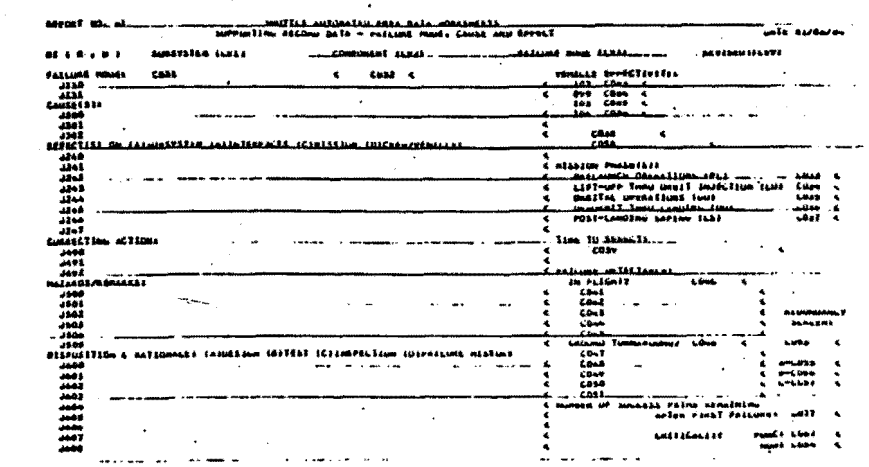


Figure 7. Data Printout, Failure Mode-Related Data

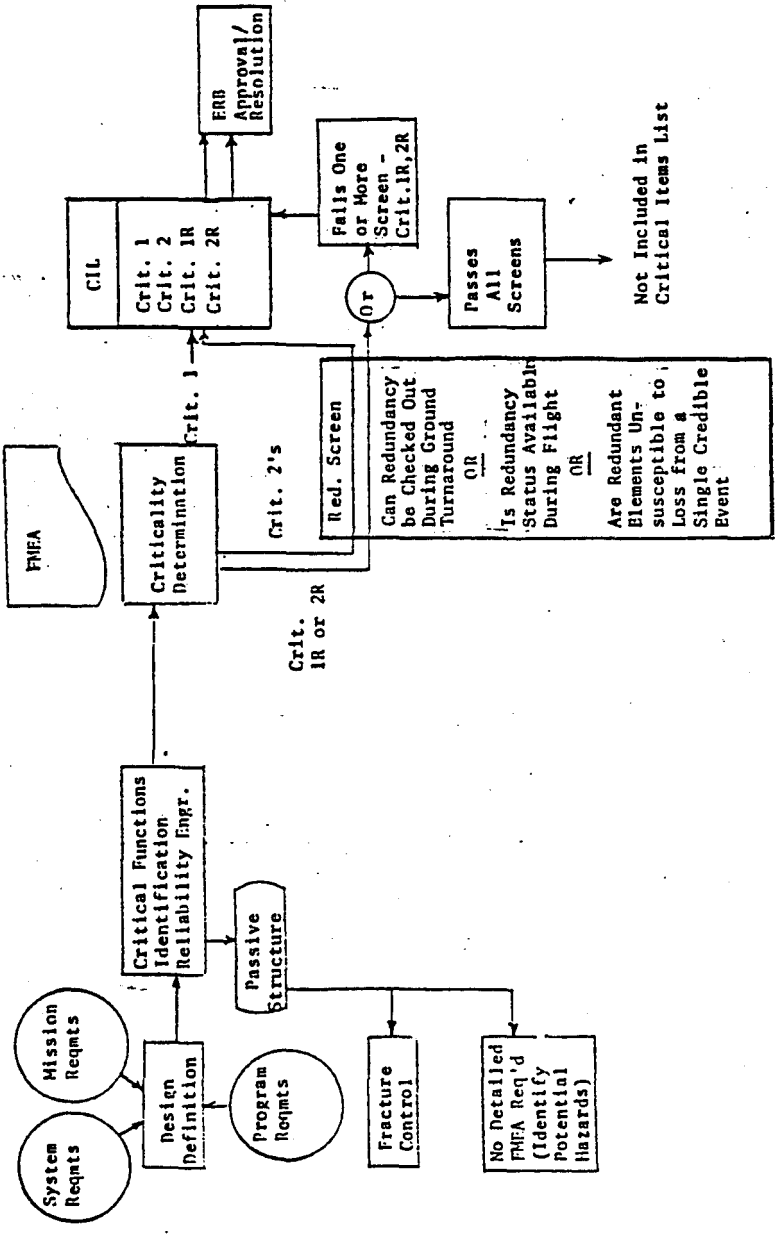


FIGURE 8 - FMEA SCREENING

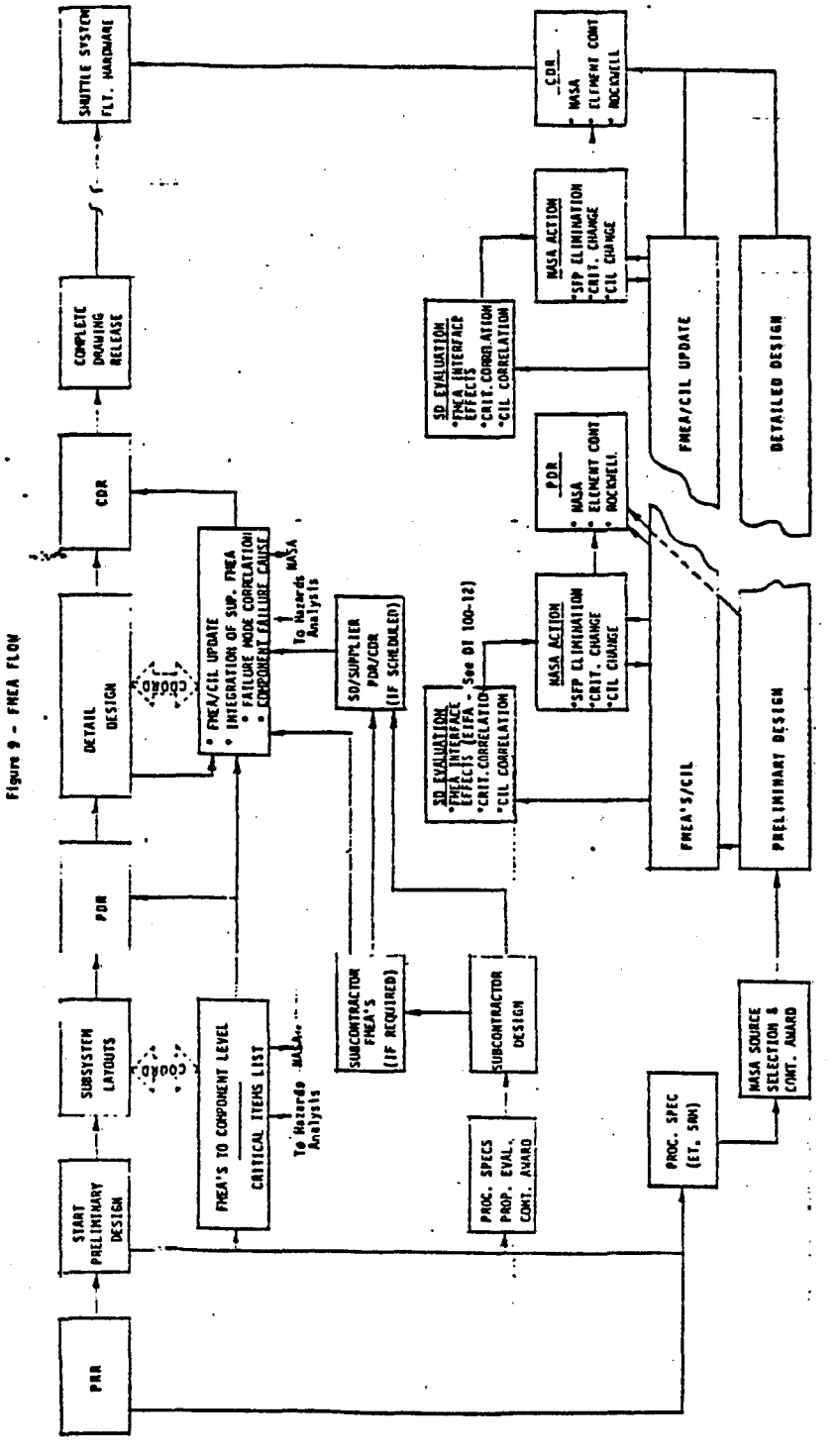


Figure 9 - FMEA FLOW

DI No. 100-2F
Page 34 of 34

CRITICALITY CATEGORY
CROSS REFERENCE TABLE

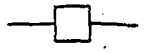
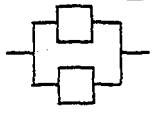
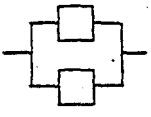
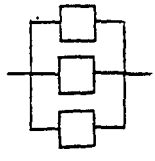
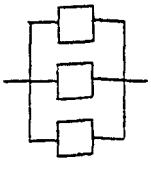
FUNCTION / LEVEL OF REDUNDANCY	BLOCK DIAGRAM	CRITICALITY CATEGORY	
		FUNCTIONAL DEFINITIONS	HARDWARE DEFINITIONS
LIFE OR VEHICLE ESSENTIAL / NO REDUNDANCY		1 (CIL)	1 (CIL)
MISSION ESSENTIAL / NO REDUNDANCY		2 (CIL)	2 (CIL)
LIFE OR VEHICLE ESSENTIAL / DUAL REDUNDANCY		1R (CIL)	2 (CIL)
MISSION ESSENTIAL / DUAL REDUNDANCY		PASSED SCREEN: 2R / FAILED SCREEN: 2R (CIL)	3
LIFE OR VEHICLE ESSENTIAL / TRIPLE REDUNDANCY		1R	1R (CIL)	3
MISSION ESSENTIAL / TRIPLE REDUNDANCY		2R	2R (CIL)	3
ALL NON-ESSENTIAL / ALL LEVELS OF REDUNDANCY		3		3

FIGURE 10

FAILURE EFFECTS

(A) ON FUNCTION OR SUBSYSTEM:
- No Effect
- Loss of Redundancy
- Functional Degradation
- Subsystem Degradation
- Loss of Function
- Loss of Subsystem

(B) ON INTERFACE FUNCTIONS OR SUBSYSTEMS:
- No Effect
- Loss of Interface Redundancy
- Degradation of Interface Function
- Degradation of Interface Subsystem
- Loss of Interface Function
- Loss of Subsystem

(C) ON MISSION:
- No Effect
- See Note Below for Criticality 2 Modes
- Mission Modification
- Loss of Entry Capability - Rescue

(D) ON CREW/VEHICLE:
- No Effect
- Possible Loss of Crew/Vehicle (Specify)
- Probable Loss of Crew/Vehicle (Conditions)
- Loss of Crew/Vehicle

NOTE: The following instruction is intended to clarify what should be entered in the FMEA/CIL under "EFFECTS ON MISSION" (item C under entry J240) for identified criticality 2 failure modes.

Criticality 2 failure (modes) are defined as: (1) single failures which would cause "loss of mission", and (2) failures wherein the next associated failure would cause loss of crew/vehicle (Appendix B, Section 3.1.1, Ground Rules, subparagraph 1).

The following chart (Mission Effects - Criticality 2 Failure Modes) is included as a guideline for entries under "EFFECTS ON MISSION". The term "abort decision" should only be used where there really is a decision.

MISSION EFFECTS - CRITICALITY 2 FAILURE MODES

ENTRIES UNDER "EFFECTS ON MISSION" IN FMEA/CIL FOR CRITICALITY 2 FAILURE MODES	RATIONALE/TYPICAL SITUATIONS AND EXAMPLES (GUIDELINES NOT TO BE ENTERED IN FMEA/CIL)
1. LOSS OF CAPABILITY TO - - - (ADD SPECIFICS OF CAPABILITY LOST DUE TO IDENTIFIED FAILURE).	FAILURE MODE AFFECTS ACCOMPLISHMENT OF MISSION OBJECTIVE(S) ONLY. EXAMPLE: JAMMED GEAR BOX (SFP) WHICH PRECLUDES RELEASE OR DEPLOYMENT OF A PAYLOAD.
2. ABORT DECISION IF FAILURE OCCURS PRIOR TO ENTRY COMMITMENT.	LOSS OF REDUNDANCY WHEREIN NEXT ASSOCIATED FAILURE WILL CAUSE LOSS OF CREW/VEHICLE. EXAMPLES: APU/HYD, ECLSS FUNCTIONS, POWER GENERATION.
3. NONE - REDUNDANT SIMULTANEOUS MEANS AVAILABLE TO - - - (SPECIFY FUNCTION TO BE PERFORMED.)	SIMULTANEOUS USAGE OF REDUNDANT MEANS TO PERFORM A TIME CRITICAL FUNCTION. EXAMPLE: FAILURE OF ONE OF TWO REDUNDANT PYROTECHNIC DEVICES IN THE SEPARATION SYSTEM.
4. CONTINUE MISSION - DUTY CYCLE SAME AS ABORT.	EXAMPLE: LOSS OF ONE MOTOR WHILE OPENING A DOOR WHICH SUBSEQUENTLY MUST BE CLOSED FOR ENTRY.
NOTE: WHEN DECISION IS TIME DEPENDENT:	
5. ABORT AOA OR ATO IF FAILURE OCCURS DURING ASCENT PHASE.	LOSS OF ONE OMS ENGINE PRECLUDES MISSION COMPLETION.
6. ABORT UNLESS MISSION CAN BE COMPLETED WITH ONE SSME OUT.	MISSION 1 CAN BE COMPLETED IF ONE SSME SHUTS DOWN BETWEEN 347.3 AND 401.6 SECONDS OF ASCENT. MISSION 2 HAS NO ENGINE OUT MISSION COMPLETION CAPABILITY.

3.0 GROUND RULES AND CRITERIA

The following ground rules and criteria are of a general category for guidance, as applicable, in conducting and interpreting an FMEA. The applicable ground rules and criteria will be a part of the information which prefaces each FMEA. (See section 3.2, FMEA Content.)

3.1 GENERAL GROUND RULES AND CRITERIA

3.1.1 GROUND RULES

1. Criticality definitions are those delineated in NHB 5300.4 (1D-1), as illustrated in FIGURE 10 (Criticality Category Cross Reference Table). For the purpose of this analysis, hardware criticality 2 is further defined; i.e., dual redundancy where:
   a. The first failure would result in loss of mission.
   b. The next related failure would result in loss of life/vehicle.

2. Criticality 1R and 2R assumes failure of all like and unlike redundancy: A backup item, if when it is called upon to work, performs a function different from the item it is backing up, it should be classified based upon the effect if it does not work when operated. If the backup item performs the same function as the item it is backing up, the backup should be classified as an unlike redundant item.

3. Loss of mission is defined as follows:
   a. Operation payload interface hardware failure as it would result in loss of payload primary performance.
   b. Orbiter subsystem failure as it would result in unplanned mission termination for non-safety of flight reasons.

4. Categorization of a hardware item by the worst case potential effect of failure of that item will define criticality.

5. Failure modes that could propagate to interfacing subsystems or experiments will be identified.

6. When defining FMEA's/CIL for a particular subsystem, interfacing subsystems will be considered to be operating within their specified tolerances.

7. GFE FMEA data will be utilized in evaluating the GFE/Rockwell International interfaces for the vehicle FMEA and CIL.

8. Failure detectability assumes the availability of telemetry or a crewman responding to monitored displays. Failure detectability also assumes other means of failure detection, where feasible, such as a crew response to physical stimuli; i.e., smell, sound, etc.

9. Specific FMEA criteria and assumptions will be defined for each subsystem.

10. Identical components used for different functions will be treated separately in the FMEA.

11. Simultaneous failure of redundant components is identified where the failure cause encompasses both components.

12. Subsystem analysis will include an evaluation of the effects of instrumentation failure upon/within the subsystem.

13. External Leakage:
    a. The external leakage mode of functional hardware items from any source (except mating of two surfaces by welding, brazing, or permaswage) will be considered. If this mode raises the criticality of the items in question, it will be documented and the potential leak source identified under "CAUSE(S)". Otherwise, the external leakage will be treated generically by media. However, in those instances where external leakage results in a hardware criticality 1 effect, the failure mode will be documented regardless of the basic criticality of the item being considered. Where applicable, seal failure should be listed as a cause and worst case (complete seal failure) shall be assumed, considering

also any restrictive protection provided by barrier design, where such data are available from Design. Hazards associated with the loss of fluid in excess of requirements will be documented and covered by Hazard Analysis, but will not affect criticality (see section 3.3, paragraph 4.)

b. The internal leak mode of functional hardware items will be considered. In those instances where internal leakage could result in a hardware criticality 1 or 2 failure mode effect (fail open or fail closed due to pressure lockup), "internal leakage" shall be entered in the "CAUSE" section of the most appropriate identified failure mode entry in lieu of "cause" is acceptable.

c. Where external or internal leak paths are protected by static or dynamic redundant (verifiable) seals, the leak path effect will be reduced by one criticality level.

d. Pressure carriers (lines, pressure vessels) will be classified by worst case mode including external leakage. Lines will be entered generically for each independent media. Special lines (i.e., mechanical bellows, flex lines, etc.) will be entered individually. Tanks will be entered individually.

14. The failure of any tank containing fluid media which, due to its location in an enclosed vehicle compartment, could cause compartment overpressurization leading to structural failure (vehicle loss) will be classified hardware criticality 1 for tank rupture mode.

15. All lines will be designated the criticality applicable to the functional loss effect resulting from loss of medium with notation in the "REMARKS" section of the FMEA as to the potential hazard due to compartment overpressurization resulting from line rupture. The main engine cryogenic feedlines will be treated as fluid tanks.

3.1.2 CRITERIA

1. The FMEA and CIL will consider failures beginning with preflight/prelaunch operations through post landing safings at Edwards Air Force Base/Kennedy Space Center (EAFB/KSC).
   a. Prelaunch operations at KSC/VAFB are defined as beginning with propellant loading for each specific propulsion subsystem. For all other subsystems, prelaunch operations commence with start of main engine conditioning.

   b. Post landing safing operations include those activities performed after landing to prepare the orbiter for hangar operations and are defined as follows:
      (1) Deservice and draining of hazardous fluids.
      (2) Safing of unused ordnance.
      (3) Application of ground power and cooling.
      (4) Removal of potentially hazardous components.
      (5) Removal of pods and payloads.
      (6) Purging and venting of gases.
      (7) Installation of protective covers.

2. Redundancy is defined as the use of more than one means of accomplishing a given task or function where all must fail before there is an overall failure of the function.
   a. Operational Redundancy - redundant elements, all of which are fully energized during the subsystem operating cycle. Operational redundancy includes load sharing redundancy wherein redundant elements are connected in such a manner that, upon failure of one unit, the remaining redundant elements will continue to perform the subsystem function. It is not necessary to switch out the failed element nor to switch in the redundant element.

   b. Standby Redundancy - redundant elements that are non-operative (i.e., have no power applied) until they are switched into the subsystem upon failuare of the primary element. In these cases, as well as pyrotechnic devices, mechanical linkage and inoperative

functional paths of any subsystem, redundancy screen B is considered not applicable and so marked. This approach is based on the fact that these areas are subject to ground checkout; they are redundant and therefore provide a degree of protection in flight; it is confirmed that when called upon to operate, the inability to operate would be detected in flight for appropriate corrective action.

3. Where redundancy exists in the subsystem, the redundancy is considered during the analysis of a failure of the component.

4. "Alternate means of operation" refers to accomplishment of a function and not necessarily to redundancy or restoration of a failed function.

5. When fire hazards resulting from short circuits or other hardware failure modes are identified, consideration will be given to the effect of fire propagation to adjacent redundant equipment as a potential loss of the function.

   Potential safety concerns created by component failure modes will be identified and handled through Hazards Analyses as required by EOM 70 1-4.2.5 and by NHB 5300.4 (1D-1).

6. Reference documents in the FMEA include released and controlled engineering drawings or specifications, when available.

7. The following are used as aids in determining the failure modes and causes of subsystem hardware failures:

   a. Generic failure modes and causes.
   b. Released and controlled component, assembly, and detail engineering drawings and specifications.
   c. Training aids, as available; e.g., cross section drawings, photographs, exploded drawings (not referenced in FMEA).
   d. Actual hardware, if available.
   e. Use experience, including failure history and similar components.

f. Controlled and released operational procedures.
g. Component FMEA's prepared by component suppliers.

8. Failure of structural items (primary and secondary) will not be considered as a part of this ananysis. (Structural items are assumed to be designed to preclude failure by use of adequate design safety factors.)

9. FMEA's of criticality 1 or 2 "black boxes" providing electrical signal interface to mechanical components are included in the applicable avionics FMEA package with appropriate cross-referencing in the "REFERENCE DOCUMENTS" section of both the appropriate avionics and mechanical FMEA report, where available.

10. The failure mode "Fails to Operate" will not be addressed for fuses. Use of a fuse with a higher current capacity then specified (wrong size installed or rating misidentified) is not considered a fuse failure mode.

VII-A

Designation: D 395 - 85

AMERICAN SOCIETY FOR TESTING AND MATERIALS
1916 Race St., Philadelphia, Pa. 19103
Reprinted from the Annual Book of ASTM Standards, Copyright ASTM
If not listed in the current combined index, will appear in the next edition.

Standard Test Methods for RUBBER PROPERTY—COMPRESSION SET[1]

This standard is issued under the fixed designation D 395; the number immediately following the designation indicates the year of original adoption or, in the case of revision, the year of last revision. A number in parentheses indicates the year of last reapproval. A superscript epsilon (ε) indicates an editorial change since the last revision or reapproval.

These methods have been approved for use by agencies of the Department of Defense and for listing in the DoD Index of Specifications and Standards.

1. Scope

1.1 These test methods cover the testing of rubber intended for use in applications in which the rubber will be subjected to compressive stresses in air or liquid media. They are applicable particularly to the rubber used in machinery mountings, vibration dampers, and seals. Two methods are covered as follows:

Method	Section
A—Compression Set Under Constant Force in Air	7–10
B—Compression Set Under Constant Deflection in Air	11–14

1.2 The choice of method is optional, but consideration should be given to the nature of the service for which correlation of test results may be sought. Unless otherwise stated in a detailed specification, Method B shall be used.

1.3 Method B is not suitable for vulcanizates harder than 90 IRHD.

1.4 The values stated in SI units are to be regarded as the standard.

1.5 *This standard may involve hazardous materials, operations, and equipment. This standard does not purport to address all of the safety problems associated with its use. It is the responsibility of whoever uses this standard to consult and establish appropriate safety and health practices and determine the applicability of regulatory limitations prior to use.*

2. Applicable Documents

2.1 *ASTM Standards:*

D 1349 Practice for Rubber—Standard Temperatures and Atmospheres for Testing and Conditioning[2]

D 3040 Practice for Preparing Precision Statements for Standards Related to Rubber and Rubber Testing[2]

D 3182 Practice for Rubber—Materials, Equipment, and Procedures for Mixing Standard Compounds and Preparing Standard Vulcanized Sheets[3]

D 3183 Practice for Rubber—Preparation of Pieces for Test Purposes from Products[3]

D 3767 Practice for Rubber—Measurement of Dimensions[3]

E 145 Specification for Gravity-Convection and Forced-Ventilation Ovens[4]

NOTE 1—The specific dated edition of Practice D 3040 that prevails in this document is referenced in the Precision section.

3. Summary of Methods

3.1 A test specimen is compressed to either a deflection or by a specified force and maintained under this condition for a specified time and at a specified temperature.

3.2 The residual deformation of a test specimen is measured 30 min after removal from a suitable compression device in which the specimen had been subjected for a definite time to compressive deformation under specified conditions.

3.3 After the measurement of the residual deformation, the compression set as specified in the

---

[1] These test methods are under the jurisdiction of ASTM Committee D-11 on Rubber and are the direct responsibility of Subcommittee D11.10 on Physical Testing.

Current edition approved Sept. 27, 1985. Published November 1985. Originally published as D 395 - 34. Last previous edition D 395 - 84.

[2] *Annual Book of ASTM Standards*, Vols 09.01 and 09.02.

[3] *Annual Book of ASTM Standards*, Vol 09.01.

[4] *Annual Book of ASTM Standards*, Vol 14.02.

appropriate method, is calculated according to Eqs (1) and (2).

4. Significance and Use

4.1 Compression set tests are intended to measure the ability of rubber compounds to retain elastic properties after prolonged action of compressive stresses. The actual stressing service may involve the maintenance of a definite deflection, the constant application of a known force, or the rapidly repeated deformation and recovery resulting from intermittent compressive forces. Though the latter dynamic stressing, like the others, produces compression set, its effects as a whole are simulated more closely by compression flexing or hystersis tests. Therefore, compression set tests are considered to be mainly applicable to service conditions involving static stresses. Tests are frequently conducted at elevated temperatures.

5. Test Specimens

5.1 Specimens from each sample may be tested in duplicate (Option 1) or triplicate (Option 2). The compression set of the sample in Option 1 shall be the average of the two specimens, expressed as a percentage. The compression set of the sample in Option 2 shall be the median (middle most value) of the three specimens expressed as a percentage.

5.2 The standard test specimen shall be a cylindrical disk cut from a laboratory prepared slab.

5.2.1 The dimensions of the standard specimens shall be:

Type	1[A]	2[B]
Thickness, mm (in.)	12.5 ± 0.5 (0.49 ± 0.02)	6.0 ± 0.2 (0.24 ± 0.01)
Diameter, mm (in.)	29.0 ± 0.5 (1.14 ± 0.02)	13.0 ± 0.2 (0.51 ± 0.01)

[A] Type 1 specimen is used in Methods A and B.
[B] Type 2 specimen is used in Method B.

5.2.2 When cutting the standard specimen, the circular die having the required inside dimensions specified in 5.2.1 shall be rotated in a drill press or similar device and lubricated by means of a soap solution. A minimum distance of 13 mm (0.51 in.) shall be maintained between the cutting edge of the die and the edge of the slab. The cutting pressure shall be as light as possible to minimize cupping of the cut edges. The dies shall be maintained carefully so that the cutting edges are sharp and free of nicks.

5.3 An optional method of preparing the standard specimen may be the direct molding of a circular disk having the dimensions required for the method used and specified in 5.2.1.

NOTE 2—It should be recognized that an equal time and temperature, if used for both the slab and molded specimen, will not produce an equivalent state of cure in the two types of specimen. A higher degree of cure will be obtained in the molded specimen. Adjustments, preferably in the time of cure, must be taken into consideration if comparisons between the specimens prepared by different methods are to be considered valid.

NOTE 3—It is suggested, for the purpose of uniformity and closer tolerances in the molded specimen, that the dimensions of the mold be specified and shrinkage compensated for therein. A two-plate mold with a cavity 13.0 ± 0.1 mm (0.510 ± 0.004 in.) in thickness and 29.20 ± 0.05 mm (1.148 ± 0.002 in.) in diameter, with overflow grooves will provide Type 1 specimens for Method A and Method B. A similar mold but having a cavity of 6.3 ± 0.3 mm (0.25 ± 0.012 in.) in thickness and 13.2 ± 0.1 mm (0.52 ± 0.004 in.) in diameter will provide Type 2 specimens for Method B.

5.4 When the standard test specimen is to be replaced by a specimen taken from a vulcanized rubber part of greater thickness than the one indicated in 5.2.1, the sample thickness shall be reduced first by cutting transversely with a sharp knife and then followed by buffing to the required thickness in accordance with Practice D 3183.

5.5 An alternative method of preparing specimens is by plying up cylindrical disks cut from a standard sheet prepared in accordance with Practice D 3182 using the specimen sizes specified in 5.2.1 and cutting as described in 5.2.2.

5.5.1 The disks shall be plied, without cementing, to the thickness required. Such plies shall be smooth, flat, of uniform thickness, and shall not exceed seven in number for Type 1 specimens and four in number for Type 2 specimens.

5.5.2 Care shall be taken during handling and placing of the plied test specimen in the test fixture by keeping the circular faces parallel and at right angles to the axis of the cylinder.

5.5.3 The results obtained on plied specimens may be different from those obtained using solid specimens and the results may be variable, particularly if air is trapped between disks.

5.5.4 The results obtained on the specimens prepared by one of the methods may be compared only to those prepared by the same method.

5.6 For routine or product specification test-

ing, it is sometimes more convenient to prepare specimens of a different size or shape, or both. When such specimens are used, the results should be compared only with those obtained from specimens of similar size and shape and not with those obtained with standard specimen. For such cases, the product specification should define the specimen as to the size and shape. If suitable specimens cannot be prepared from the product, the test method and allowable limits must be agreed upon between the producer and the purchaser.

6. Conditioning

6.1 Store all vulcanized test specimens or product samples to be tested at least 24 h but not more than 60 days. When the date of vulcanization is not known, make tests within 60 days after delivery by the producer of the article represented by the specimen.

6.2 Allow buffed specimens to rest at least 30 min before specimens are cut for testing.

6.3 Condition all specimens before testing for a minimum of 3 h at 23 ± 2°C (73.4 ± 3.6°F). Specimens whose compression set properties are affected by atmospheric moisture, shall be conditioned for a minimum of 24 h in an atmosphere controlled to 50 ± 5 % relative humidity.

METHOD A—COMPRESSION SET UNDER CONSTANT FORCE IN AIR

7. Apparatus

7.1 *Dial Micrometer*—A dial micrometer, for measuring specimen thickness, in accordance with Practice 3767, Method A1.

7.2 *Compression Device,* consisting of a force application spring and two parallel compression plates assembled by means of a frame or threaded bolt in such a manner that the device shall be portable and self-contained after the force has been applied and that the parallelism of the plates shall be maintained. The force may be applied in accordance with either 7.2.1 or 7.2.2.

7.2.1 *Calibrated Spring Force Application*—The required force shall be applied by a screw mechanism for compressing a calibrated spring the proper amount. The spring shall be of properly heat-treated spring steel with ends ground and perpendicular to the longitudinal axis of the spring. A suitable compression device is shown in Fig. 1. The spring shall conform to the following requirements:

7.2.1.1 The spring shall be calibrated at room temperature 23 ± 5°C (73.4 ± 9°F) by applying successive increments of force not exceeding 250 N (50 lbf) and measuring the corresponding deflection to the nearest 0.2 mm (0.01 in.). The curve obtained by plotting the forces against the corresponding deflections shall have a slope of 70 ± 3.5 kN/m (400 ± 20 lbf/in.) at 1.8 kN (400 lbf). The slope is obtained by dividing the two forces above and below 1.8 kN by the difference between the corresponding deflections.

7.2.1.2 The original dimensions of the spring shall not change due to fatigue by more than 0.3 mm (0.01 in.) after it has been mounted in the compression device, compressed under a force of 1.8 kN (400 lbf), and heated in the oven for one week at 70°C ± 2°C (158 ± 3.6°F). In ordinary use, a weekly check of the dimensions shall show no greater change than this over a period of 1 year.

7.2.1.3 The minimum force required to close the spring (solid) shall be 2.4 kN (530 lbf).

7.2.2 *External Force Application*—The required force shall be applied to the compression plates and spring by external means after the test specimen is mounted in the apparatus. Either a calibrated compression machine or known masses may be used for force application. Provision shall be made by the use of bolts and nuts or other devices to prevent the specimen and spring from losing their initial deflections when the external force is removed. The spring shall have essentially the same characteristics as described in 7.2.1, but calibration is not required. A suitable compression device is shown in Fig. 2.

7.3 *Plates*—The plates between which the test specimen is compressed shall be made of steel of sufficient thickness to withstand the compressive stresses without bending. The surfaces against which the specimen is held shall have a highly polished chromium-plated finish and shall be cleaned thoroughly and wiped dry before each test.

7.4 *Oven,* conforming to the specification for a Type IIB laboratory oven given in Specification E 145.

8. Procedure

8.1 *Original Thickness Measurement*—Measure the original thickness of the specimen to the nearest 0.02 mm (0.001 in.). Place the specimen

on the anvil of the dial micrometer so that the presser foot will indicate the thickness at the central portion of the top and bottom faces.

8.2 *Application of Compressive Force*—Assemble the specimens in the compression device, using extreme care to place them exactly in the center between the plates to avoid tilting. If the calibrated spring device (Fig. 1) is used, apply the compressive force by tightening the screw until the deflection as read from the scale is equivalent to that shown on the calibration curve for the spring corresponding to a force of 1.8 kN (400 lbf). With the external loading device (Fig. 2), apply this force to the asembly in the compression machine or by adding required masses, but in the latter case, take care to add the mass gradually without shock. Tighten the nuts and bolts just sufficiently to hold the initial deflections of the specimen and spring. It is imperative that no additional force be applied in tightening the bolts.

8.3 *Test Time and Test Temperature*—Choose a suitable temperature and time for the compression set, depending upon the conditions of the expected service. In comparative tests, use identical temperature and heating periods. It is suggested that the test temperature be chosen from those listed in Practice D 1349. Suggested test periods are 22 h and 70 h. The specimen shall be at room temperature when inserted in the compression device. Place the assembled compression device in the oven within 2 h after completion of the assembly and allow it to remain there for the required test period in dry air at the test temperature selected. At the end of the test period, take the device from the oven and remove the specimens immediately and allow it to cool.

8.4 *Cooling Period*—While cooling, allow the specimens to rest on a poor thermally conducting surface, such as wood, for 30 min before making the measurement of the final thickness. Conduct the cooling period at a standard laboratory temperature of 23 ± 2°C (73.4 ± 3.6°F). Specimens whose compression set property is affected by atmospheric moisture shall be cooled in an atmosphere controlled to 50 ± 5 % relative humidity.

8.5 *Final Thickness Measurement*—After the rest period, measure the final thickness at the center of the specimen in accordance with 8.1.

9. Calculation

9.1 Calculate the compression set as a percentage of the original thickness as follows:

$$C_A = [(t_o - t_i)/t_o] \times 100 \quad (1)$$

where:

C_A = compression set (Method A) as a percentage of the original thickness,
t_o = original thickness (8.1), and
t_i = final thickness (8.5).

10. Report

10.1 The report shall include the following:

10.1.1 Original dimensions of the test specimen, including the original thickness, t_o,

10.1.2 Actual compressive force on the specimen as determined from the calibration curve of the spring and spring deflection reading (7.2.1) or as applied by an external force (7.2.2),

10.1.3 Thickness of the test specimen 30 min after removal from the clamp, t_i,

10.1.4 Type of test specimen used, together with the time and temperature of test,

10.1.5 Compression set, expressed as a percentage of the original thickness,

10.1.6 Method used (Method A), and

10.1.7 Number of specimens tested.

METHOD B—COMPRESSION SET UNDER CONSTANT DEFLECTION IN AIR

11. Apparatus

11.1 *Dial Micrometer*—A dial micrometer, for measuring the specimen thickness, in accordance with Practice D 3767, Method A1.

NOTE 4—For vulcanizates having a hardness below 35 IRHD, the force on the presser foot should be reduced to 0.2 ± 0.05 N (0.04 ± 0.01 lbf).

11.2 *Spacer Bars*, to maintain the constant deflection required under Method B.

11.2.1 Spacer bars for Type 1 samples shall have a thickness of 9.5 ± 0.02 mm (0.375 ± 0.001 in.).

11.2.2 Spacer bars for Type 2 samples shall have a thickness of 4.50 ± 0.01 mm (0.1770 ± 0.0005 in.).

11.3 *Compression Device*, consisting of two or more flat steel plates between the parallel faces of which the specimens may be compressed as shown in Fig. 3. Steel spacers for the required percentage of compression given in 12.2 shall be

placed on each side of the rubber specimens to control their thickness while compressed. The steel surfaces contacting the rubber specimens shall be ground to a maximum roughness of 250 μm (10 μin.) and then chromium plated and polished.

11.4 *Oven,* conforming to the specification for a Type IIB laboratory oven given in Specification E 145.

11.5 *Plates*—The plates between which the test specimen is compressed shall be made of steel of sufficient thickness to withstand the compressive stresses without bending. The surfaces against which the specimen is held shall have a highly polished chromium-plated finish and shall be cleaned thoroughly and wiped dry before each test.

12. Procedure

12.1 *Original Thickness Measurement*—Measure the original thickness of the specimen to the nearest 0.02 mm (0.001 in.). Place the specimen on the anvil of the dial micrometer so that the presser foot will indicate the thickness at the central portion of the top and bottom faces.

12.2 *Application of Compressive Force*—Place the test specimen between the plates of the compression device with the spacers on each side, allowing sufficient clearance for the bulging of the rubber when compressed (Fig. 3). Where a lubricant is applied, it shall consist of a thin coating of a lubricant having substantially no action on the rubber. For most purposes, a silicon or fluorosilicon fluid is suitable. Tighten the bolts so that the plates are drawn together uniformly until they are in contact with the spacers. The amount of compression employed shall be approximately 25 %. A suitable mechanical or hydraulic device may be used to facilitate assembling and disassembling the test fixture.

12.3 *Test Time and Temperature*—Choose a suitable temperature and time for the compression set, depending upon the conditions of the expected service. In comparative tests, use identical temperature and test periods. It is suggested that the test temperature be chosen from those listed in Recommended Practice D 1349. Suggested test periods are 22 h and 70 h. The test specimen shall be at room temperature when inserted in the compression device. Place the assembled compression device in the oven within 2 h after completion of the assembly and allow it to remain there for the required test period in dry air at the test temperature selected. At the end of the test period, take the device from the oven and remove the test specimen immediately and allow them to cool.

12.4 *Cooling Period*—While cooling, allow the test specimen to rest on a poor thermally conducting surface, such as wood, for 30 min before making the measurement of the final thickness. Maintain the conditions during the cooling period in accordance with 8.4.

12.5 *Final Thickness Measurement*—After the rest period, measure the final thickness at the center of the test specimen in accordance with 12.1.

13. Calculation

13.1 Calculate the compression set expressed as a percentage of the original deflection as follows:

$$C_B = [(t_o - t_i)/(t_o - t_n)] \times 100 \quad (2)$$

where:

C_B = compression set (Method B) expressed as percentage of the original deflection,
t_o = original thickness of specimen (12.1),
t_i = final thickness of specimen (12.5), and
t_n = thickness of the spacer bar used.

NOTE 5—Lubrication of the operating surfaces of the compression device is optional while giving more reproducible results, lubrication may somewhat alter the compression set values.

14. Report

14.1 The report shall include the following:

14.1.1 Original dimensions of the test specimen including the original thickness, t_o,

14.1.2 Percentage compression of the specimen actually employed,

14.1.3 Thickness of the test specimen 30 min after removal from the clamp, t_i,

14.1.4 Type of test specimen used, together with the time and temperature of test,

14.1.5 Whether or not the surfaces of the compression device are lubricated. If they are, what type lubrication was used,

14.1.6 Compression set, expressed as a percentage of the original deflection,

14.1.7 Method used (Method B), and

14.1.8 Number of specimens tested.

15. Precision[5] and Bias

15.1 These precision statements have been prepared in accordance with Practice D 3040-81. Please refer to this practice for terminology and other testing and statistical concepts.

15.2 Prepared test specimens of two rubbers, Methods A and B, were supplied to five laboratories. These were tested in duplicate each day on two separate testing days. A test result, therefore, is the average of two test specimens, for both Methods A and B.

15.3 One laboratory did not run the Method A testing, therefore the precision for Method A is derived from four laboratories.

15.4 The precision results are given in Tables 1 and 2.

15.5 *Bias*—In test method statistical terminology, bias is the difference between an average test value and the reference or true test property value. Reference values do not exist for this test method since the value or level of the test property is exclusively defined by the test method. Bias, therefore, cannot be determined.

[5] Supporting data are available from ASTM Headquarters. Request RR: D-11-1138.

TABLE 1 LQC Precision Data Compression Set—Method A

Material	Mean Level	Within Laboratories		Among Laboratories	
		S	*CV*	*S*	*CV*
A	1.73 (%)	0.0500	0.0277	0.190	0.1096
B	26.1	0.898	0.0336	2.37	0.0908
Average or Pooled Values		0.636	0.0308	1.681	0.1006
		Repeatability		Reproducibility	
Standard Deviation, (*S*)[A]		0.636		1.743	
Coefficient or Variation, (*CV*)		0.0308		0.103	
Least Significant Difference, (*LSD*)[B,C]		8.8 %		29.1 %	

[A] An average value, the value of *S* varies with mean level.
[B] *LSD* based on 95 % confidence level; two results are considered significantly different if their difference, expressed as a percentage of their average, exceeds the stated percent value.
[C] The *LSD* values are relative percent, that is, a percent of the "percent" values used to measure the tested property.

TABLE 2 LQC Precision Data Compression Set—Method B

Material	Mean Level	Within Laboratories		Among Laboratories	
		S	*CV*	*S*	*CV*
A	13.7 (%)	0.591	0.0420	1.543	0.113
B	52.8	0.567	0.0110	5.924	0.112
Average or Pooled Values		0.579	0.0307	4.329	0.1124
		Repeatability		Reproducibility	
Standard Deviation, (*S*)[A]		0.579		4.348	
Coefficient or Variation, (*CV*)		0.0307		0.114	
Least Significant Difference, (*LSD*)[B,C]		8.7 %		32.4 %	

[A] An average value, the value of *S* varies with mean level.
[B] *LSD* based on 95 % confidence level; two results are considered significantly different if their difference, expressed as a percentage of their average, exceeds the stated percent value.
[C] The *LSD* values are relative percent, that is, a percent of the "percent" values used to measure the tested property.

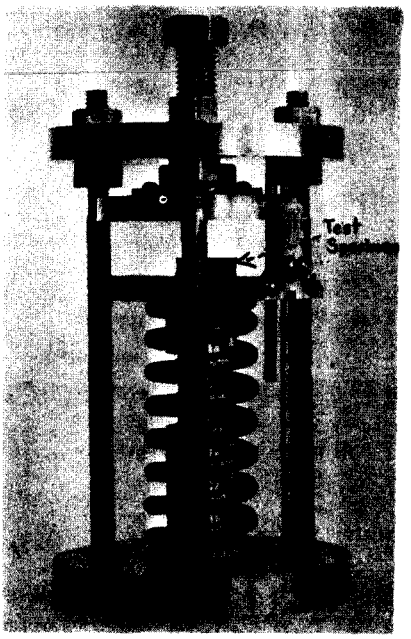


FIG. 1 Device for Compression Set Test, Using Calibrated Spring Loading, Method A

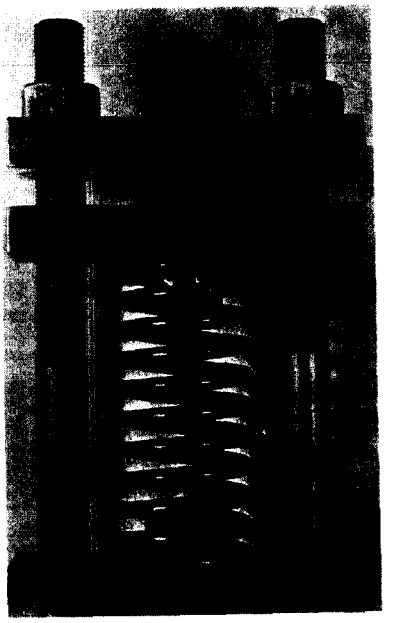


FIG. 2 Device for Compression Set Test, Using External Loading, Method A

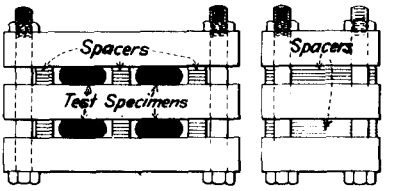


FIG. 3 Device for Compression Set Test Under Constant Deflection, Method B

The American Society for Testing and Materials takes no position respecting the validity of any patent rights asserted in connection with any item mentioned in this standard. Users of this standard are expressly advised that determination of the validity of any such patent rights, and the risk of infringement of such rights, are entirely their own responsibility.

This standard is subject to revision at any time by the responsible technical committee and must be reviewed every five years and if not revised, either reapproved or withdrawn. Your comments are invited either for revision of this standard or for additional standards and should be addressed to ASTM Headquarters. Your comments will receive careful consideration at a meeting of the responsible technical committee, which you may attend. If you feel that your comments have not received a fair hearing you should make your views known to the ASTM Committee on Standards, 1916 Race St., Philadelphia, Pa. 19103.

VII-B

NASA
National Aeronautics and Space Administration
Washington, D.C. 20546

JUL 17 1985

Reply to Attn of MPS

TO: M/Associate Administrator for Space Flight

FROM: MPS/Irv Davids

SUBJECT: Case to Case and Nozzle to Case "O" Ring Seal Erosion Problems

As a result of the problems being incurred during flight on both case to case and nozzle to case "O" ring erosion, Mr. Hamby and I visited MSFC on July 11, 1985, to discuss this issue with both project and S&E personnel. Following are some important factors concerning these problems:

A. Nozzle to Case "O" ring erosion

There have been twelve (12) instances during flight where there have been some primary "O" ring erosion. In one specific case there was also erosion of the secondary "O" ring seal. There were two (2) primary "O" ring seals that were heat affected (no erosion) and two (2) cases in which soot blew by the primary seals.

The prime suspect as the cause for the erosion on the primary "O" ring seals is the type of putty used. It is Thiokol's position that during assembly, leak check, or ignition, a hole can be formed through the putty which initiates "O" ring erosion due to a jetting effect. It is important to note that after STS-10, the manufacturer of the putty went out of business and a new putty manufacturer was contracted. The new putty is believed to be more susceptible to environmental effects such as moisture which makes the putty more tacky.

There are various options being considered such as removal of putty, varying the putty configuration to prevent the jetting effect, use of a putty made by a Canadian manufacturer which includes asbestos, and various combination of putty and grease. Thermal analysis and/or tests are underway to assess these options.

Thiokol is seriously considering the deletion of putty on the QM-5 nozzle/case joint since they believe the putty is the prime cause of the erosion. A decision on this change is planned to be made this week. I have reservations about doing it, considering the significance of the QM-5 firing in qualifying the FWC for flight.

PC 038840

It is important to note that the cause and effect of the putty varies. There are some MSFC personnel who are not convinced that the holes in the putty are the source of the problem but feel that it may be a reverse effect in that the hot gases may be leaking through the seal and causing the hole track in the putty. 10

Considering the fact that there doesn't appear to be a validated resolution as to the effect of putty, I would certainly question the wisdom in removing it on QM-5.

B. Case to Case "O" Ring Erosion

There have been five (5) occurrences during flight where there was primary field joint "O" ring erosion. There was one case where the secondary "O" ring was heat affected with no erosion. The erosion with the field joint primary "O" rings is considered by some to be more critical than the nozzle joint due to the fact that during the pressure build up on the primary "O" ring the unpressurized field joint secondary seal unseats due to joint rotation.

The problem with the unseating of the secondary "O" ring during joint rotation has been known for quite some time. In order to eliminate this problem on the FWC field joints a capture feature was designed which prevents the secondary seal from lifting off. During our discussions on this issue with MSFC, an action was assigned for them to identify the timing associated with the unseating of the secondary "O" ring and the seating of the primary "O" ring during rotation. How long it takes the secondary "O" ring to lift off during rotation and when in the pressure cycle it lifts are key factors in the determination of its criticality.

The present consensus is that if the primary "O" ring seats during ignition, and subsequently fails, the unseated secondary "O" ring will not serve its intended purpose as a redundant seal. However, redundancy does exist during the ignition cycle, which is the most critical time.

It is recommended that we arrange for MSFC to provide an overall briefing to you on the SRM "O" rings, including failure history, current status, and options for correcting the problems.

Irving Davids

cc:
M/Mr. Weeks
M/Mr. Hamby
ML/Mr. Harrington
MP/Mr. Winterhalter

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NAS
August 1985

In Reply Refer to Attn of: SA42-349-85

TO: NASA Headquarters
Attn: Code M/W. E. Hamby

FROM: SA41/L. B. Mulloy

SUBJECT: SRM Joint/O-Ring Erosion

On July 11, 1985 you and Irv Davids were briefed by Jim Thomas of my office on the history and the effort underway to resolve the issues and concerns of the above subject. During this briefing the following information was requested:

1. Question: If the field joint secondary seal lifts off the metal mating surfaces during motor pressurization, how soon will it return to a positon where contact is re-established?

   Answer: Bench test data indicates that the o-ring resiliency (its capability to follow the metal) is a function of temperature and rate of case expansion. MTI measured the force of the o-ring against Instron plattens, which simulated the nominal squeeze on the o-ring and approximated the case expansion distance and rate.

   At 100°F the o-ring maintained contact. At 75°F the o-ring lost contact for 2.4 seconds. At 50°F the o-ring did not re-establish contact in ten minutes at which time the test was terminated.

   The conclusion is that secondary sealing capability in the SRM field joint cannot be guaranteed.

2. Question: If the primary o-ring does not seal, will the secondary seal seat in sufficient time to prevent joint leakage?

   Answer: MTI has no reason to suspect that the primary seal would ever fail after pressure equilibrium is reached, i.e., after the ignition transient. If the

25

primary o-ring were to fail from 0 to 170 milliseconds, there is a very high probability that the secondary o-ring would hold pressure since the case has not expanded appreciably at this point. If the primary seal were to fail from 170 to 330 milliseconds, the probability of the secondary seal holding is reduced. From 330 to 600 milliseconds the chance of the secondary seal holding is small. This is a direct result of the o-ring's slow response compared to the metal-case segments as the joint rotates.

3. Question: Headquarters was not aware that the secondary o-ring may not seat due to joint rotation, when was this incorporated into the FMEA/CIL?

Answer: MTI submitted TWR-13520 "Retention Rationale, SRM Simplex Seal" to MSFC on 12/1/82. The SRB CIL requirement change was approved by Level III CCB on 1/21/83. Level II authorized submittal of Level I change request M22106L on 3/2/83. On 5/2/83 Level II issued FRCBD S22106LR1 to implement approved Level I change request M22106L and M22106M.

L. B. Mulloy
Manager, SRB Project

cc:
SA41/Messrs. Wear/Thomas

MAR 12 '86 12:11 NASA HQ WASH. D.C. PAGE.04

NASA
Space
Flight

SPO.PD. 710.5A
DATE 9/26/83

IN EFFECT FOR
STS 51-L

PROGRAM DIRECTIVE

Responsible Office: SF/Space Shuttle Operations, Integration Division

SUBJECT: Space Shuttle Flight Readiness Reviews

1. PURPOSE

This Directive defines the responsibilities, requirements, and procedures to insure effective planning for and conduct of Flight Readiness Reviews for the Space Transportation System (STS) missions.

2. SCOPE

This Directive is applicable to all STS missions.

3. POLICY

It is the policy of the Associate Administrator for Space Flight (AA-SF) to make an assessment of mission readiness prior to each flight. This will be accomplished by a consolidated Flight Readiness Review (FRR) of all activities/elements necessary for safe and successful conduct of the launch, flight, and post-landing operations. The review will be supported by the NASA Chief Engineer and the Center Directors from JSC, MSFC, and KSC.

The FRR will be preceded by detailed readiness reviews (pre-FRR's) on individual elements, including cargo, under the cognisance of the responsible Managers.

4. RESPONSIBILITIES

a. The conduct of the FRR is the responsibility of the AA-SF or his designated representative.

b. The Director, Space Shuttle Operations, is responsible for FRR planning and requirements, coordinating the FRR agenda, FRR action items and action item closeouts, and preparing the readiness assessment.

c. The JSC Level II National Space Transportation Systems (NSTS) Program Manager is responsible for implementing readiness review plans and requirements, organizing the FRR, and for the program assessment of flight readiness.

d. The Project/Element Managers will conduct pre-FRR's to develop their readiness assessment and are responsible for the FRR briefing content

Att 1-1

45

in their particular area.

5. REVIEW REQUIREMENTS

a. Review Concept: The FRR will employ a delta review concept from prior reviews (eg. Acceptance, Orbiter Rollout/Pre-Stack, Cargo Readiness) and previous STS missions. The review will be conducted by telecon using the NASA Teleconference Network.

b. Schedule: The FRR will be held between one and two weeks prior to launch.

c. Agenda: The major agenda items and responsibilities are:

(1) Introduction ——— Headquarters/AA-SF

(2) Integrated System ——— JSC/Manager, MCCS Program

(3) Orbiter/Crew CPT ——— JSC/Manager, Space Shuttle Projects

(4) SSME/ET/SRB ——— MSFC/Manager, Space Shuttle Projects

(5) Cargo ——— JSC/Manager, Mission Integration

(6) Launch & Landing Operations ——— KSC/Director, Shuttle Projects Management

(7) Flight Operations ——— JSC/Director, Mission Operations

(8) Safety, Reliability & Quality Assurance ——— JSC/SR & QA Manager, Space Shuttle

(9) Action Items/Readiness Poll ——— Headquarters/AA-SF

d. Presentation Emphasis: The presentation of agenda items will normally include a brief status summary with appropriate supporting detail on significant items and conclude with a readiness assessment. The presentation topics and scope should be developed from the pre-FRR's and should:

(1) be that required to provide the AA-SF with the information needed to make a judgment as to flight readiness;

(2) review recent significant resolved problems and prior flight anomalies when necessary to establish confidence;

(3) cover all problems, open items, and constraints remaining to be resolved before the mission;

(4) establish the mission baseline configuration in terms of all significant changes since the last STS mission (changes to be considered include hardware, software, vehicle servicing/checkout, launch commit criteria, flight plans,

flight rules, and crew procedures);

Within the above guidelines, the scope of the review should cover status and issues in areas such as: vehicle checkout, shortages and open work, unexplained anomalies, hardware failures, prior flight anomalies, certification/verification, as-built hardware configuration versus certified hardware list, Critical Item List (CIL), development, qualification, and reliability testing, waivers and deviations, limited life components, launch critical spares, sneak circuits, system safety/hazards, and flight margins. In addition, the review should also cover readiness of ground systems/facilities (e.g., launch site, landing sites, network, Mission Control Center), operations teams (e.g., launch, recovery, flight crew, flight control), and support operations (e.g., DOD, contingency, weather, medical, security).

6. DOCUMENTATION REQUIREMENTS

a. Presentations: Vugraphs will be used for FRR presentations, and a paper copy will be provided to the reviewing officials. A copy of the presentation material shall be provided to telecon locations 24 hours in advance of the review.

b. Certificate of Flight Readiness (CoFR): CoFR endorsements, including a complete listing of exceptions, will be executed and submitted at the conclusion of the review (Reference: Space Shuttle Program Procedure for Certification of Flight Readiness (CoFR), JSC 08117, current revision).

c. FRR Assessment: A FRR assessment letter will be issued by OSF within two working days after the review and will include the following:

(1) Action items;

(2) Significant decisions;

(3) CoFR exceptions and endorsement summary.

7. FRR ACTION ITEM/OPEN ITEM CLOSEOUT REPORTING REQUIREMENTS

a. Status: The NSTS Program Manager will report periodically on the closeout status of action items/open items identified at the FRR. This may be accomplished by the Daily Special Level II PRCB.

Significant items occurring subsequent to the FRR will also be reported to the AA-ST. Actions that can be easily accomplished without safety, mission, or launch impact and do not violate flight vehicle or launch complex configuration integrity or cause basic changes to launch commit criteria, flight rules, flight plan, or abort and alternate mission plans, need not be reported.

b. Action Item Closeouts: Closeouts to FRR action items will be submitted to the AA-ST by Level III/II in writing or TWX and will state fully the basis for closeout, that is: action taken, results

contained, and determinations made. An action will not be closed until signoff is completed by the AA-SF or his designated representative.

c. GSFC Exceptions: The resolution of all exceptions identified in the GSFC endorsement will be completed by two days prior to launch, and a copy of the endorsement will be submitted to the Director, Space Shuttle Operations.

8. RESPONSIBILITIES

a. Single Points of Contact: FRR planning and procedures will be coordinated through single points of contact established at the following locations:

Headquarters - Space Shuttle Operations, Integration Division

JSC - Level II NSTS Program Office

MSFC - Space Shuttle Projects Office

KSC - Space Shuttle Projects Management Directorate

Individuals will be designated by the responsible organizations and the names provided to the Director, Space Shuttle Operations. Duties will be defined by separate correspondence issued by the Director, Space Shuttle Operations.

b. Guidelines: Approximately three weeks prior to the FRR, guidelines will be issued by the Director, Space Shuttle Operations, establishing the review date and any special requirements not covered by this directive.

c. FRR Agenda: Proposed agenda items/times will be submitted to Level I/II by the single points of contact at the respective Centers two weeks prior to the FRR. Responsible Space Shuttle Operations Directors will coordinate detailed presentation items with their Project and will accept items determined to be completed and not requiring the AA-SF review at the FRR. The final FRR agenda will be coordinated by the Director, Space Shuttle Operations, and issued about one week prior to the FRR.

9. ACTION

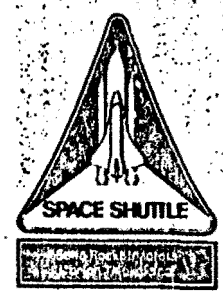
This Directive supersedes STS-PD 710.5 dated May 13, 1981, and shall be implemented by the JSC/NSTS Program Manager effective STS-9 and subsequent missions.

JAMES A. ABRAHAMSON
Lieutenant General, USAF
Associate Administrator for
Space Flight

PC 056353

TWR-15380

STS-51L (STS-33) SOLID ROCKET MOTOR (SRM-25)
FLIGHT READINESS REVIEW



11 DECEMBER 1985

MORTON THIOKOL, INC.

WASATCH DIVISION
P.O. BOX 524, BRIGHAM CITY, UTAH 84302 (801) 863-3511

AGENDA

SECTION		
1.0	0	STS-61C (STS-32) (SRM-24) PERFORMANCE
2.0	0	PROBLEM SUMMARY
3.0	0	STS-51L (STS-33) (SRM-25) CHANGES
	0	CHANGE SUMMARY
	0	DESIGN
	0	HARDWARE CHANGEOUT
	0	PROCESS CHANGES
	0	FIRST TIME CONFIGURATION REUSED
	0	OMRSD
	0	OMD
4.0	0	CERTIFICATION/VERIFICATION STATUS
5.0	0	STS-51L (STS-33) (SRM-25) PERFORMANCE PREDICTIONS
6.0	0	TECHNICAL ISSUES
7.0	0	FLIGHT READINESS REVIEW

STS-61C (STS( ) (SRM-24)
PERFORMANCE
(CHART NO. 1-1)

PROBLEM	CONCERN	RESOLUTION	TO BE DISCUSSED
TBD			

Organizational Issuance

NASA
National Aeronautics and Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

Effective Date: December 29, 1983	Number: SOP 8000.1
Organization: Shuttle Projects Office	

Subject: SHUTTLE PROJECT FLIGHT READINESS REVIEW

1. PURPOSE

This procedure defines the responsibilities, requirements, and procedures to insure effective planning for and conduct of the Shuttle Projects Flight Readiness Review.

2. SCOPE

This procedure is applicable to all STS missions.

3. POLICY

a. It is the policy of the Manager, Shuttle Projects to make an assessment of flight readiness of the Shuttle projects prior to each flight. This will be accomplished by a consolidated Flight Readiness Review (FRR) of all MSFC Shuttle Projects Office elements necessary for safe and successful conduct of the launch, flight, and post-landing operations. The review will be supported by all MSFC organizations which participate in MSFC Shuttle Projects activities.

b. The Shuttle Project FRR will be preceded by separate detailed readiness reviews (pre-FRR's) of individual elements by the prime contractor's and the element project offices, under the cognizance of the responsible Managers.

4. RESPONSIBILITIES

a. The conduct of the Shuttle Projects FRR is the responsibility of the Manager, Shuttle Projects or his designated representative.

b. The Program Plans and Management Systems Office is responsible for FRR scheduling, planning and requirements, coordinating the FRR agenda, FRR action items and action item closeouts, and preparing the readiness assessment and maintenance of all records associated therewith. The Program Plans and Management Systems Office will be the focal point with the JSC Level II National Space Transportation System (NSTS) Program Manager and the Level I FRR under the cognizance of the Director, Space Shuttle Operations.

PC 05654

c. The Project Managers will assure the effectivity of a detailed prime contractor pre-FRR.

d. The Project Managers will conduct FRR's to develop their readiness assessment and are responsible for the Shuttle Projects FRR briefing content in their particular area.

5. PRIME CONTRACTOR PRE-FRR

Project Managers will assure that each prime contractor conducts a pre-FRP in preparation for the Project Offices FRR. The contractors review shall be chaired by a level of management at least one level above the contractor Project Manager.

6. SHUTTLE ELEMENT FRR

a. The Project will conduct a FRR in preparation for the Shuttle Projects FRR. The respective Project Manager of element under review will serve as Chairman. The membership will consist of representatives from the following:

Shuttle Projects Office

S&E Directorate

Reliability and Quality Assurance

Safety

Contractors
USBI, MMC, Thiokol, Rocketdyne

b. Each Project Office will make the necessary conference arrangements, notify review members, designate secretary, prepare presentations, record and track action items, closures and retain a copy of the presentation material in the Project Office record file.

7. SHUTTLE PROJECTS FRR REQUIREMENTS

a. Review Concept: The Shuttle Projects FRR will employ a delta review concept from prior reviews and previous STS missions.

b. Schedule: The Shuttle Projects FRR will be held prior to the Center FRR.

c. Agenda: The major agenda items and responsibilities are:

(1)	Introduction	Programs Plans and Management Systems Office
(2)	Systems	Manager, Systems Management Office
(3)	External Tank	Manager, External Tank Project Office

PC 0056335

74

(4)	Solid Rocket Booster	Manager, Solid Rocket Booster Project Office
(5)	Flight Engine	Manager, Flight Engine Project Office
(6)	Action Items/Readiness Poll	Program Plans and Management Systems Office

d. Presentation Emphasis: The presentation of agenda items will normally include a brief status summary with appropriate supporting detail on significant items and conclude with a readiness assessment. The presentation topics and scope should be developed from the Project FRR's and should:

(1) be that required to provide the Shuttle Projects Manager and Review Team with the information needed to make an independent judgement as to flight readiness;

(2) review recent significant resolved problems and prior flight anomalies when necessary to establish confidence;

(3) cover all problems, technical issues, open items, and constraints remaining to be resolved before the flight;

(4) establish the flight baseline configuration in terms of all significant changes since the last flight and/or applicable STS flight.

Within the above guidelines, the scope of the review should cover status, changes and issues in areas such as:

(1) Hardware/Software Anomalies, Failures including development and acceptance test failures
(2) Launch Commit Criteria
(3) Flight Plans/Rules
(4) Vehicle Checkout
(5) Shortages and Open Work
(6) Prior Flight Anomalies
(7) As-built Hardware Configuration versus Certified Hardware List
(8) Critical Item List (CIL)/Hazards
(9) Development, Qualification, and Reliability Testing (Certification/Verification)
(10) Waivers and Deviations
(11) Limited Life Components
(12) Launch Critical Spares
(13) Sneak Circuits
(14) Flight Margins
(15) PAS Assessment
(16) Safety
(17) Process Changes: Design, manufacturing, checkout and launch processing

PC 056836

e. Shuttle Projects FRR Membership: The Manager, Shuttle Projects will establish review membership and serve as Chairman. Membership will comprise representation from the following organizations:

Program/Project Offices

Science & Engineering Directorate

Reliability & Quality Assurance

Safety

Contractors
USBI, MMC, Thiokol, Rocketdyne

f. Documentation Requirements

(1) Presentations: Vugraphs will be used for FRR presentations and paper copies will be provided to the reviewing officials.

(2) Statement of Flight Readiness: Statement of Flight Readiness will be executed by all Project Managers and submitted at the conclusion of the review.

g. FRR Action Item/Open Item Closeout Reporting Requirements

(1) Subsequent to the conclusion of the Shuttle Projects FRR, a copy of assigned action items will be provided to each actionee by the Program Plans and Management Systems Office.

(2) The FRR secretary will track all action items and provide status to the Shuttle Projects Manager.

(3) Closeouts to FRR action items will be submitted to the Program Plans and Management Systems Office in writing and will state fully the basis for closeout, that is: action taken, results obtained, and determinations made. The Program Plans and Management Systems Office will submit closures to the Manager, Shuttle Projects or his designated representative for signature.

h. Procedures

(1) Single Points of Contact: FRR planning and procedures will be coordinated through a single point of contact in the Program Plans and Management Systems Office.

Individual element for points of contact will be designated by the responsible Project Manager and the names provided to the Program Plans and Management Systems Office. These individuals will be responsible for these duties outlined in paragraph 6.b.

PC 056887

(2) Guidelines: Approximately three weeks prior to the MSFC Shuttle Projects FRR, guidelines will be prepared by Program Plans and Management Systems Office and issued by the Manager, Shuttle Projects, establishing Projects FRR date, Shuttle Projects FRR date, with applicable membership, and any special requirements not covered by this procedure.

8. EFFECTIVE DATE

This procedure is effective on date of issue.

Robert E. Lindstrom
Manager, Shuttle Projects

PC 355884

SOP 8000.1

cc:
SA01/Dr. Lovingood
SA01/Mr. Askew
SA04/Mr. Harden
SA11/Mr. Lombardo
SA31/Mr. Bridwell
SA41/Mr. Mulloy
SA51/Mr. Taylor
SA71/Mr. Boze
SA81/Mr. Zoller
EA01/Mr. Kingsbury
EA01/Dr. Thomson
EE01/Mr. Hardy
EE11/Mr. Horton
EE21/Mr. Thomason
EE31/Mr. Nichols
EE51/Mr. Goetz
EG01/Mr. Brooks
EG03/Mr. Bunn
EB01/Mr. Bradford
EH01/Mr. Schwinghamer
ES01/Dr. Dessler
EP01/Mr. McCool
EP03/Mr. Worlund
EL01/Mr. Hopson
ED01/Dr. McDonough
ET01/Mr. Taylor
PD01/Mr. Marshall
TA01/Mr. Odom
NA01/Mr. Thomas
JA01/Mr. Downey
DA01/Dr. Lucas
DD01/Mr. Lee
DE01/Mr. Bethay
DR01/Mr. Sneed

PC 055869

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

Reply to Attn of DA01

JAN 7 1986

TO: Distribution

FROM: DA01/W. R. Lucas

SUBJECT: MSFC Flight Readiness Review (FRR) Board for MSFC Elements for Mission 51-L

An MSFC FRR Board of senior MSFC management personnel will convene at 8:30 a.m. on January 13, 1986, in Building 4663, HOSC/MCR, to review and assess the readiness status of the MSFC Mission 51-L elements for flight.

This meeting will be held at the SECRET level and all documentation will be handled and presented in accordance with the NASA Security regulations. Attendance will be restricted and all Projects are to coordinate this activity through the Program Planning and Management Systems Office, Tom Staples, 3-0338.

The Center Board is composed of the following:

DA01/W. R. Lucas:	Chairman
DD01/T. J. Lee	Vice Chairman
EA01/J. E. Kingsbury	
DS01/F. A. Speer	
EG01/J. P. Madole	
JA01/J. A. Downey	
PA01/W. R. Marshall	
CS01/J. C. Walker	
CA76/S. G. Henderson:	Secretariat

Each project manager must certify the flight readiness of his hardware and present supporting rationale and data so the Board can independently assess the flight readiness. The Shuttle Projects Office manager is responsible for preparation and coordination of the meeting, presenting an overall assessment of flight readiness, recording of minutes and action items, and tracking action items for closure by the Review Board.

ATTACHMENT 2-9

PC 014983

Emphasis will be placed on safety of flight and mission success, including potential impact of prior flight anomalies; ground test anomalies; revisions to hardware, software, launch commit criteria, or redlines which have not been flight verified; revisions to SRM recovery risks since the previous flight; any waiver which has not been flight verified or which requires external approval; and any revisions to hazard or critical item lists. Issues, concerns, and risks should be clearly identified as well as methods of closure.

In an effort to minimize administrative control requirements associated with the dissemination of classified data, an effort is to be made to present classified information only through viewgraphs. If Projects elect to incorporate classified data within their handouts, it is the Project's responsibility to assure that the handouts are marked and handled in accordance with the NASA Security regulations. If any assistance is needed in this matter, please contact the MSFC Security Division at 3-4310.

A preliminary agenda is enclosed.

W. R. Lucas
Director

Enclosure

Distribution:
See Page 3

PC 014984

STS 51-L

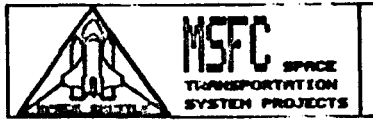
LEVEL I

FLIGHT READINESS REVIEW

ARNOLD D. ALDRICH
JANUARY 15, 1986

ATTACHMENT 2-7

PC 014660



STS 51-L FLIGHT READINESS REVIEW

SOLID ROCKET BOOSTER

L. B. MULLOY
MANAGER, SOLID ROCKET BOOSTER
JANUARY 15, 1986

PC 0147737

AGENDA

- STS 61-C FLIGHT ANOMALIES
- MAJOR ISSUES/PROBLEMS
- MAJOR CONFIGURATION DIFFERENCES
- CERTIFICATION/VERIFICATION STATUS
- READINESS STATEMENT

397

PC 014738

ANOMALIES/PROBLEMS

- NO 61-C FLIGHT ANOMALIES
- NO MAJOR PROBLEMS OR ISSUES

FC 014739

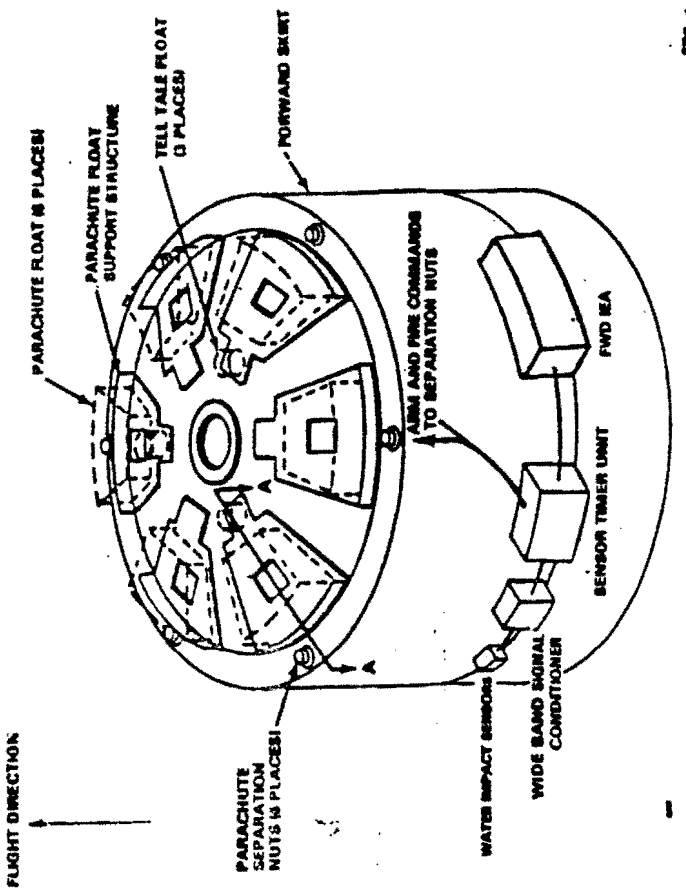
STS 51-L

SIGNIFICANT CONFIGURATION DIFFERENCE

CHANGE	REASON	BASIS FOR CERTIFICATION
IMPLEMENTS CAPABILITY TO SEPARATE SRM NOZZLE AT APOGEE AND TO SEPARATE MAIN PARACHUTES AT WATER IMPACT	0 NOZZLE SEPARATION 0 PRECLUDE FRAGMENTS FROM NOZZLE EXTENSION DESTROYING DROGUE PARACHUTE	0 ANALYSIS - STRUCTURAL - DYNAMIC - FMEA
	0 REDUCE DESIGN LOAD EXCEEDANCES ON MAIN PARACHUTE SYSTEM	0 QUALIFICATION TEST
	0 MAIN PARACHUTE SEPARATION	0 FORMAL PDR/CDR
	0 SRB RETRIEVAL PERSONNEL SAFETY	0 DCR

PC 014740

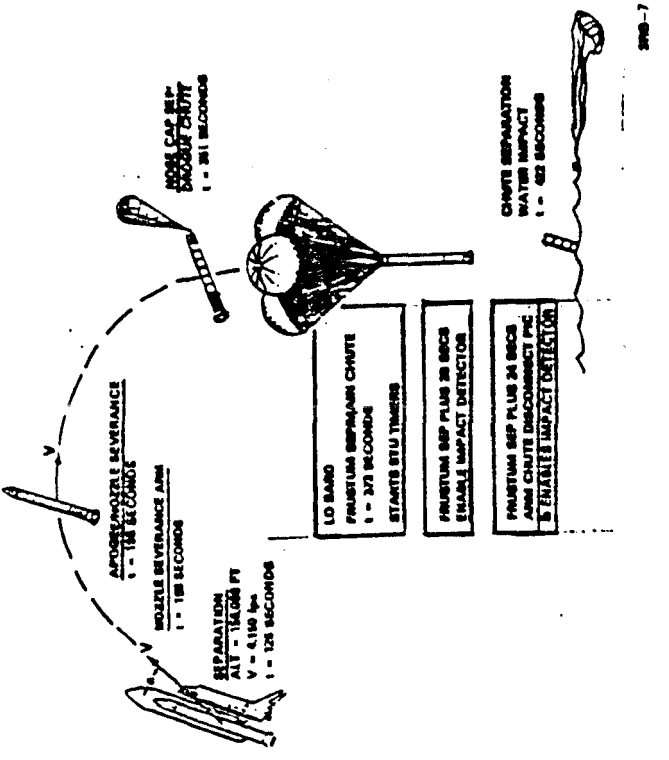
PC 014742



JAN 14 '86 11:16 AL-826/MSFC 4282 345A PAGE.05

PC 014743

NOMINAL SRB REENTRY PROFILE
(136 FOOT MAINS)



JAN 14 '88 11:17 RL-826/MSFC 4282 3439 P.06/.88

SUMMARY OF DESIGN CERTIFICATION

- DESIGN ASSURES SINGLE FAILURE WILL NOT CAUSE PREMATURE SEPARATION EVENTS:

SEQUENCE	PROTECTION
SHUTTLE ENGINE/MOTOR IGNITION	STU NOT POWERED UP
STS BOOST PHASE	STU NOT POWERED UP
SRB SEPARATION	STU NOT POWERED UP
NOZZLE JETTISON	STU POWERED - PARACHUTE PIC'S NOT ARMED WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED
DROGUE DEPLOYMENT	STU POWERED - PARACHUTE PIC'S NOT ARMED, WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED
FRUSTUM SEPARATION	STU POWERED - PARACHUTE PIC'S NOT ARMED WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED

- FMEA VERIFIES
  - TWO OR MORE FAILURES REQUIRED FOR MALFUNCTION
  - ALL PYRO EVENTS OCCUR PRIOR TO ARMING PIC'S FOR PARACHUTE SEPARATION
  - ALL OTHER FUNCTIONS OCCUR PRIOR TO ENABLING SEPARATION CIRCUITS
  - BOTH WATER IMPACT SENSORS MUST INITIATE SEPARATION SIGNAL
- QUALIFICATION TESTS AND SYSTEMS CHECKOUT VERIFY PROPER SYSTEM OPERATION

PC 014744

CERTIFICATION/VERIFICATION STATUS

- 0 NO OPEN CERTIFICATION OF QUALIFICATION
- 0 NO OPEN HAZARDS/CIL'S
- 0 NO OPEN WAIVERS
- 0 NO OPEN UNPLANNED KSC WORK

PC 014745

READINESS STATEMENT

PENDING SATISFACTORY COMPLETION OF NORMAL OPERATIONS FLOW (OMRSD) AND OPEN ITEMS IDENTIFIED, WE CERTIFY THE SRB FLIGHT HARDWARE READY TO SUPPORT MISSION 51-L.

/S/ MR. KILMINSTER	/S/ MR. MURPHY
J. KILMINSTER, MTI VICE PRESIDENT SPACE BOOSTER PROGRAMS	O. MURPHY, USBI-BPC EXECUTIVE VICE PRESIDENT
/S/ MR. SMITH	/S/ MR. MULLOY
J. D. SMITH CHIEF ENGINEER, SRB PROJECT	L. B. MULLOY MANAGER, SRB PROJECT

405

PC 014746

STS 51-L

LEVEL II

FLIGHT READINESS REVIEW

ARNOLD D. ALDRICH
JANUARY 14, 1986

ATTACHMENT 2-10

PC 015543

STS 51-L FLIGHT READINESS REVIEW

SOLID ROCKET BOOSTER

L. B. MULLOY
MANAGER, SOLID ROCKET BOOSTER
JANUARY 15, 1986

PC 015321

AGENDA

- STS 61-C FLIGHT ANOMALIES
- MAJOR ISSUES/PROBLEMS
- MAJOR CONFIGURATION DIFFERENCES
- CERTIFICATION/VERIFICATION STATUS
- READINESS STATEMENT

FC 013082

ANOMALIES/PROBLEMS

- NO 61-C FLIGHT ANOMALIES
- NO MAJOR PROBLEMS OR ISSUES

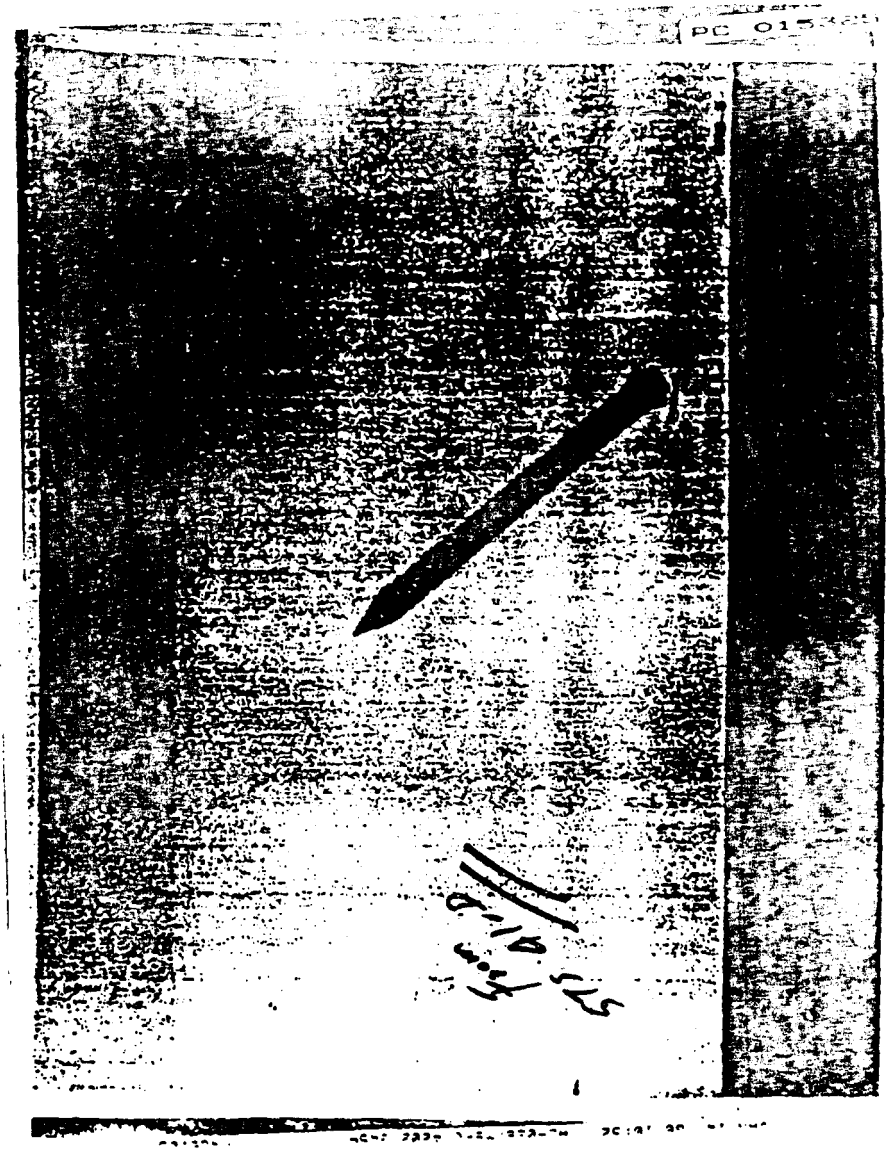
PC 0153.3

STS 51-L

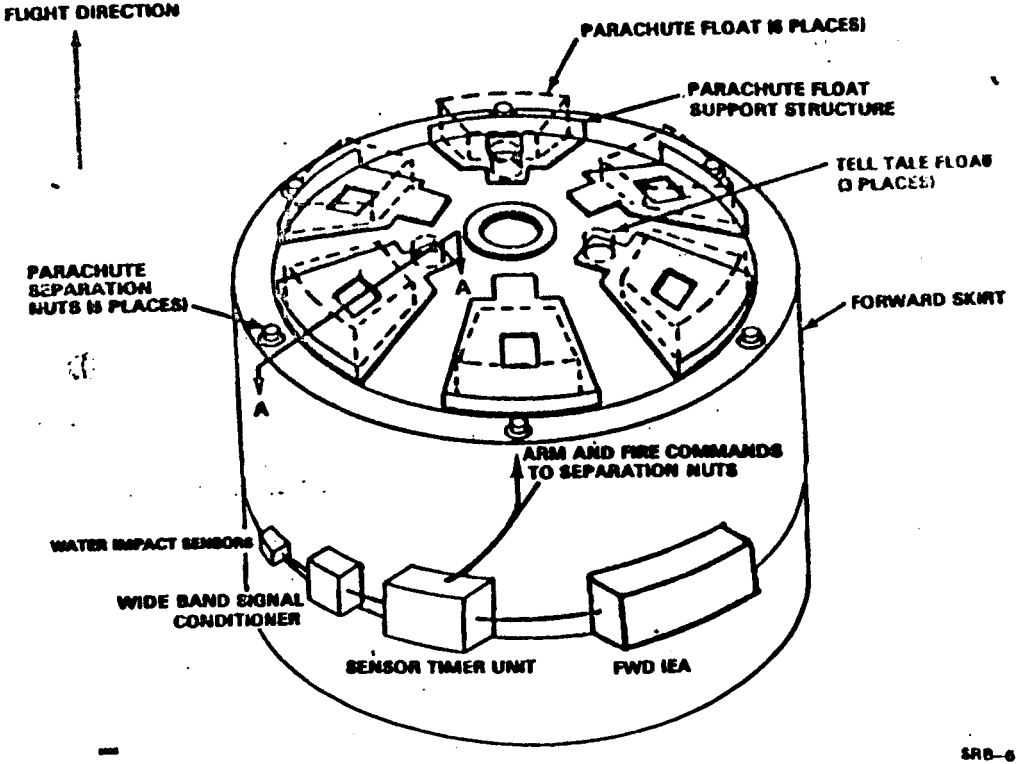
SIGNIFICANT CONFIGURATION DIFFERENCE

CHANGE	REASON	BASIS FOR CERTIFICATION
IMPLEMENTS CAPABILITY TO SEPARATE SRM NOZZLE AT APOGEE AND TO SEPARATE MAIN PARACHUTES AT WATER IMPACT	0 NOZZLE SEPARATION 0 PRECLUDE FRAGMENTS FROM NOZZLE EXTENSION DESTROYING DROGUE PARACHUTE	0 ANALYSIS - STRUCTURAL - DYNAMIC - FMEA
	0 REDUCE DESIGN LOAD EXCEEDANCES ON MAIN PARACHUTE SYSTEM	0 QUALIFICATION TEST
	0 MAIN PARACHUTE SEPARATION	0 FORMAL PDR/CDR
	0 SRB RETRIEVAL PERSONNEL SAFETY	0 DCR

PC 0153324



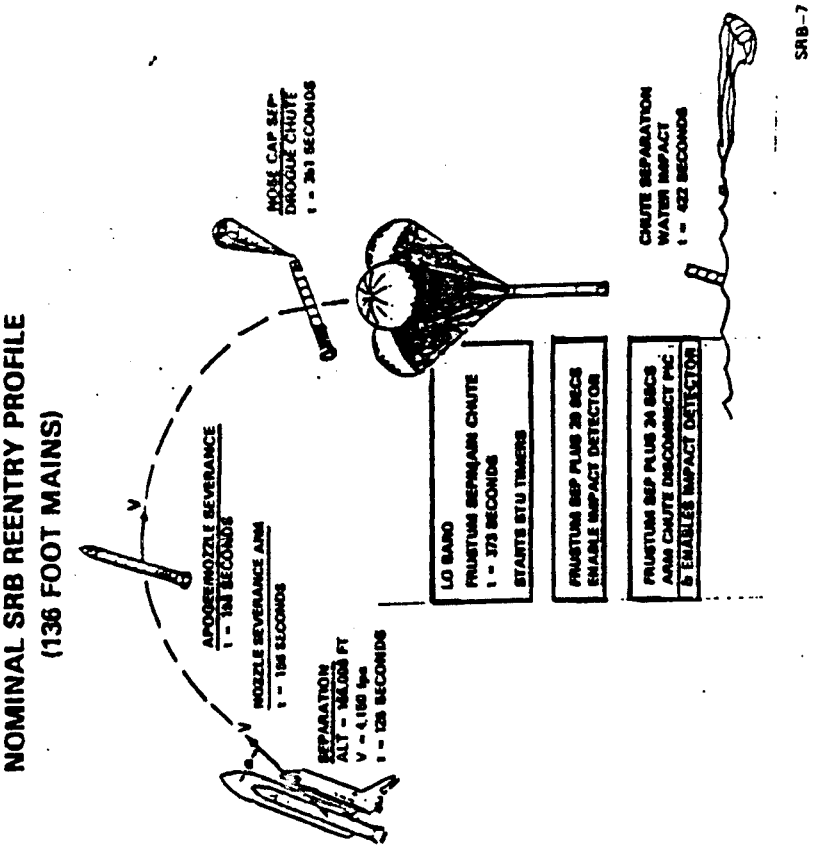
PARACHUTE SEPARATION SYSTEM



SRB-6

PC 015321

PC 015327


NOMINAL SRB REENTRY PROFILE
(136 FOOT MAINS)
SEPARATION
ALT = 184,000 FT
V = 4,150 fps
t = 126 SECONDS
NOZZLE SEVERANCE ARM
t = 156 SECONDS
APOGEE/NOZZLE SEVERANCE
t = 193 SECONDS
NOSE CAP S/P
DROGUE CHUTE
t = 261 SECONDS
LO BARO
FRUSTUM SEPARATION CHUTE
t = 373 SECONDS
STARTS RTU TIMERS
FRUSTUM S/P PLUS 20 SECS
ENABLE IMPACT DETECTOR
FRUSTUM S/P PLUS 24 SECS
ARM CHUTE DISCONNECT PIC
& ENABLES IMPACT DETECTOR
CHUTE SEPARATION
WATER IMPACT
t = 422 SECONDS
SRB-7

SUMMARY OF DESIGN CERTIFICATION

- DESIGN ASSURES SINGLE FAILURE WILL NOT CAUSE PREMATURE SEPARATION EVENTS:

SEQUENCE	PROTECTION
SHUTTLE ENGINE/MOTOR IGNITION	STU NOT POWERED UP
STS BOOST PHASE	STU NOT POWERED UP
SRB SEPARATION	STU NOT POWERED UP
NOZZLE JETTISON	STU POWERED - PARACHUTE PIC'S NOT ARMED WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED
DROGUE DEPLOYMENT	STU POWERED - PARACHUTE PIC'S NOT ARMED, WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED
FRUSTUM SEPARATION	STU POWERED - PARACHUTE PIC'S NOT ARMED WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED

- FMEA VERIFIES
  - TWO OR MORE FAILURES REQUIRED FOR MALFUNCTION
  - ALL PYRO EVENTS OCCUR PRIOR TO ARMING PIC'S FOR PARACHUTE SEPARATION
  - ALL OTHER FUNCTIONS OCCUR PRIOR TO ENABLING SEPARATION CIRCUITS
  - BOTH WATER IMPACT SENSORS MUST INITIATE SEPARATION SIGNAL
- QUALIFICATION TESTS AND SYSTEMS CHECKOUT VERIFY PROPER SYSTEM OPERATION

PC 015321

CERTIFICATION/VERIFICATION STATUS

- 0 NO OPEN CERTIFICATION OF QUALIFICATION
- 0 NO OPEN HAZARDS/CIL'S
- 0 NO OPEN WAIVERS
- 0 NO OPEN UNPLANNED KSC WORK

PC 01532

READINESS STATEMENT

PENDING SATISFACTORY COMPLETION OF NORMAL OPERATIONS FLOW (OMRSD) AND OPEN ITEMS IDENTIFIED, WE CERTIFY THE SRB FLIGHT HARDWARE READY TO SUPPORT MISSION 51-L.

/S/ MR. KILMINSTER	/S/ MR. MURPHY
J. KILMINSTER, MTI VICE PRESIDENT SPACE BOOSTER PROGRAMS	G. MURPHY, USBI-BPC EXECUTIVE VICE PRESIDENT
/S/ MR. SMITH	/S/ MR. MULLOY
J. D. SMITH CHIEF ENGINEER, SRB PROJECT	L. B. MULLOY MANAGER, SRB PROJECT

PC 015330

TWR-15284C

FLIGHT READINESS REVIEW
SRM-24 (STS-61C)
02 DECEMBER 1985
L. O. WEAR

FLIGHT READINESS REVIEW
SRM-24 (STS-61C)

1.0	INTRODUCTION	MSFC
2.0	ELEMENT BOARD REVIEW	MSFC
	0 PROBLEM SUMMARY	
3.0	SRM-24 REVIEW	MTI
	0 STS-61B (SRM-23) PERFORMANCE	
	0 CHANGE SUMMARY	
	0 PERFORMANCE PREDICTION	
	0 CERTIFICATION/VERIFICATION	
	0 PROBLEM SUMMARY TOPICS	
	0 FLIGHT READINESS CERTIFICATION	

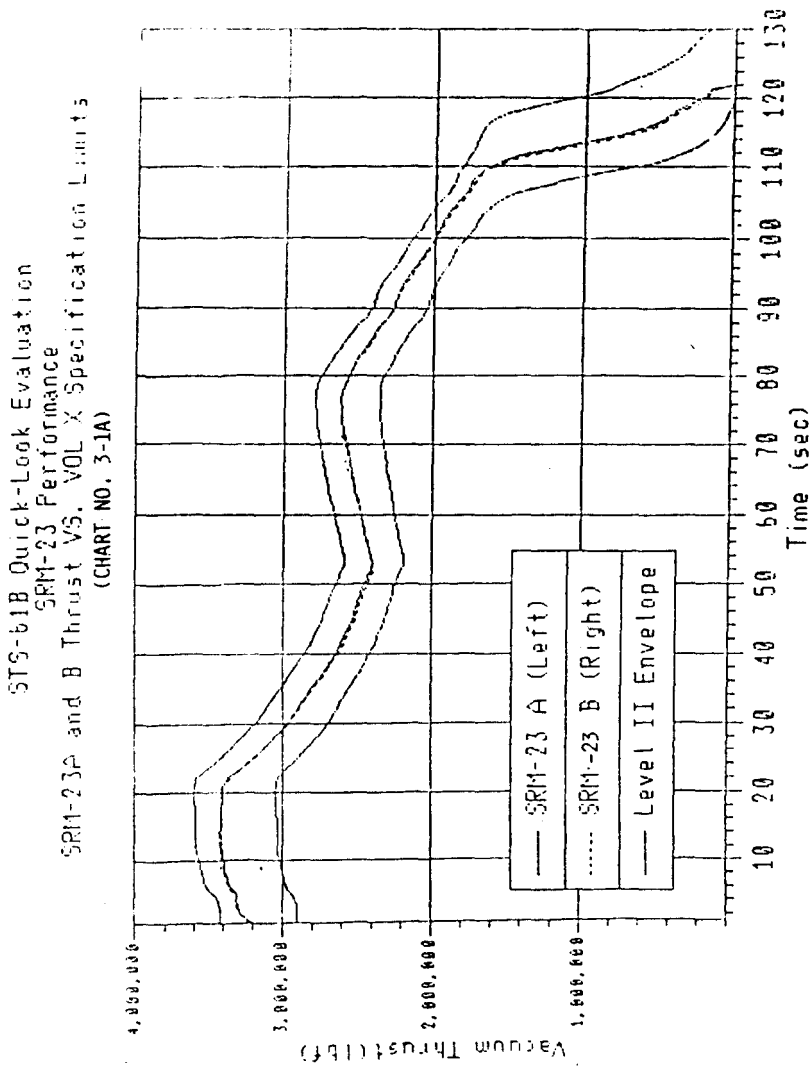
PC 0655503

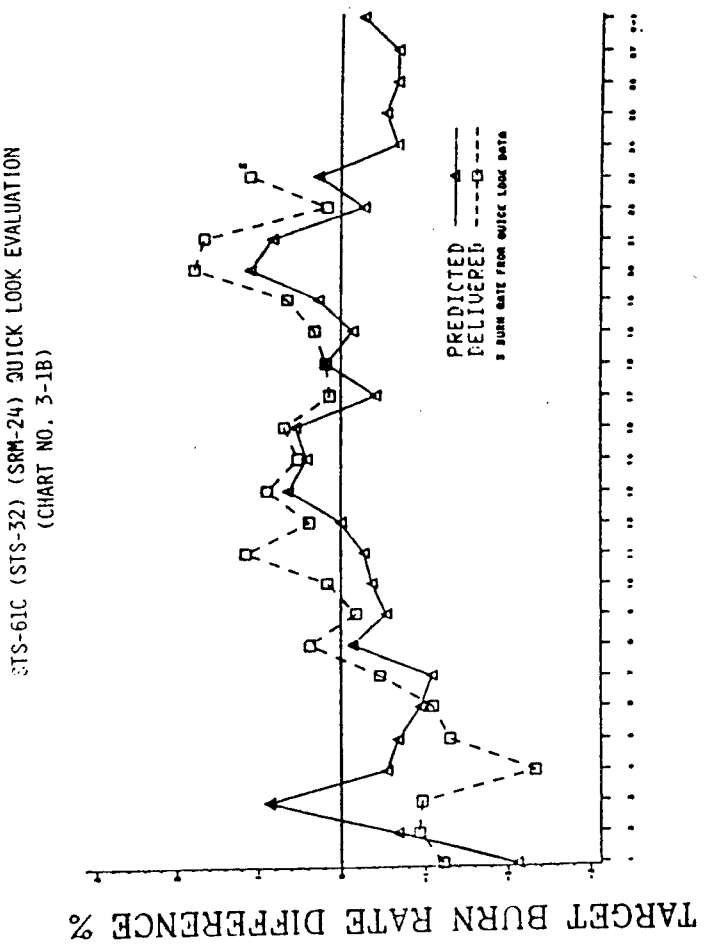
STS-61B QUICKLOOK EVALUATION
SRM-23 PERFORMANCE
(CHART NO. 3-1)

	PREDICTED (73°F)	EVALUATION (73°F)
o AVERAGE HEAD PRESSURE (5-20 SEC, PSIA)		
o LEFT MOTOR	901.0	926.2
o RIGHT MOTOR	899.7	928.7
o WEB TIME (SEC)		
o LEFT MOTOR	109.7	108.4
o RIGHT MOTOR	109.7	108.0
o ACTION TIME (SEC)		
o LEFT MOTOR	121.4	121.6
o RIGHT MOTOR	121.5	120.6
o SEPARATION COMMAND TIME BASED ON 50 PSIA ON LAST SRM + 5.7 SEC	124.8	123.9
o PROP. BURN RATE (IPS)		
o LEFT MOTOR	.372 (369)*	.374 (.371)*
o RIGHT MOTOR	.372 (.369)*	.376 (.373)*

* ADJUSTED TO A TEMPERATURE OF 60°F AND PRESSURE OF 625 PSIA

INFORMATION ON THIS PAGE WAS PREPARED TO SUPPORT AN ORAL PRESENTATION


STS-61B Quick-Look Evaluation
SRM-23 Performance
SRM-23A and B Thrust VS. VOL X Specification Limits
(CHART NO. 3-1A)
Vacuum Thrust (lbf)
4,000,000
3,000,000
2,000,000
1,000,000
10 20 30 40 50 60 70 80 90 100 110 120 130
Time (sec)
SRM-23 A (Left)
SRM-23 B (Right)
Level II Envelope


STS-61C (STS-32) (SRM-24) QUICK LOOK EVALUATION
(CHART NO. 3-1B)
PREDICTED
DELIVERED
TARGET BURN RATE DIFFERENCE %
SRM FLIGHT MOTOR SETS

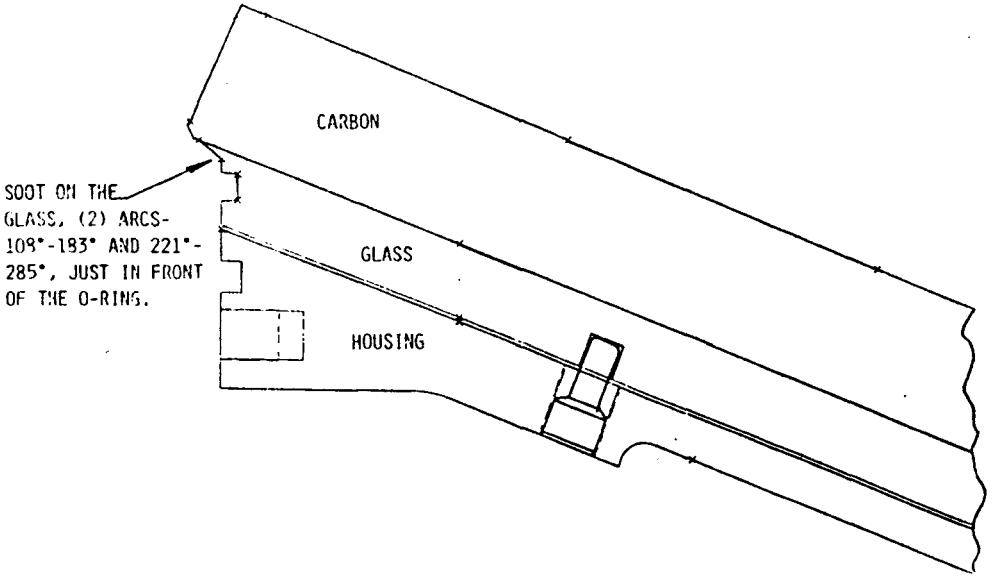
STS-61B (STS-31) (SRM-23) PERFORMANCE
(CHART NO. 3-2)

PROBLEM	CONCERN	RESOLUTION	TO BE DISCUSSED
ALL THREE STIFFENER RINGS ON BOTH SRM'S EXPERIENCED WATER IMPACT DAMAGE. CRACKS AND BUCKLES OBSERVED IN ALL THE RINGS. NINE OF THE 18 STIFFENER RING SEGMENTS WILL HAVE TO BE REPLACED.	REUSE OF SRM HARDWARE	ACCEPTABLE	NO
NO REPORTABLE POCKETS, GOUGES, OR WASHES IN THE INLET RINGS FOR BOTH NOZZLES.	SRM NOZZLE PERFORMANCE	ACCEPTABLE	NO
ALL ABLATIVE RINGS IN PLACE BUT SOME DISPLACEMENT FORWARD. APPARENTLY DUE TO WATER IMPACT, HAS CAUSED LOCAL RAISED AREAS AT SOME OF THE RING INTERFACES.	SRM NOZZLE PERFORMANCE	ACCEPTABLE	NO
SOOT OBSERVED ON SRM-23A (LH) AFT EXIT CONE SEALING SURFACE BUT THERE WAS NO APPARANT DAMAGE TO O-RING.	SRM NOZZLE PERFORMANCE	ACCEPTABLE	NO
SRM-23A (LH) HAD NO GAS PATHS THROUGH THE PUTTY AT ANY FIELD JOINT. (SEE SKETCH ON SUBSEQUENT CHART)	SRM PERFORMANCE	ACCEPTABLE	NO

PC 39511-

STS-61B (STS-31) (SRM-23) POST FLIGHT
(CHART NO. 3-2A)

AFT EXIT CONE



STS-61B (STS-31) (SRM-23) PERFORMANCE
(CHART NO. 3-3)

PROBLEM	CONCERN	RESOLUTION	TO BE DISCUSSED
SRM-23B (RH) HAD ONE GAS PATH AT THE AFT/ AFT CENTER JOINT AT 58° BUT IT DID NOT PROGRESS TO THE PRIMARY O-RING.	SRM PERFORMANCE	ACCEPTABLE	NO
LH MOTOR NOZZLE JOINT GAS PATH AT 340°. SOOT ON ALL SIDES OF O-RING GROOVE FROM 210.6° - 0° - 176.4° (316.8° TOTAL ARC). SOOT PAST PRIMARY SEAL FROM 325.8° - 354.6° (28.8° TOTAL ARC). SOOT PENETRATED ONLY ¼ INCH BEYOND PRIMARY SEAL. IT DID NOT TURN CORNER ONTO THE POLAR BOSS. PRIMARY O-RING DAMAGED TO UNKNOWN EXTENT SECONDARY O-RING NOT DAMAGED.	SRM PERFORMANCE	ACCEPTABLE	NO
RH MOTOR NOZZLE JOINT. GAS PATH AT 0°. SOOT BETWEEN PUTTY AND PRIMARY SEAL FROM 281° - 0° - 212° (291° TOTAL ARC) NO SOOT PAST PRIMARY SEAL. PRIMARY O-RING DAMAGED TO UNKNOWN RADIAL DEPTH, CIRCUMFERENTIAL LENGTH OF MAJOR ERODED REGION IS ABOUT 1 IN. SECONDARY O-RING NOT DAMAGED.	SRM PERFORMANCE	ACCEPTABLE	NO

PC 065514

NSFC PROBLEM ASSESSMENT SYSTEM
FEBRUARY 26, 1908

PAGE 1 OF 3
PCRS RECORD NUMBER
A09940
CONTRACTOR REPORT NUMBER
DR4-5745

PROBLEM TITLE: O-RING EROSION IN THE CASE TO NOZZLE JOINT

LAUNCH	CONTRACTOR	SYSTEM	CRIT	SYS LEVEL	MISC CODE	ABCDEFGHIJKLMNO
	THIOKOL	STRUCTURE	1R	YES		4 4 A69

SRM IDENTIFICATION	NOMENCLATURE	PART NUMBER	SERIAL/LOT	MANUFACTURER
SEGMENT ITEM	PRIMARY O-RING	1U52557-00	SRM-16A16A	HYDRAPAC
NOZZLE ASSY	AFT SEGMENT	1U50454-01	RSM-16A16A	THIOKOL
ARTICLE	SRB ASSY	10101-0001	BOTH	THIOKOL

FAI / OP	PREV COND	LOCATION	SYMPTOM	FAIL/UNFAT	FAIL MODE	CAUSE	FMEA NUMBER
F	PT-FLT INS	KSC-K	121	UC	123	M20	10-01-01

LAUNCH CONSTRAINTS
STS - STSL STSL STSL 61AL 61BL 61CL

CONCUR	TIME CYCLES	WUC	DATE OCCURED	DATE REC
JWT	N/A	A. MCDONALD	04/29/85	07/10/85
		R/C	EST RES DATE	CONT CLOSURE REC.
		M20		12/10/85

VEHICLE EFFECTIVITY
STS 6 SUBS

FA DESCRIPTION:

REF: SIMILAR REPORTS - A07934, A08299, A08814, A08815, A08807, A08939, A09240
PRIMARY O-RING EROSION WAS EXPERIENCED ON THE AFT SEGMENT OF BOTH RSM 16A & B. SRM-16A HAD THREE
RADIAL EROSION POCKETS, THE GREATEST BEING 0.171 IN., WITH SOOT DEPOSITS ON THE PRIMARY & SECONDARY O-RING
& SOME EROSION ON THE SECONDARY O-RING. SRM-16A HAD MINOR EROSION. THE GREATEST POCKET BEING
.085 IN., WITH NO SOOT BETWEEN THE PRIMARY & SECONDARY O-RINGS.

ELIMINATION / RESOLUTION:

7/11/85 - POST FLT INSPECTION OF SRM 16A REVEALED A GAS PATH THROUGH THE VACUUM PUTTY AT 54 DEGR.
THE PRIMARY O-RING HAD 1.2 IN. OF CIRCUMFERENTIAL EROSION WITH EVIDENCE OF BLOW BY AT THAT LOCATION. MAJOR EROSION OCCURRED BETWEEN 18 TO 1 O'CLOCK WITH THE PUTTY FOR 1.50 IN. OF THE CIRCUMFERENTIAL LENGTH IN DIRECTION WITHIN THE 50 IN. OF MAJOR EROSION THERE WERE THREE SEP EROSION POCKETS. THE GREATEST AMOUNT OF EROSION WAS 0.171 IN. & THE GREATEST AMOUNT OF CROSS SECTIONAL PERIMETER ERODED WAS 132 DEGS. BETWEEN THE PRIMARY & SECONDARY O-RING THERE WAS HEAVY DEPOSIT of SOOT FOR 90 DEGS TO EACH SIDE OF THE GAS PATH WITH A LIGHTER DEPOSIT FOR THE REMAINING 180 DEGS.

AT 54 DEGS THE SECONDARY O-RING EXPERIENCED 0.032 IN. OF EROSION & THE CROSS SECTIONAL PERIMETER WAS ERODED FOR 145 DEGS. THE CIRCUMFERENTIAL EROSION WAS 3.1 IN. WITH NO ADDITIONAL HEAT AFFECTED LENGTH. THERE WAS NO SOOT PAST THE SECONDARY O-RING.

POST FLT INSPECTION OF SRM 16A REVEALED A GAS PATH THROUGH THE VACUUM PUTTY AT 14.4 DEGS. AT THIS LOCATION THE PRIMARY O-RING WAS ERODED 0.085 IN. DEEP WITH 124 DEGS OF CROSS SECTIONAL PERIMETER ERODED. MAJOR EROSION EXTENDED FOR 3.4 IN. IN THE CIRCUMFERENTIAL DIRECTION WITHIN A HEAT AFFECTED LENGTH OF 4.7 IN. THE SECONDARY O-RING WAS UNAFFECTED.

BASED ON THE AMOUNT OF CHARRING, THE EROSION PATHS ON THE PRIMARY O-RING & WHAT IS UNDERSTOOD ABOUT THE EROSION PHENOMENON, IT IS BELIEVED THAT THE PRIMARY O-RING OF SRM 16A NEVER SEATED.

ORIG	R. MCINTOSH	D. NEWMAN	YEAR	J. FLETCHER	PAC REVIEW COMPLETE
DEPUTY		R & Q.A.	PROJECT	PAC	
				NSFC CLOSURE DATE	STATUS OPEN-R

PC 037710

CONFIDENTIAL (2)

7/13/05 - THIS PROBLEM IS NOT CONSIDERED A CONSTRAINT TO SIF. A 100 PSIG LEAK CHECK IS PERFORMED WHICH CONFIRMS SEATING OF THE SECONDARY O-RING. NOZZLE O-RINGS HAVE BEEN SHOWN TO SURVIVE EROSION DEPTHS OF 125 MILS IN HOT SUBSCALE TESTS. THE MAIN MODEL PREDICTS THE MAXIMUM EROSION DEPTH FOR THE SECONDARY O-RING TO BE 75 MILS IN THE WORST CASE CONDITION. THE EROSION ON THE SECONDARY O-RING WAS 32 MILS, WELL BELOW THE DEMONSTRATED HEATING CAPABILITY. ANALYTICAL RESULTS INDICATE THAT A WORST CASE EROSION CONDITION WOULD NOT FROM THE SECONDARY O-RING BEYOND ITS SEALING CAPABILITY. FINAL CAVITY LEAK CHECK AT 100 PSIG CAN BE MADE BY THE VACUUM PUTTY. HOWEVER, AT 200 PSIG THE VACUUM PUTTY CANNOT MASK THE LEAK CHECK. STS-51F (SRM 19) HAS THE 200 PSIG LEAK CHECK AND THEREFORE SHOULD EXHIBIT BEHAVIOR MORE LIKE STS-51D (SRM-19) AND STS-51B (SRM 18) WHICH EXHIBITED PRIMARY O-RING EROSION WHICH WAS WITHIN EXPERIENCE BASE WITH NO SECONDARY O-RING EROSION. ALL JOINTS WITH VACUUM PUTTY WILL BE LEAK CHECKED TO 200 PSIG IN THE FUTURE.

REF. TWR 15088 "STS-51F SOLID ROCKET MOTOR (SRM 19) FLT READINESS REVIEW," DATED 7-1-85, SRM SPECIAL TOPIC, NOZZLE O-RING ANOMALY OBSERVED ON STS-51B (SR-14A) - ---JWT 7-19-85---.

7/23/85 UPDATED FROM 7-11-85 PRB - RATIONALE FOR ACCEPTANCE WAS DISCUSSED AS INDICATED BY THE 7-2-85 UPDATE. NO OTHER SIGNIFICANT INFO WAS PROVIDED.

8/9/85 - STS-51F (SRM-19) FIELD JOINT O-RING INSPECTION REVEALED SEVERAL GAS PATHS THROUGH THE PUTTY BUT NO O-RING EROSION WAS FOUND ON ANY OF THE CASE JOINTS. THE NOZZLE TO CASE O-RING JOINT REVEALED A GAS PATH AT 21.6 DEGR WITH THE PRIMARY O-RING EXPERIENCING EROSION OF APPROX. 0.010 IN. THIS IS WITHIN THE FLT EXPERIENCE DATA BASE & IS NOT A FLT CONSTRAINT TO STS-51I (SRM-20) - ---JAMES W. THOMAS 8-1-85---.

8/19/85 UPDATED FROM 8-8-85 PRB - NOTHING NEW TO REPORT AT THIS MEETING.

9/20/85 UPDATED FROM 21 DAY REPORT - RESULTS OF THE TEAM INVESTIGATION ARE THAT THE PRIMARY O-RING FAILED TO SEAT AT 36 DEG WITH THE GAS JET IMPINGED UPON IT. A PORTION OF THE GAS JET THEN PASSED THE PRIMARY O-RING & CONTINUED UNTIL IT IMPINGED ON THE SECONDARY O-RING. THE SECONDARY O-RING SEATED & SEALED THE JOINT. THE PRIMARY O-RING CONTINUED TO ERODE DUE TO A COMBINATION OF HOT GAS JET IMPINGEMENT & CONVECTION HEATING UNTIL THE IGNITION TRANSIENT WAS COMPLETED & THE MOTOR REACHED OPERATING PRESSURE APPROX. 600 MILLISECONDS AFTER IGNITION.

9/20/85 - ON STS-51I NO O-RING EROSION WAS EXPERIENCED THAT EXCEEDED THE CURRENT EXPERIENCE BASE. ONE PRIMARY O-RING EROSION WAS NOTED ON THE LH NOZZLE TO CASE JOINT. NO SOOT WAS FOUND BEHIND THE PRIMARY O-RING. THIS PROBLEM IS NOT A CONSTRAINT TO STS-51J - ---L.J.W.---.

10/14/85 - ON STS-51J NO O-RING EROSION WAS EXPERIENCED THAT EXCEEDED THE CURRENT EXPERIENCE BASE. THIS PROBLEM IS NOT A CONSTRAINT TO STS-61A - ---JAMES W. THOMAS 10-17-85---.

11/6/85 - ON STS-61A NO O-RING EROSION WAS EXPERIENCED THAT EXCEEDED THE CURRENT EXPERIENCE BASE. THIS PROBLEM IS NOT A CONSTRAINT TO STS-61B - --- JWT 11-14-85---.

12/10/85 - ON STS-61B NO O-RING EROSION WAS EXPERIENCED ON THE FIELD JOINTS. HOWEVER, MINOR O-RING EROSION WAS EXPERIENCED ON BOTH THE LH AND RH NOZZLE TO CASE JOINTS. SOOT BLOW-BY WAS NOTED ON THE LH NOZZLE JOINT. THE EROSION EXPERIENCED DURING THE STS-61B LAUNCH WAS WELL WITHIN THE CURRENT EXPERIENCE BASE. THIS PROBLEM IS NOT A CONSTRAINT TO STS-61C - ---JWT 12-12-85---.

1/23/86 RESOLUTION - SRM FIELD AND NOZZLE JOINTS HAVE EXPERIENCED EROSION OF THE PRIMARY O-RINGS DURING SEVERAL MISSIONS AND STATIC TEST AS DETERMINED BY POST FLIGHT INSPECTION.

PC 037711

THE PRIMARY O-RING IN A NUMBER OF INSTANCES WAS SUFFICIENTLY DAMAGED AND/OR DISPLACED TO ALLOW BLOWBY. THEREFORE EXPOSING THE SECONDARY O-RING TO HOT COMBUSTION GASES. SOME EROSION WAS EXPERIENCED ON THE SECONDARY O-RING AT THE NOZZLE JOINT BUT NOT TO THE POINT WHERE THE O-RING SEALING CAPABILITY WAS JEOPARDIZED.

THE CAUSE OF THE O-RING EROSION HAS BEEN ATTRIBUTED TO HOT GAS PATHS THROUGH THE VACUUM PUTTY ALLOWING THE HOT GASES TO IMPINGE UPON THE O-RING. IMPROPER SEATING OF THE PRIMARY O-RING ALLOWED BLOWBY AND MINOR HOT GAS IMPINGEMENT ON THE SECONDARY O-RING. THE MAGNITUDE OF THE O-RING EROSION IS A FUNCTION OF ORIFICE SIZE AND FACE VOLUME (AREA BETWEEN THE PRIMARY O-RING AND THE PUTTY).

PRIMARY O-RING EROSION IS EXPECTED TO CONTINUE SINCE NO CORRECTIVE ACTION HAS BEEN ESTABLISHED THAT WILL PREVENT HOT GASES FROM REACHING THE PRIMARY O-RING CAVITY. STEPS HAVE BEEN TAKEN TO ASSURE THAT THE SECONDARY O-RING WILL BE SEATED AND ANALYTICAL ANALYSIS HAVE INDICATED THAT UNDER A WORST CASE SITUATION, EROSION OF THE SECONDARY O-RING WILL NOT BE SEVERE ENOUGH TO ALLOW A LEAK PATH PAST THE SECONDARY O-RING. STW7-2747 HAS CHANGED EFFECTING AN INCREASE IN THE LEAK CHECK PRESSURE FROM 100 PSIA TO 200 PSIA. WITH THE 100 PSIG LEAK TEST THE SECONDARY O-RING WOULD BE SEATED BUT ANY GAS PATHS THROUGH THE PUTTY WOULD BE UNDETECTABLE. BY INCREASING THE LEAK TEST PRESSURE TO 200 PSIG, ANY LEAK PATHS THROUGH THE PUTTY WOULD BE DETECTABLE. THE 200 PSIG LEAK TEST WAS IMPLEMENTED ON SRM-19 AND SUBSEQUENT MOTORS.

THE STACKING PROCEDURES HAVE BEEN CLARIFIED TO ASSURE MOTOR-TO-MOTOR CONSISTENCY IN JOINT MATING. THE OPERATIONAL MAINTENANCE DOCUMENT CHANGES WERE EFFECTIVE ON SRM L024 AND SUBS..

ANALYTICAL STUDIES BASED ON BOTH IMPINGEMENT EROSION AND BLOWBY EROSION SHOW THAT THIS PHENOMENON HAS AN ACCEPTABLE CEILING SINCE IMPLEMENTING THE ABOVE CHANGES. RECENT EXPERIENCE HAS BEEN WITHIN THE PROGRAM DATA BASE.

THIS FINAL IMPROVEMENT PROGRAM PLAN WILL CONTINUE UNTIL THE PROBLEM HAS BEEN ISOLATED AND CAUSE ELIMINATED TO THE SRM SEALS. STATUS WILL CONTINUE TO BE PROVIDED IN THE FLIGHT READINESS REVIEWS AND IN FORMAL TECHNICAL REVIEWS AT MTI AND MSFC. AT THE CONCLUSION OF THE PROGRAM, A COMPREHENSIVE REPORT WILL BE WRITTEN TO CONSOLIDATE THE RESULTS, CONCLUSIONS, AND RECOMMENDATIONS.

THIS PROBLEM IS CONSIDERED CLOSED BASED ON MTI REPORT TWR-14359, REV. A, "IMPROVEMENTS OF SPACE SHUTTLE SRM MOTOR SEALS" DATED 9-30-85 AND MTI LETTER EI50/BOR-86-114 "RATIONALE FOR CLOSURE OF THE O-RING EROSION PROBLEM", A07934, DR4-5/301 DATED 1-2-86.

constraint can be lifted by Project Manager

PC 037712

MSFC PROBLEM ASSESSMENT S
PAGE 2, 1300

PAGE 1 OF
MSFC NCR/DR
A00914
CONTROLLED BY
ON 4-5-84

PROBLEM TITLE: BLOWBY JOINT PRIMARY O-RING DAMAGE

CRITICAL	CERTIFICATION	X SYSTEM	UNIT	OTS LEVEL	MSFC CASE	ASSESSED
NSA	ENGINEAL	STRUCTURE	10	YES		C

HARDWARE IDENTIFICATION	NOMENCLATURE	PART NUMBER	SERIAL/LOT	PROBE ACTIVE
FAILURE DURING ITEM	PRIMARY O-RING	1U53220-24	BRT 0015A	THROUGH
NEXT HIGHER ASSY	BNB ASSY	10101-0001	BRT 0015A	THROUGH
TEST ARTICLE	BNB ASSY	10101-0001	BTD10C	THROUGH

TEST / OP	PREV COND	LOCATION	DIRECTION	FAIL/UNBAT	FAIL MODE	CYCLE	TIME HI
FLT	FT-FLT INS	KSC-K	1X1	UC	124	020	10:01:0

DESIGN CONSTRAINTS	CONCUR	FINAL CYCLES	CAR	DATE RECORD	DATE REC
YES	JOE	N/A	R. LANEY	02/03/84	02/17/84

CRITICAL EFFECTIVITY	R/C	EST REC DATE	CONT CLOSD
BI019 & GUNB	M20		12/10/85

PROBLEM DESCRIPTION:

REF: SIMILAR REPORTS A00014 & A00299, A00615, A00607, A00339, A09360, A09208
WEIGHT CHAR CONDITION ON PRIMARY O-RING SEAL IN FWD FIELD JOINT ON BRM A97 OF BTS-11 FLT, MISSION 410.

INVESTIGATION / RESOLUTION:

REMEDIAL ACTION - NONE REQ'D; PROB OCCURRED DURING FLT. THE PRIMARY O-RING SEAL IN THE FWD FIELD JOINT EXHIBITED A CHARRED AREA APPROX 1 IN. LONG, 0.030-0.050 IN. DEEP, & 0.100 IN. WIDE. THIS WAS DISCOVERED DURING POST-FLT DISASSY AT KSC.

3/8/84 - POSSIBLY EXIST FOR SOME O-RING EROSION ON FUTURE FLTS. ANALY INDICATED MAX. EROSION POSSIBLE IS 0.090 IN. ACCORDING TO THE FLT READINESS REVIEW FINDINGS FOR BTS-10. LABORATORY TEST SHOWS SEALING INTEGRITY AT 3500 PSI USING AN O-RING WITH A SIMULATED EROSION DEPTH OF 0.095 IN., THEREFORE, THIS IS NOT A CONSTRAINT TO FUTURE LAUNCHES. ADDRESSED PRB ON 3-8-84.

4/27/84 UPDATED FROM 5 DAY REPORT - THE FAILURE DEFINED ABOVE WAS A SECONDARY PROB. THE VACUUM PUTTY LAY-UP IS CONSIDERED THE PRIME PROB. THE EFFECT ON THE O-RING SEAL BEING CHARRED HAS BEEN REVISED TO APPROX 3 IN. IN LENGTH, 0.040 IN. DEEP & AFFECTED 21% DEPTH OF THE O-RING CROSS-SECTION PERIMETER. THE O-RINGS IN THE BNB SEGMENT ASSY JOINTS ARE DESIGNED AS PRIMARY SEALS & ARE NOT INTENDED TO BE EXPOSED TO HOT GASES. D104-20147 VACUUM SEAL PUTTY IS LAID UP IN THE JOINT AREA DURING MOTOR SEGMENT MATING TO PROVIDE A SEAL/HOT GAS BARRIER TO PROTECT THE O-RINGS. THE PRIME PROB IS THE LAY UP OF THE VACUUM PUTTY ALLOWING A HOT GAS PATH TO THE O-RING SEAL. THE O-RING IS USED IN THE FWD FIELD JOINT WHICH JOINED THE FWD MOTOR SEGMENT (P/N 1U53216-04, S/N 0000001) & FWD CENTER MOTOR SEGMENT (P/N 1U52217-03, S/N 00000011).

4/17/84 UPDATED FROM 4-12-84 PRB & MARCH MONTHLY SUMMARY REPORT - TEST FIXTURE & TEST PLANS HAVE BEEN PREPARED & SHT STARTED TO SIMULATE THE EFFECT OF THE HOT GAS ON THE O-RING. THESE TESTS ARE EXPECTED TO CONFIRM THAT HOT GAS WILL NOT ERODE THE O-RING TO A POINT WHERE SEAL INTEGRITY WOULD FAIL. THE PUTTY LAY-UP IS STILL TO BE STUDIED.

ORIGINATOR	R. MCINTOSH	G. NEWMAN	L. DEAN	J. FLETCHER	PAC REVIEW COMP
	DESIGN	R & Q.A.	PROJECT	PAC	OF
APPROVAL				MSFC CLOSURE DATE	STATUS OPEN-R

5/1/84 UPDATED FROM 21-DAY REPORT - THE FORWARD FIELD JOINT DESIGN IS BEING EVALUATED. IT HAS BEEN DETERMINED THAT THE FREE VOLUME, THAT PORTION OF THE JOINT FWD OF THE PRIMARY O-RING NOT FILLED WITH EITHER VACUUM PUTTY OR GREASE, IS WHERE THE PROB LIES. IT IS BELIEVED THAT THE O-RING CHAR IS CAUSED BY A SHORT TIME DURATION IMPINGEMENT OF A HIGH ENERGY JET WHICH IS INDUCED DURING IGNITION PRESSURIZATION BY A COMBINATION OF VOIDS IN THE PROTECTIVE VACUUM PUTTY & THE FILLING OF AVAILABLE FREE VOLUMES CREATED BY THE TOL. OF MATING PARTS & THE O-RING SLOT. THE TEST BEING CONDUCTED SHOULD VERIFY THAT THIS CONDITION WILL NOT CAUSE A SERIOUS PROB.

5/23/84 UPDATED FROM 5-10-84 PRB - INVESTIGATIONS ARE CONTINUING. O-RING DAMAGE IS NOT TO THE POINT WHERE SEAL INTEGRITY WOULD FAIL (LESS THAN 0.090 IN. POSSIBLE). O-RING CHAR HAS ALSO BEEN FOUND ON STS-13.

5/30/84 UPDATED FROM MAY MONTHLY SUMMARY REPORT - TWR-14936 "PERFORMANCE CHARACTERISTICS OF THE SRM O-RING ASSY TEST PLAN" HAS BEEN RELEASED & THE 5 IN. CP HOT TESTING INITIATED. DATA FROM THE TESTS IS BEING ANALYZED & WILL BE IN NEXT MONTH'S REPORT.

6/20/84 UPDATED FROM 6-14-84 PRB & JUNE MONTHLY SUMMARY REPORT - PRELIMINARY FINDINGS FROM THE O-RING TEST CONFIRM THE HOT JET IMPINGEMENT SCENARIO & THE MAJOR CONCLUSIONS DRAWN ARE:

1. O-RING EROSION IS A FUNCTION OF ORIFICE SIZE & FREE VOLUME.
2. GREASE ON THE O-RINGS MODERATELY REDUCES EROSION.
3. NO DAMAGE TO THE O-RING RESULTS WHEN THE ORIFICE IS ABSENT, SIMULATING THE ABSENCE OF VACUUM PUTTY.

A METHOD OF SOLUTION HAS NOT BEEN DETERMINED. THIOKOL WILL HAVE RECOMMENDED CHANGES IN JULY 1984.

A07934, A08014 & A08229 DOCUMENT DIFFERENT OCCURRENCES OF THE SAME GENERAL PROB. ALL ANALYSES DONE & CONCLUSIONS MADE WILL APPLY TO ALL 3 OCCURRENCE & WILL BE REPORTED UNDER A07934 HENCEFORTH.

7/10/84 ADDRESSED AT PRB ON 7-12-84 - TWO SPECS WERE RELEASED CHANGING THE LAYUP OF THE VACUUM PUTTY & GREASE APPLICATION. THESE CHANGES WERE TESTED SATISFACTORILY, HOWEVER ADDITIONAL TEST ARE REQ'D.

9/24/84 UPDATED FROM AUG. MONTHLY SUMMARY REPORT & 9-13-84 PRB - VACUUM SEAL PUTTY LAYUP SPECS STW7-3193 & STW7-3220 WITH NEW LAYUP PROCEDURES HAVE BEEN RELEASED. STW7-3194 IS STILL IN THE SIGN OFF PROCESS. INVESTIGATIONS ARE CONT'D. PLANS ARE TO PERFORM FULL SCALE NOZZLE & JOINT TESTS ALSO LOOKING AT NEW PUTTY. THIS PROBLEM OCCURRED AGAIN ON FLT 41D, SEE PROBLEM REPORT A08615.

10/2/84 UPDATED FROM SEPT. MONTHLY SUMMARY REPORT - VACUUM SEAL PUTTY LAYUP SPEC STW7-3220 HAS BEEN REVISED TO REDUCE THE AMOUNT OF GREASE & STW7-3194 IS STILL IN SIGN OFF.

11/8/84 UPDATED FROM OCT. MONTHLY SUMMARY REPORT - THE JOINT FILLER NTRL TEST PLAN IS RELEASED & FULL SCALE JOINT ASSY TESTING WAS INITIATED 10-10-84 AT MORTON THIOKOL'S H-7 CLEARFIELD FACILITY.

11/14/84 UPDATED FROM PRB MTG 11-8-84 - THE TEST PLAN SHOULD BE REVISED & TEST STARTED WITHIN APPROX. 2 WKS.

1/2/85 UPDATED FROM PRB MTG 12-13-84 - A NEW PROGRAM WAS REQUESTED BY MSFC FOR EVALUATION & TESTING. THIS IS NOW IN WORK & THE ESTIMATED COMPLETION DATE IS 1-4-85. ANY DESIGN CHANGES ARE PENDING THE

PC 037557

TEST RESULTS. A COPY OF THE 40-LB & 5 IN. CP MOTOR TEST SCHEDULE IS INCLUDED IN THIS PRSR. THE O-RING CHAR PROBLEM IS A REVIEW ITEM FOR EACH FLT READINESS REVIEW (FRR).

1/22/85 UPDATED FROM PRB MTG ON 1-10-85 - ALL SCALE MODEL TESTING WILL BE COMPLETED IN JAN. FULL SCALE TESTS ARE TO FOLLOW.

2/15/85 UPDATED FROM 2-7-84 PRB & JAN. MONTHLY SUMMARY REPORT - TESTING OF VACUUM SEAL PUTTY LAYUP CONFIG. USING 5 IN. CP MOTOR IS CONTINUING. TORCH TESTS HAVE BEEN DONE ON ASBESTOS FILLED PUTTY MFR'D BY RANDOLPH & INHONT & A NONASBESTOS PUTTY MFR'D BY INHONT. BOTH ASBESTOS PUTTIES PERFORMED COMPARABLY. THE HEAT TRANSFER RATE OF THE NONASBESTOS PUTTY WAS MORE THAN TWICE THAT OF THE ASBESTOS FILLED PUTTY. FORTY LB. CHAR TEST MOTORS ARE SCHEDULED TO FURTHER TEST THE PUTTY TYPES.

EROSION WAS AGAIN EXPERIENCED ON FLT 51C. THE O-RING BURNS WERE AS BAD OR WORSE THAN PREVIOUSLY EXPERIENCED, SEE PROBLEM REPORT A09939/CONT. NO. 4-3/41. DESIGN CHANGES ARE PENDING TEST RESULTS. CHANGES BEING CONSIDERED ARE: MODIFYING THE O-RINGS, DOUBLE BARRIER VENTING & ADDING GREASE AROUND O-RINGS TO FILL THE VOID LEFT BY THE PUTTY.

3/18/85 UPDATED FROM 2-7-84 PRB - TESTING & EVALUATION PROGRAM IS CONTINUING. DESIGN CHANGES ARE PENDING TEST RESULTS.

3/29/85 UPDATED FROM FEB. MONTHLY SUMMARY REPORT - DISASSY OF THE SRM 15A & 15B NOZLE TO AFT CASE JOINTS REVEALED GAS PATHS THRU THE PUTTY AT 109 DEG & 64 DEG, RESPECTIVELY. BOTH PRIMARY O-RINGS HAD HEAT AFFECTED AREAS (NO EROSION) AT THE GAS PATH LOCATIONS. SOOT WAS FOUND BETWEEN THE PRIMARY & SECONDARY O-RINGS IN EACH JOINT BUT THE SECONDARY O-RINGS WERE UNTOUCHED. THE SOOT WAS MOST LIKELY DEPOSITED DURING IGNITION PRESSURIZATION WHEN GAS PASSED THE PRIMARY O-RINGS BEFORE THEY SEATED.

A DR4-S PROBLEM REPORT WAS NOT WRITTEN FOR THIS OCCURRENCE FOR THE FOLLOWING REASONS:

1) THE PRIMARY O-RINGS SEATED & THE PRESSURE INTEGRITY OF THE JOINTS WAS MAINTAINED.
2) THE O-RINGS WERE NOT ERODED.
3) GAS PATHS & O-RING EROSION HAVE OCCURRED ON PREVIOUS FLTS & ARE EXPECTED TO OCCUR RANDOMLY ON FUTURE FLTS.

IT HAS BEEN REPORTED HERE TO DOCUMENT THE OCCURRENCE.

4/17/85 UPDATED FROM PRB MEETING 4-11-85 - O-RING INVESTIGATION & PUTTY TESTING IS CONTINUING. MSFC'S FINDING THAT THE PUTTY DEBONDED IN GREASE COULD NOT BE DUPLICATED BY THIOKOL.

4/24/85 UPDATED FROM APRIL MONTHLY SUMMARY REPORT - AT NASA REQUEST, A SOLUTION FOR O-RING EROSION WILL NOT INVOLVE A RADICAL DESIGN CHANGE. THEREFORE, THE POSSIBLE SOLUTIONS UNDER CURRENT INVESTIGATION ARE LIMITED TO:

1. NEW O-RING MTRL AND/OR DIAMETER.
2. NEW VACUUM PUTTY AND/OR LAYUP PROCEDURE.

6/6/85 UPDATED FROM MAY MONTHLY SUMMARY REPORT - DISASSY OF SRM 17A & 17B REVEALED GAS PATHS & EROSION IN BOTH NOZZLE TO CASE JOINTS. THE GAS PATH IN THE NOZZLE TO AFT CASE JOINT OF SRM 17A WAS CENTERED AT 14.4 DEGS & THE O-RING HAD A .010 IN. OF EROSION. SRM 17B'S NOZZLE JOINT HAD A GAS PATH AT 116 DEGS & THE O-RING HAD A .068 IN. OF EROSION. THERE WAS SOOT DEPOSITED IN THE PRIMARY

PC 037598

O-RING GROOVES OF EACH JOINT. HOWEVER, NO BOOT PASSED THE PRIMARY O-RING.

6/25/85 UPDATED FROM 6-13-85 PRB - PLAN TO GROVE O-RING PAST THE .095 IN. TO DETERMINE WHEN A LEAK WILL OCCUR. HOT FIRE JOINT TESTS ARE PLANNED FOR AUG., 1985.

7/16/85 UPDATED FROM JUNE MONTHLY SUMMARY REPORT DATED 6-17-85 - ONE POSSIBLE EXPLANATION FOR THE GREATER EROSION DEPTHS OBSERVED ON DRD 170 (60 MILS) & NOZ DR-7 (80 MILS) RELATED TO AN ASPECT OF NOZZLE JOINT ASSEMBLY: AN ECCENTRIC NOZZLE JOINT CAN OCCUR MORE EASILY THAN AN ECCENTRIC FIELD JOINT DUE TO A) ITS HORIZONTAL VS VERTICAL POSITION DURING ASSEMBLY, B) SMALL JOINT CLEARANCES, AND C) THE ABSENCE OF SHIMS. CONSEQUENTLY, IT IS POSTULATED THAT A REDUCED JOINT GAP (& THEREFORE GAS JET ORIFICE) OCCURRED AT THE CIRCUMFERENTIAL LOCATION OF THE BLOWHOLE, THEREBY CAUSING ENHANCED EROSION. THE VOLUME FILLING MODEL PREDICTS THAT AN ECCENTRICITY OF ONLY 7 MILS WILL ROUGHLY DOUBLE THE 40 MILS EROSION PREDICTED FOR CONCENTRIC NOZZLE JOINTS. THUS AN O-RING EROSION EXCEEDING 80 MILS CAN BE CAUSED BY JOINT ECCENTRICITY IF THE REDUCED GAP ALIGNS WITH THE BLOWHOLE. INVESTIGATION OF THIS SCENARIO IS CONTINUING.

7/23/85 UPDATED FROM 7-11-85 PRB - THIS PROBLEM WAS SCHEDULED FOR STATUSING, BUT NOT DISCUSSED SINCE NO SIGNIFICANT PROGRESS HAS BEEN MADE SINCE THE LAST UPDATE.

8/19/85 UPDATED FROM 8-8-85 PRB & JULY 1985 MONTHLY SUMMARY REPORT - THE ANALYTICAL MODEL OF O-RING EROSION HAS BEEN USED TO PREDICT A WORST CASE EROSION OF 75 MILS ON THE SECONDARY O-RING SHIELD THE PRIMARY O-RING FAIL TO SEAL. PRESSURE TESTS OF O-RINGS WITH METAL REMOVED TO SIMULATE EROSION & SUBSCALE MOTOR FIRINGS HAVE DEMONSTRATED THAT THIS IS WELL BELOW THE SEAL CAPABILITY OF ERODED O-RINGS. EVALUATION OF THE RESULTS OF STS-51F ARE CONTINUING.

9/23/85 UPDATED FROM JULY MONTHLY SUMMARY REPORT DATED 8-15-85 - DR4-S/48 (A09260) REPORTED .080 INCH OF EROSION OBSERVED IN THE NOZZLE TO CASE JOINT ON DEVELOPMENT MOTOR NO. 7.

DR4-S/49 (A09288) REPORTED .171 IN. OF EROSION ON PRIMARY O-RING & .032 IN. OF EROSION ON THE SECONDARY O-RING IN THE NOZZLE TO CASE JOINT OF RSR 16A.

THE INVESTIGATIONS ARE CONTINUING & WILL BE REPORTED HERE AS THEY ARE THE SAME GENERIC PROBLEM.

INSPECTION OF RSR 18A FIELD JOINTS REVEALED NO EROSION HOWEVER, THE NOZZLE TO CASE JOINT HAD PRIMARY O-RING EROSION AT 191.2 DEGS WITH A MAX. DEPTH OF .013 IN. RSR 18B HAD NO FIELD JOINT EROSION. THE NOZZLE TO CASE JOINT EXPERIENCED GAS PATHS AT 63, 93.6, 270 & 342 DEGS. THE PRIMARY O-RING WAS ERODED ONLY AT 270 & 342 DEGS WITH DEPTH OF .009 & .023 IN., RESPECTIVELY. A PROBLEM REPORT WAS NOT WRITTEN FOR THESE OCCURRENCES SINCE THEY WERE WITHIN HISTORICAL LEVELS. THEY ARE REPORTED HERE FOR INFORMATION.

10/2/85 UPDATED FROM AUGUST MONTHLY SUMMARY REPORT DATED 9-16-85 - RSR 15B (STS-51F) EXPERIENCED A GAS PATH AT 321.6 DEGS. AT THIS LOCATION THE O-RING WAS ERODED APPROX. 0.001 IN., WELL WITHIN THE THE EXPERIENCE BASE. QM-5 WILL BE DELAYED TO ALLOW FOR DEVELOPMENT OF O-RING SOLUTION.

11/4/85 UPDATED FROM OCTOBER MONTHLY SUMMARY REPORT & 10-17-85 PRB - ANALYSIS OF DATA FROM STS-51J & QM-5 TEST ARE IN WORK.

11/19/85 UPDATED FROM 11-7-85 PRB MEETING. STS-61A EXPERIENCED NO FIELD JOINT EROSION, HOWEVER, BOOT BLOWBY WAS FOUND ON 3 OF THE 6 PRIMARY O-RINGS. STS-61A R.H. NOZZLE EXPERIENCED EROSION OF THE PRIMARY O-RING. THE EROSION WAS WELL WITHIN THE CURRENT DATA BASE.

PC 027599

BASED ON THE JOINT PUTTY & O-RING PROBLEM, A PROGRAM WAS INITIATED TO ESTABLISH ACCEPTABLE MATERIALS & CONFIGURATION THAT COULD BE TESTED ON QM-5 & ULTIMATELY INCORPORATED IN FLIGHT MOTORS.

12/17/85 - UPDATE FROM NOVEMBER MONTHLY SUMMARY REPORT.
THE O-RING TASK FORCE HAS STARTED ASSEMBLY AND COLD FLOW TESTING OF NOZZLE PUTTY AND JOINT FILLERS. TESTING IS CONTINUING.

SRM 21 - NO DAMAGE
SRM 22 - RIGHT HAND (B) NOZZLE

THERE WAS EROSION AT 97.2 DEGREE. IT WAS .075 INCH DEEP AND HAD 162 DEGREE OF CROSS SECTIONAL PARIMETER AFFECTED WITH 13 INCHES OF HEAT DAMAGE LENGTH AND 23 INCHES OF HEAT AFFECTED LENGTH. THIS IS NOT THE WORST CASE TO DATE. THERE WAS NO OTHER REPORTED O-RING DAMAGE.

12/20/85 UPDATED FROM 12-12-85 PRB MEETING - THIOKOL IS CONTINUING STUDIES RELATIVE TO THE O-RING EROSION AND PUTTY LAYUP ON AN ON-GOING BASIS. A TOTAL SOLUTION TO THIS PROBLEM IS LONG RANGE; HOWEVER, AN ACCEPTABLE DATA BASE HAS BEEN ESTABLISHED AND SUFFICIENT ENHANCEMENT HAS BEEN MADE TO PROVIDE THE NECESSARY CONTROLS TO PREVENT RECURRENCE OF THE PROBLEM OUTSIDE THE ESTABLISHED DATA BASE. CLOSURE PAPER HAS BEEN PREPARED AND IS BEING EVALUATED.

1/20/86 UPDATED FROM DECEMBER MONTHLY SUMMARY REPORT. THE TASK FORCE HAS MADE ONE HOT GAS TEST. THE PRELIMINARY RESULTS INDICATE THAT THE TEST CHAMBER NEEDS TO BE REDESIGNED.

SPACE SHUTTLE 6M FLIGHT SET 23 WAS LAUNCHED NOVEMBER 27. INSPECTION OF THE FIELD JOINTS REVEALED NO DAMAGE. WHILE BOTH OF THE NOZZLE-TO-CASE JOINTS HAD EROSION:

	23A	23B
EROSION LENGTH (INCHES)	3.0	7.0
HEAT AFFECTED LENGTH (INCHES)	22.0	25.0
EROSION DEPTH (INCHES)	.017	.037
DEGREE LOCATION	346 DEGREES	0 DEG

1/23/86 RESOLUTION - SRM FIELD AND NOZZLE JOINTS HAVE EXPERIENCED EROSION OF THE PRIMARY O-RINGS DURING SEVERAL MISSIONS AND STATIC TEST AS DETERMINED BY POST FLIGHT INSPECTION. THE PRIMARY O-RING IN A NUMBER OF INSTANCES WAS SUFFICIENTLY DAMAGED AND/OR DISPLACED TO ALLOW BLOWBY. THEREFORE EXPOSING THE SECONDARY O-RING TO HOT COMBUSTION GASES. SOME EROSION WAS EXPERIENCED ON THE SECONDARY O-RING AT THE NOZZLE JOINT BUT NOT TO THE POINT WHERE THE O-RING SEALING CAPABILITY WAS JEOPARDIZED.
THE CAUSE OF THE O-RING EROSION HAS BEEN ATTRIBUTED TO HOT GAS PATHS THROUGH THE VACUUM PUTTY ALLOWING THE HOT GASES TO IMPINGE UPON THE O-RING. IMPROPER SEATING OF THE PRIMARY O-RING ALLOWED BLOWBY AND MINOR HOT GAS IMPINGEMENT ON THE SECONDARY O-RING. THE MAGNITUDE OF THE O-RING EROSION IS A FUNCTION OF ORIFICE SIZE AND FREE VOLUME (AREA BETWEEN THE PRIMARY O-RING AND THE PUTTY).

PRIMARY O-RING EROSION IS EXPECTED TO CONTINUE SINCE NO CORRECTIVE ACTION HAS BEEN ESTABLISHED THAT WILL PREVENT HOT GASES FROM REACHING THE PRIMARY O-RING CAVITY. STEPS HAVE BEEN TAKEN TO ASSURE THAT THE SECONDARY O-RING WILL BE SEATED AND ANALYTICAL ANALYSIS HAVE INDICATED THAT UNDER A WORST CASE SITUATION, EROSION OF THE SECONDARY O-RING WILL NOT BE SEVERE ENOUGH TO ALLOW A LEAK PATH PAST THE SECONDARY O-RING. STW7-2747 WAS CHANGED EFFECTING AN INCREASE IN THE LEAK CHECK PRESSURE FROM

PC 037600

100 PSIG TO 200 PSIG. WITH THE 100 PSIG LEAK TEST THE SECONDARY O-RING WOULD BE SEATED BUT ANY GAS PATHS THROUGH THE PUTTY WOULD BE UNDETECTABLE. BY INCREASING THE LEAK TEST PRESSURE TO 200 PSIG, ANY LEAK PATHS THROUGH THE PUTTY WOULD BE DETECTABLE. THE 200 PSIG LEAK TEST WAS IMPLEMENTED ON SRM-19 AND SUBSEQUENT MOTORS.

THE STACKING PROCEDURES HAVE BEEN CLARIFIED TO ASSURE MOTOR-TO-MOTOR CONSISTENCY IN JOINT MATING. THE OPERATIONAL MAINTENANCE DOCUMENT CHANGES WERE EFFECTIVE ON SRM L024 AND SUBS.

ANALYTICAL STUDIES BASED ON BOTH IMPINGEMENT EROSION AND BLOWBY EROSION SHOW THAT THIS PHENOMENON HAS AN ACCEPTABLE CEILING SINCE IMPLEMENTING THE ABOVE CHANGES. RECENT EXPERIENCE HAS BEEN WITHIN THE PROGRAM DATA BASE.

THIS SEAL IMPROVEMENT PROGRAM PLAN WILL CONTINUE UNTIL THE PROBLEM HAS BEEN ISOLATED AND DAMAGE ELIMINATED TO THE SRM SEALS. STATUS WILL CONTINUE TO BE PROVIDED IN THE FLIGHT READINESS REVIEWS AND IN FORMAL TECHNICAL REVIEWS AT MTI AND MSFC. AT THE CONCLUSION OF THE PROGRAM, A COMPREHENSIVE REPORT WILL BE WRITTEN TO CONSOLIDATE THE RESULTS, CONCLUSIONS, AND RECOMMENDATIONS.

THIS PROBLEM IS CONDITIONED CLOSED BASED ON MTI REPORT TOR-14235, REV. A, "IMPROVEMENTS OF SPACE SHUTTLE SRM MOTOR SEALS" DATED 8-30-85 AND MTI LETTER E150/DOR-86-114 "RATIONALE FOR CLOSURE OF THE O-RING EROSION PROBLEM", A07934, DR4-5/30) DATED 1-2-86.

PC 037601

VIII-J

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

SEP 15 1980

Reply to Attn of SA01

TO: Distribution

FROM: SA01/Mr. Lindstrom

SUBJECT: Assigning Launch Constraints on Open Problems Submitted to MSFC PAS

The Shuttle Projects Office has established a requirement for identification of launch constraints for problems being reported to the MSFC Problem Assessment System (PAS) by element contractors. Each element contractor (Rocketdyne, MMC, USBI, and Thiokol) has been directed to support this requirement by providing launch constraint information on each new problem submitted to the MSFC PAS. The launch constraint information provided by the contractor is based upon their preliminary technical evaluation and will require final concurrence by the responsible element project manager.

a. The following guidelines have been established to aid in making constraint decisions on open problems and are limited to recurrence control determination only. In accordance with practices established on past programs, remedial actions (e.g. removal and replacement of defective hardware, etc.) for correcting discrepancies on the vehicle to be launched are considered launch constraints and are tracked by the KSC system.

(1) All open problems coded criticality 1, 1R, 2, or 2R will be considered launch constraints until resolved (recurrence control established and its implementation effectivity determined) or sufficient rationale, i.e., different configuration, etc., exists to conclude that this problem will not occur on the flight vehicle during prelaunch, launch, or flight.

(2) Problems coded criticality 3 will not be considered launch constraints unless (a) the potential exists of leading to a criticality 1 or 2 failure mode; or (b) the failed component has multiple use on the element and more than one occurrence could lead to a criticality 2 condition; or (c) the failure could result in multiple loss of flight instrumentation channels. If a criticality 3 is determined to be a launch constraint, it will be treated the same as a.(1) above.

b. To assure that each reported problem is reviewed for correct criticality and constraint assignment by the appropriate MSFC personnel, the following procedure will be followed:

(1) The responsible S&E design actionee and the element project office actionee will review each problem upon receipt to assure that the criticality and constraint assignments meet with their approval. Exceptions to criticality or constraint assignment will be coordinated with the Problem Assessment Center (PAC) actionee within two working days from receipt of the problem report.

(2) The Problem Assessment Center will prepare a weekly constraints list by element. This list will be submitted to the Shuttle R&QA Support Office, EG03, for input to the Shuttle Projects Manager, SA01. Copies of the constraint list will be furnished concurrently to each Shuttle Element Project Manager.

(3) Launch constraints will be reviewed at each Problem Review Board (PRB) meeting.

The Problem Assessment Center will be responsible for coordinating all launch constraint activity and assuring that information is properly documented in the Problem Assessment System (PAS) data base and transmitted to MSFC management.

Robert E. Lindstrom
Manager
Shuttle Projects Office

Distribution:
See page 3

Distribution:

CS01/R. Benritze
CS11/J. Jordan/R. Mize/R. Penny/P. Moore
EA01/J. Kingsbury
EC01/F. Moore
EC12/A. Henry/E. Paschal/J. Shirey
EC14/W. Karigan/J. Burson/S. Boyle/J. Lewis/J. McEuen
EC22/E. Fikes/I. Morgan
EC23/J. Goldstein/T. Escue/E. Thompson/J. Douglass/
R. Holder/W. White
EC24/R. Bynum
EC25/C. Cornelius/C. Martin
EC33/L. Malone/D. Stone/L. Thomas/L. Bell/D. Harris
EC41/H. Garrett
EC42/D. Williams/J. Blanche/C. Souther/B. McPeak
ED01/G. Hopson
EE01/F. Wojtalik
EE11/W. Horton/R. Butler/W. Jackson/T. McKay/W. Mann/J. Roach/
J. Appling/R. Henry/J. Rowell/J. Rountree/V. Henson/
L. Sommers
EE21/W. Mitchell/J. Thomson/C. Smith
EE31/E. Bryan/B. Davis/L. Guzinsky/L. Mulloy/R. Zagrodzky
EE51/G. Eudy/F. Uptagrafft
EF01/J. Powell
EF12/J. Gattis
EF13/J. Clubb/M. Camp/C. Morris/M. Goode/B. Seiler
EF36/C. Cozelos
EG01/C. Brooks
EG03/E. Hendricks/J. Madole/E. Smith/E. Mintz/W. Barrell
EG21/G. Butler
EG22/T. Turner/R. Donaldson/H. Wiggins/R. Kelly/Q. Soprano/
R. Jackson/J. Gross/R. Riemer
EG23/E. Board/D. Herda/D. Gladish/A. Torruella
EG24/E. Lowe/J. Davis/E. Baker/B. Nunnelley
EH01/R. Schwinghamer
EL01/T. Isbell
EP01/A. McCool
EP13/C. Gaines/E. Engler/W. Earrison
EP14/H. Nein/R. White/R. Beck/R. Runkle/R. Dotson/W. Bell
EP21/O. Goetz/G. Smith/E. Jacobs
EP22/R. Rodgers
EP23/F. Pitsenberger/H. Stinson/C. Ratliff/D. Pryor/R. Counts
EP25/J. Hester/J. Niblett
EP31/G. Geller
EP32/M. Leberman
EP33/F. Cunningham/C. Boover/J. Potter/R. Hughes/J. Potter/
R. Burns/V. Neiland/W. Dunaway/N. Langson/L. Hein
EP35/T. Bechtel/F. Jankowski
EP43/E. Hyde/J. Pratt

Distribution: (Cont'd)

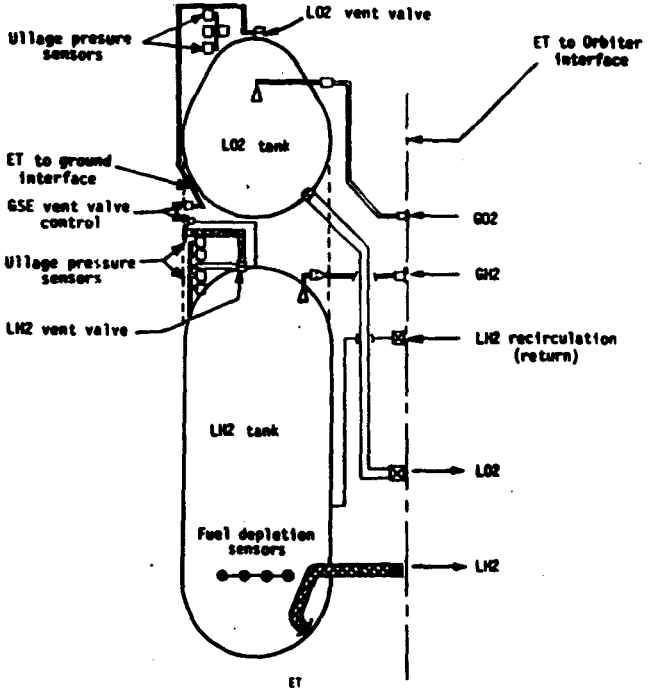
EP44/R. Fisher/J. Vaniman
ES81/W. Vaughan
ET01/E. Cagle
SA31/J. Odom/P. Bridwell
SA32/R. Abraham/C. Cavalaris/C. Crane/M. Pessin
SA32/D. Newman
SA41/W. Rice/G. Hardy
SA43/J. Strickland/D. Andrews
SA51/J. R. Thompson, Jr./L. Wear
SA52/M. Bryant/J. Eaton/B. Mitchell/B. Morris/J. Sloan/
J. Splawn/R. Weesner
SA49/Brigham City/E. Skrobiszewski
SM21/MAF/M. Marchese
SA99/Canoga Park/R. Tjulander
EG03/Canoga Park/B. Pagenkopf
SK-QAL/T. Bridges/J. Hughes/J. Cook
SK-SRB/J. King

Rockwell/Hsv/R. Strand/J. Fletcher/B. Cox/G. Miller/
W. McGahee/M. Glass/O. Burnett/R. Roberts/
J. Ash/C. Newton/R. Saliba/B. Gonterman/
P. Pollack

SECTION 5.1 EXTERNAL TANK SUBSYSTEMS

PCML ID Nos. T41, T56, and Ground Measurement Nos. GGO, GLH, GLO

PC 001853



5.1-3 REV C

PC 301354

MEAS. NO.	MEASUREMENT DESCRIPTION	CAT.	REDLINE MINIMUM	REDLINE MAXIMUM	UNITS	CODE	TIME PERIOD, ACTION, AND NOTES
T4117[illegible]	Nose cone comp temp 1	1 of 2 R	45	135	°F	99	From Prior to Start of Chilldown to T-3 Min
T4117[illegible]	Nose cone comp temp 1		43	135	°F	99	
			(1)			(A)	(1) Minimum may be exceeded for up to 15 minutes.
							For confirmed instrumentation loss switch to backup and continue (A).

BYPASSING, SCHEMATIC, OR FLOW DIAGRAM:

BIT VALUE	PCM RANGE LOW	PCM RANGE HIGH	METER RANGE LOW	METER RANGE HIGH	C AND W MIN	C AND W MAX	SM MIN	SM MAX
	-6	130						

REDLINE DERIVATION:
Redlines established for following reasons:
1. Minimum
   A. LH2 ullage pressure transducers calibrated to 60°F
   B. Avoid ice build-up at nose cone fairing exit
2. Maximum
   A. Protect CBF/LSC in cable tray area
   B. LO2 ullage pressure transducers Calibrated to 140°F
   C. SLA long term bond line exposure is 200°F

	Min	Max	
	40	140	Basic Redline
	5	5	ET Instr Error
	45	135	Redline

CONSEQUENCE OF EXCEEDING REDLINE:
1. Ice build-up and possible impact to Orbiter
2. Inaccurate ullage pressure readings
3. Possible detonation of CBF/LSC and loss of vehicle
4. Unbonding of SLA

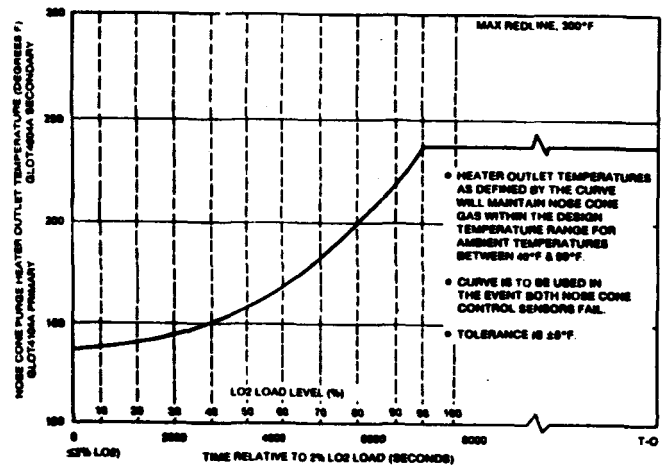
CAUSES OF EXCEEDING REDLINE:
1. Heater malfunction.
2. Regulator malfunction.
3. Improper purge flow.
4. Pneumatic valve is closed.
5. Clogged filter.

SPACE SHUTTLE SYSTEMS HANDBOOK	NAS NO.	TBD	SHEET	ZONE	TBD	NOTES:
	NAS NO.		SHEET	ZONE		(A) The following ground purge temps are backup measurements
	NAS NO.		SHEET	ZONE		GL0T4100A PRI nose cone heated purge temp — See curve on
INTEGRATED SYSTEMS SCHEMATIC:	NAS NO.		SHEET	ZONE		GL0T4100A SEC nose cone heated purge temp — page 6.1-43

EFFECTIVITY	CONTRACTS	CONTRACTOR: MCDONNELL, J. H. LANTERS	CHANGE CONTROL CONTROLS	CHANGE NO. REL. C REDLINE VER
ELEMENT: ET			AND BACKGROUND	PAGE 6.1-5
MISSION: STS 51-G & SUBS		COVERT: JSC, M. MANNACK, MSFC, M. PUDETA		

01424/0013A:mr

PC 001855



ET NOSE CONE PURGE HEATER OUTLET TEMPERATURE REQUIREMENT

PC 001857

SHUTTLE REQUIREMENTS EXCEPTION/WAIVER LIST STS —381 DATE:

NUMBER: WE0907
ELEMENT: ET (CLOU)
CHRSD TITLE: LCC PAGE 5.1-4/VERIFY ET NOSE CONE PAD(CHRSD)00F00-552
EFFECTIVITY: STS 6 & SUBS

REQT. NO.	DESCRIPTION	TECHNICAL RATIONALE
(PR # 3W-0147)	NOSE CONE TEMP. EXCEEDS THE LCC LOW LIMIT TEMP. 4/B 45°F. TEMP INDICATES 33°F. [redacted]	NO VISABLE ICE BUILDUP ON THE NOSE CAP OR FAIRING EXIT AREA. TEMPERATURE IS 12°F BELOW REQUIRE. FOR STS-33 MIN. LCC ALLEPTABLE IS 29°F (WAS 45°F) ULLAGE TRANSDUCERS ARE ALLEPIABLE DOWN TO 29°F (WAS 40°F) REFER. NOTE A LCC 5.1-4

PART NAME	DRAWING NUMBER	QTY & SERIAL NUMBER
ET NOSE CONE HEATER (ET ANTI-ICE PANEL) 372-0694-17	79K09194	1

5686

SUSPENSE DATE:
CONSTRAINT:
NSC FAILURE:
LEVEL III PROJECT OFFICE CONCURRENCE:
[illegible]:

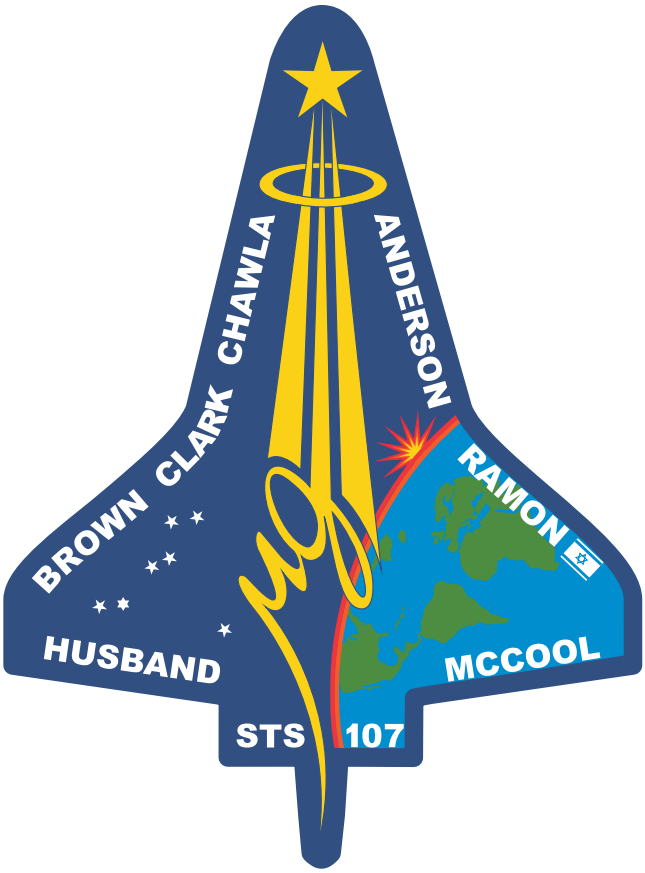
APPROVALS

REQUESTED BY: PE — 1/27/86
LSS — 1/27/86
LSOC SYS MGR — 1/27/86
LSOC PE — 1/27/86
DESIGN CENTER — S. Bennett 1/27/86; C. Houston 1-27-86
MSFC SYS MGR — 1-27-86; 1/27/86
MSFC QUA CNTRD — 1-27-86
MSFC PE (EXCEPTION ONLY) — 1-27-86

National Aeronautics and Space Administration

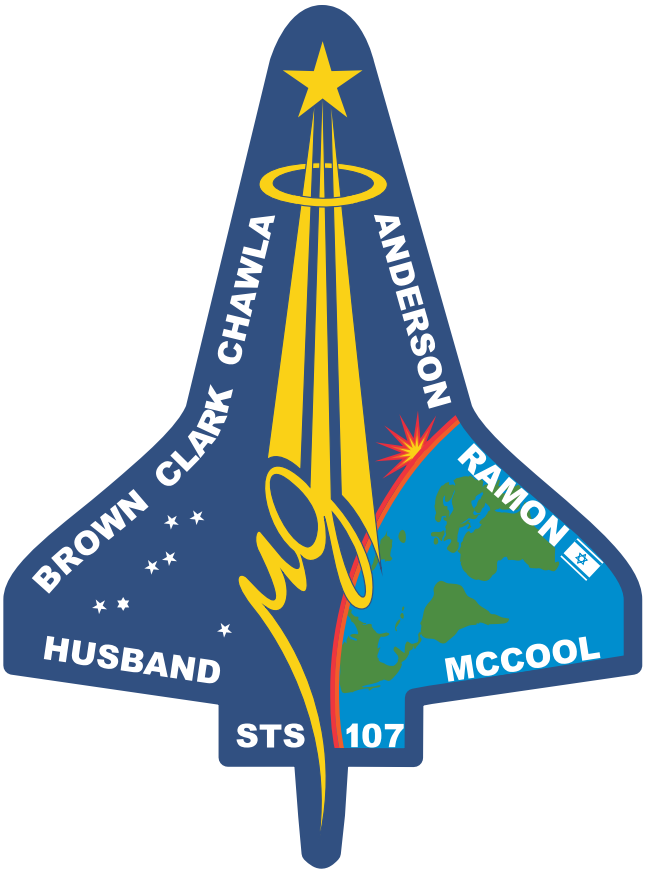
Columbia Crew Survival Investigation Report

NASA/SP-2008-565



Columbia Crew Survival Investigation Report

NASA/SP-2008-565



Contents

Executive Summary ..... **xix**
Background xix
Results xix
Improving Crew Survival Investigations xxi
Introduction ..... **xxii**
The *Columbia* Accident Investigation Board Report xxii
Summary of Conclusions and Recommendations xxiii
Lethal events xxiii
Crew survival accident investigation xxvi
Other xxvii
Summary xxvii
Report Format xxviii
Conclusions and Recommendations ..... **xxix**

1 Integrated Story

1.1 Integrated Investigation Results **1-1**
1.1.1 Events with lethal potential 1-1
1.1.2 Integrated summary of events 1-1
1.1.2.1 *Phase 1: Deorbit preparation to entry interface* 1-5
1.1.2.2 *Phase 2: Entry interface to loss of signal* 1-7
1.1.2.3 *Phase 3: Loss of signal to Catastrophic Event* 1-15
1.1.2.4 *Phase 4: Catastrophic Event to Crew Module Catastrophic Event* 1-20
1.1.2.5 *Phase 5: Crew Module Catastrophic Event to Total Dispersal* 1-26
1.1.2.6 *Phase 6: Total Dispersal to ground impact* 1-29
1.2 Master Timeline **1-30**
1.2.1 Phase 1: Deorbit preparation to entry interface 1-31
1.2.2 Phase 2: Entry interface to loss of signal 1-40
1.2.3 Phase 3: Loss of signal to Catastrophic Event 1-60
1.2.4 Phase 4: Catastrophic Event to Crew Module Catastrophic Event 1-72
1.2.5 Phase 5: Crew Module Catastrophic Event to Total Dispersal 1-75
1.2.6 Phase 6: Total Dispersal to ground impact 1-77

2 Vehicle Failure Assessment

2.1 Motion and Thermal Analyses **2-1**
2.1.1 Intact orbiter trajectory 2-3
2.1.2 Forebody trajectory 2-7
2.1.3 Intact orbiter attitude dynamics through the Catastrophic Event 2-9
2.1.4 Post-Catastrophic Event attitude dynamics of the forebody 2-15
2.1.5 Relative motion comparison 2-20
2.1.6 Thermal analyses introduction 2-24
2.1.6.1 *Thermal analysis – crew module* 2-25
2.1.6.2 *Thermal analysis – helmet* 2-30
2.1.6.3 *Thermal testing – boot soles* 2-30
2.1.6.4 *Thermal analysis – lithium hydroxide stowage volume door* 2-30
2.1.6.5 *Thermal analysis – payload and general support laptop computers* 2-35

2.1.6.6	*Thermal analysis – Modular Auxiliary Data System/orbiter experiment recorder*	2-35
2.1.6.7	*Thermal analysis – x-links*	2-37
2.1.6.8	*Thermal analysis – payload bay door roller*	2-41
2.1.7	Shock-shock interaction and combustion	2-43
2.1.8	Synopsis of motion and thermal analyses	2-46
2.2	**Orbiter Breakup Sequence**	**2-47**
2.2.1	Ground-based video analysis	2-48
2.2.1.1	*Significant events seen in the video*	2-49
2.2.2	Ballistic analysis	2-54
2.2.2.1	*Major structural release times*	2-56
2.2.3	Cluster analysis	2-58
2.2.3.1	*Debris clusters*	2-58
2.2.4	Structural analysis	2-65
2.2.4.1	*Background*	2-65
2.2.4.2	*Forebody structure prior to the Catastrophic Event*	2-68
2.2.4.3	*Payload bay doors*	2-69
2.2.4.4	*Forebody separation event*	2-71
2.2.5	Synopsis of orbiter breakup sequence	2-87
2.3	**Crew Cabin Pressure Environment Analysis**	**2-88**
2.3.1	Depressurization timeline boundaries	2-88
2.3.2	Start of depressurization	2-89
2.3.3	Depressurization rate	2-90
2.3.4	Depressurization complete	2-91
2.3.5	Location(s) of the cabin breach(es)	2-91
2.3.6	Synopsis of crew cabin pressure environment analysis	2-93
2.4	**Forebody Breakup Sequence**	**2-94**
2.4.1	Ground-based video analysis	2-95
2.4.2	Ballistic analysis	2-98
2.4.3	Cluster analysis	2-100
2.4.4	Structural analysis	2-111
2.4.4.1	*Events at the Catastrophic Event*	2-111
2.4.4.2	*Crew Module Catastrophic Event/forebody breakup*	2-122
2.4.4.3	*Orbiter window analysis*	2-133
2.4.5	Synopsis of forebody breakup sequence	2-139

3 Occupant Protection

3.1	**Crew Seats**	**3-1**
3.1.1	Seat design and construction	3-2
3.1.2	STS-107 seat configuration	3-5
3.1.3	Seat structure	3-6
3.1.4	Upper seatbacks/inertial reels	3-14
3.1.5	Inertial reel straps	3-19
3.1.6	Inertial reel locking mechanisms	3-27
3.1.7	Belt adjusters	3-30
3.1.8	Restraint buckle	3-32
3.1.9	Sequence	3-34
3.1.10	Lesson learned – equipment serialization and marking	3-34
3.2	**Crew Worn Equipment**	**3-36**
3.2.1	Shuttle suits	3-38
3.2.1.1	*Advanced crew escape suit*	3-39
3.2.1.2	*Parachute harness*	3-42
3.2.1.3	*Personal parachute assembly*	3-43
3.2.2	Crew worn equipment configuration	3-44
3.2.3	Aircraft in-flight breakup case studies	3-44
3.2.4	Recovered debris	3-46

3.2.5	Helmets	3-50
3.2.5.1	*General condition*	3-50
3.2.5.2	*Thermal condition*	3-51
3.2.5.3	*Mechanical condition*	3-52
3.2.6	Glove disconnects	3-54
3.2.7	Dual suit controllers	3-55
3.2.8	Boots	3-56
3.2.9	Emergency Oxygen System	3-57
3.2.10	Seawater Activated Release System	3-58
3.2.11	Telonics Satellite Uplink Beacon-A search and rescue satellite-aided tracking beacon	3-59
3.2.12	Army/Navy personal radio communications-112 radio	3-59
3.2.13	Ground plot analysis	3-60
3.2.14	Lessons learned	3-62
3.2.14.1	*Equipment serialization and marking*	3-62
3.2.14.2	*Suit requirements and design*	3-63
3.3	**Crew Training**	**3-64**
3.3.1	Overview	3-64
3.3.2	*Columbia* crew training	3-67
3.3.3	Analysis and discussion	3-67
3.3.4	Training effectiveness case study	3-68
3.4	**Crew Analysis**	**3-71**
3.4.1	Crew awareness	3-73
3.4.1.1	*Preflight*	3-73
3.4.1.2	*Launch*	3-73
3.4.1.3	*Orbital operations*	3-73
3.4.1.4	*Deorbit preparations*	3-74
3.4.1.5	*Entry*	3-75
3.4.1.6	*Loss of control*	3-78
3.4.2	Injury classifications	3-83
3.4.2.1	*Exposure to high altitude*	3-83
3.4.2.2	*Mechanical injuries*	3-85
3.4.2.3	*Thermal exposure*	3-89
3.4.3	Identified events with lethal potential	3-89
3.4.4	Synopsis of crew analysis	3-90
4	**Investigative Methods and Processes**	**4-1**
4.1	**Background**	**4-3**
4.2	**Spacecraft Crew Survival Integrated Investigation Team Structure and Personnel**	**4-4**
4.2.1	Team membership	4-5
4.3	**Investigative Process**	**4-9**
4.3.1	Public release of information	4-11
4.3.2	Using members of affected organizations in the investigation	4-12
4.4	**Medical Process Issues**	**4-12**
4.5	**Analysis Methods, Processes, and Tools**	**4-13**
4.5.1	*Columbia* debris repository	4-13
4.5.2	Physical reconstruction	4-13
4.5.3	Virtual reconstruction	4-14
4.5.4	Motion analysis tools	4-16
4.5.4.1	*Trajectory and attitude analyses*	4-16
4.5.4.2	*Ballistic analysis*	4-19
4.5.4.3	*Thermal analysis*	4-19
4.5.5	Video analysis	4-21
4.5.5.1	*Ground-based video analysis*	4-21
4.5.5.2	*Forebody triangulation*	4-23

4.5.6	Debris mapping/plots	4-23
4.5.7	Structural analysis	4-25
4.5.7.1	*Structural capability*	4-26
4.5.7.2	*Crew seats*	4-27
4.5.8	Materials analysis	4-27
4.5.8.1	*Crew seats*	4-28
4.5.8.2	*Boots materials thermal testing*	4-28
4.5.8.3	*Helmets*	4-28
4.5.9	Medical processes	4-28
4.5.9.1	*Crew event timeline*	4-28
4.5.9.2	*Cause of death determination*	4-29
4.5.10	Other methods and processes	4-30
4.5.10.1	*Hygiene/drink package depressurization testing*	4-30
4.5.10.2	*Advanced crew escape suits and crew worn equipment*	4-30
4.5.10.3	*Helmets*	4-30
4.5.11	Cabin depressurization modeling	4-31

Future Work

Appendix: Ballistic Tutorial

Figures

1.1-1	Depiction of the orbiter forebody, midbody, and aftbody elements	1-2
1.1-2	Depiction of the crew module flight deck and middeck within the forebody	1-2
1.1-3	Depiction of the flight deck seats	1-3
1.1-4	Depiction of the middeck seats	1-3
1.1-5	Overall timeline	1-4
1.1-6	Phase 1 timeline with key events	1-5
1.1-7	This early image from the recovered middeck video shows progress of the crew in deorbit preparation tasks	1-6
1.1-8	Phase 2 timeline with key events	1-7
1.1-9	The effects of wing Thermal Protection System damage as reported by the *Columbia* Accident Investigation Board	1-8
1.1-10	Negative trend starts at GMT 13:51:46 (EI+457) and exceeds flight experience at GMT 13:53:38 (EI+569)	1-9
1.1-11	The sharp divergence from previous flight experience starts at approximately GMT 13:58:19 (EI+850)	1-10
1.1-12	This is the first frame of ground-based video	1-11
1.1-13	Video capture of the first observed incident of debris being shed	1-11
1.1-14	The brightest debris event known to have occurred prior to loss of signal	1-11
1.1-15	Aileron trim discrepancy	1-12
1.1-16	The Littlefield tile	1-12
1.1-17	Landing gear indicator panel, identical on both sides of the flight deck forward display panels	1-13
1.1-18	Reaction Control System thruster status display	1-14
1.1-19	Phase 3 timeline with key events	1-15
1.1-20	Sequence (1-second intervals) showing a simulation of orbiter loss of control pitch-up from GMT 13:59:37 (EI+928) to GMT 13:59:46 (EI+937)	1-17
1.1-21	Video frame captures from ground-based video, beginning at GMT 13:59:35.5 (EI+926) and ending at GMT 13:59:43.5 (EI+937)	1-18
1.1-22	Resultant acceleration (G) vs. time prior to Catastrophic Event in the *crew* coordinate frame from the aerodynamic simulation model	1-19
1.1-23	Phase 4 timeline with key events	1-20
1.1-24	The Catastrophic Event is depicted in these three frames of video that cover 0.1 second	1-21
1.1-25	Port x-link, debris item no. 1678, and starboard x-link, debris item no. 1765, from the *Columbia* Reconstruction Database	1-22
1.1-26	Scenario showing how the crew module pressure vessel could impact the forward fuselage, and the middeck Volume E could impact the crew module pressure vessel, with resultant damage	1-22
1.1-27	Estimated change in total G experienced by the forebody due to a change in ballistic number from the orbiter breakup	1-23

1.1-28	X-axis center-of-gravity locations for the intact orbiter, the crew module, the forward fuselage, and the forebody (crew module plus forward fuselage)	1-24
1.1-29	Predicted acceleration and thermal data for an idealized trajectory of the intact orbiter with a ballistic number of 42 pounds per square foot, and post-Catastrophic Event free-flying forebody with a ballistic number of 122 pounds per square foot	1-26
1.1-30	Video frame from the Apache video at GMT 14:00:55 (EI+1006) showing two similar luminosity objects separating from the forebody	1-27
1.1-31	Middeck floor debris in original relative orientation showing varying thermal exposure	1-28
1.2-1	Start of the recovered middeck video	1-32
1.2-2	Middeck configuration during duration of the recovered video	1-32
1.2-3	End of the recovered middeck video	1-33
1.2-4	Pilot/Seat 2 adjusting g-suit setting	1-36
1.2-5	Mission Specialist 4/Seat 3 donning glove	1-36
1.2-6	Location of Commander's seat and rotational hand controller	1-37
1.2-7	Digital autopilot button on the C3 panel is illuminated	1-37
1.2-8	Digital autopilot is no longer in AUTO, INRTL light is ON; Commander/Seat 1 restores digital autopilot to AUTO, INRTL light is OFF	1-38
1.2-9	Location of the "INRTL" button that illuminated when the digital autopilot moded out of AUTO, and the "AUTO" button that the Commander pressed to restore the digital autopilot to AUTO	1-38
1.2-10	Commander/Seat 1 inputting an ITEM 27	1-39
1-2-11	Entry keypad on the C2 panel	1-39
1.2-12	Pilot/Seat 2 observed with gloves ON and MATED	1-40
1.2-13	Commander/Seat 1 visor down and latched and Mission Specialist 2/Seat 4 with helmet on and left glove off	1-40
1.2-14	Mission Specialist 2/Seat 4 donning left glove; Commander/Seat 1 and Pilot/Seat 2 with gloves ON and MATED	1-41
1.2-15	Mission Specialist 4/Seat 3 right glove still ON and MATED	1-42
1.2-16	Last discernable image from recovered flight deck video	1-42
1.2-17	Attitude of *Columbia* at the start of the first planned roll	1-43
1.2-18	Attitude of *Columbia* at the end of the first planned roll	1-43
1.2-19	Location of the yaw jet indicator light	1-44
1.2-20	STS-107 sideslip angle vs. time	1-45
1.2-21	*Columbia* approaching the California coastline	1-46
1.2-22	This is the first frame of ground-based video	1-47
1.2-23	*Columbia* crossing the California coastline	1-47
1.2-24	Video capture of the first observed incident of debris being shed	1-48
1.2-25	Elevon trim display	1-48
1.2-26	*Columbia* crossing the California-Nevada border	1-49
1.2-27	Plot of main oxygen supply pressure showing second series of suit pressure checks	1-49
1.2-28	The brightest debris event known to have occurred prior to loss of signal	1-50
1.2-29	Attitude at the start of the roll reversal	1-51
1.2-30	Attitude at the completion of the roll reversal	1-51
1.2-31	Infrared photograph of the shuttle during entry	1-52
1.2-32	Enhanced video captures showing the first observed flare in the trail	1-53
1.2-33	Enhanced video captures showing the second observed flare in the trail	1-54
1.2-34	The sharp divergence from previous flight experience starts at approximately GMT 13:58:19 (EI+850)	1-55

1.2-35	The Littlefield tile	1-56
1.2-36	Landing gear indicator panel, identical on both sides of the flight deck forward display panels	1-58
1.2-37	Reaction Control System thruster status display	1-59
1.2-38	Illustration of the last observed elevon deflections	1-60
1.2-39	Location of the master alarm light	1-62
1.2-40	Sequence (1-second intervals) showing a simulation of orbiter loss of control pitch-up from GMT 13:59:37 (EI+928) to GMT 13:59:46 (EI+937)	1-64
1.2-41	Video frame captures from ground-based video, beginning at GMT 13:59:35.5 (EI+926.5) and ending at GMT 13:59:43.5 (EI+937.5)	1-65
1.2-42	Debris D is depicted in this frame	1-67
1.2-43	The Catastrophic Event is depicted in these three frames of video that cover 0.1 second	1-72
1.2-44	Simulation of the Catastrophic Event	1-72
1.2-45	X-axis center-of-gravity locations for the intact orbiter, the crew module, the forward fuselage, and the forebody (crew module plus forward fuselage)	1-74
1.2-46	Color-inverted video images of the start of the double star event, GMT 14:00:26.6 (EI+977.6)	1-75
1.2-47	Image from Apache video showing crew module debris and the three main engines	1-76
1.2-48	Image is from the Apache video	1-77
2.1-1	Depiction of the body axis coordinate system and pitch, yaw, and roll definitions	2-3
2.1-2	Depiction of angle of attack	2-3
2.1-3	Depiction of sideslip	2-4
2.1-4	Entry guidance drag-velocity profile and limits	2-5
2.1-5	Noticeable change in brightness at GMT 13:59:37±1 second	2-6
2.1-6	Noticeable debris shedding event at GMT 13:59:39±1 second	2-6
2.1-7	Sequence (1-second intervals) showing a simulation of orbiter loss of control pitch-up from GMT 13:59:37 to GMT 13:59:46	2-7
2.1-8	Comparison of 150-pounds-per-square-foot and 122-pounds-per-square-foot forebody projected trajectories	2-8
2.1-9	Estimated change in total G experienced by the forebody due to a change in ballistic number from the orbiter breakup	2-9
2.1-10	STS-107 simulated angle of attack after loss of signal	2-10
2.1-11	STS-107 simulated roll angle after loss of signal	2-11
2.1-12	STS-107 simulated sideslip angle after loss of signal	2-11
2.1-13	Entry simulation snapshot sequence	2-12
2.1-14	Representation of resultant body axis loads imparted from aerodynamic drag in nominal entry attitude	2-13
2.1-15	Representation of change in resultant body axis loads imparted from aerodynamic drag in off-nominal attitude	2-13
2.1-16	Estimated accelerations at the crew module in the X axis from the loss of signal to the Catastrophic Event	2-14
2.1-17	Estimated accelerations at the crew module in the Y axis from the loss of signal to the Catastrophic Event	2-14
2.1-18	Estimated accelerations at the crew module in the orbiter Z axis from the loss of signal to the Catastrophic Event	2-15
2.1-19	Video-capture showing *Challenger* crew module pointed nose into the direction of travel	2-16
2.1-20	Forward fuselage/crew module (forebody) configuration	2-16
2.1-21	Pitch stable attitudes for the forebody	2-16

2.1-22	Contour plot of pitch, roll, and yaw moments for a crew module showing no stability due to slight offset in lateral center of gravity	2-17
2.1-23	Modeled trailing wire configuration for forebody	2-18
2.1-24	Free-flying forebody roll rates, with and without initial rates	2-18
2.1-25	Free-flying forebody pitch rates, with and without initial rates	2-19
2.1-26	Free-flying forebody yaw rates, with and without initial rates	2-19
2.1-27	Accelerations at a representative seat location	2-20
2.1-28	Depiction of the triangulation tool	2-21
2.1-29	Relative motion between the forebody and the main engines	2-22
2.1-30	Fast-Fourier transform of relative motion in X, Y, and Z axes in hertz	2-23
2.1-31	Altitude vs. velocity for selected trajectory profiles	2-27
2.1-32	Crew module stress vs. temperature	2-28
2.1-33	Recovered lithium hydroxide door with some seat legs still attached	2-30
2.1-34	Example of an intact lithium hydroxide door with seats 6 and 7 attached	2-30
2.1-35	Upper surface of the lithium hydroxide door, facing the middeck	2-31
2.1-36	Top surface of the recovered lithium hydroxide door, facing the middeck	2-31
2.1-37	Lower surface of the lithium hydroxide door, facing the lower equipment bay	2-31
2.1-38	Lower surface of the recovered lithium hydroxide door, facing the lower equipment bay	2-31
2.1-39	Aerodynamic model of the lithium hydroxide door exposed by itself at an angle of attack of 90 degrees	2-32
2.1-40	Pressure distribution of the lithium hydroxide door and seats at an angle of attack of 96 degrees	2-33
2.1-41	Pressure distribution of the lithium hydroxide door and seats at an angle of attack of –92 degrees	2-33
2.1-42	Geometry of a single suited occupant on the lithium hydroxide door	2-34
2.1-43	Pressure distribution of the lithium hydroxide door and seats with suited crew members at an angle of attack of 122 degrees	2-34
2.1-44	Recovered orbiter experiment recorder from STS-107	2-35
2.1-45	Temperature vs. time of the orbiter experiment recorder shell if released at GMT 14:01:02	2-36
2.1-46	Drawings of pristine port and starboard x-links	2-37
2.1-47	Comparison of port and starboard x-links recovered from *Columbia*	2-37
2.1-48	Comparison of port and starboard x-link flanges	2-38
2.1-49	Close-up of the hole in the starboard x-link	2-38
2.1-50	Specific heat of titanium vs. temperature	2-39
2.1-51	Heating rates required to ablate a titanium hole of 0.25 inch in depth	2-40
2.1-52	Payload bay door roller showing inner material and Inconel sleeve	2-41
2.1-53	Recovered payload cylinder bay door roller with inner material absent and Inconel sleeve remaining	2-41
2.1-54	Computational fluid dynamics analysis of heating at the tip of the payload bay door roller for orthogonal geometry into the direction of travel at M = 10.5	2-42
2.1-55	Comparison of the object reentry survival analysis tool and computational fluid dynamics heating rates for the front face of the payload bay roller for various Mach numbers	2-43
2.1-56	Schlieren photograph of the orbiter showing a shock-shock interaction region for nominal entry	2-43
2.2-1	First frame of video of the Eastern timeline, GMT 13:59:32.5	2-50
2.2-2	GMT 13:59:37.00	2-50
2.2-3	GMT 13:59:37.43, 13 frames after figure 2.2-2	2-50
2.2-4	GMT 13:59:37.80	2-50

2.2-5	The Catastrophic Event is depicted in these three frames of video that cover 0.1 second	2-51
2.2-6	Split trail after the Catastrophic Event	2-51
2.2-7	Persistence of the split trail 23 seconds after the Catastrophic Event	2-51
2.2-8	Color-inverted video images of the start of the double star event (GMT 14:00:26.6)	2-52
2.2-9	Debris tree	2-53
2.2-10	Clear visual separation of the three main engines	2-54
2.2-11	A cannonball is an example of a ballistic object	2-55
2.2-12	Depiction of the orbiter forebody, midbody, and aftbody elements	2-58
2.2-13	Debris field of the left and right wings	2-59
2.2-14	Partial debris field of the left wing, right wing, and tail	2-60
2.2-15	Histogram of the left wing, right wing, and tail debris	2-60
2.2-16	Partial debris field of the left wing, right wing, midbody, and tail	2-61
2.2-17	Histogram of the percentage of debris found vs. longitude for the left wing, right wing, midbody, and tail	2-61
2.2-18	Longitude vs. the X_o origin of the structural debris of the mid fuselage	2-63
2.2-19	Partial debris field of the left wing, right wing, and aftbody	2-63
2.2-20	Histogram of debris of the wings and aftbody (including orbiter main engines)	2-64
2.2-21	Histogram of the relative order of the major debris groups recovered	2-65
2.2-22	Forward fuselage structure	2-66
2.2-23	Thirteen links to the crew module inside the forward fuselage	2-67
2.2-24	Crew module support links, starboard-side view	2-67
2.2-25	Picture of the X_o 576 bulkhead and the X_o 582 ring frame bulkhead relative to each other	2-68
2.2-26	Resultant acceleration (G) vs. time prior to the Catastrophic Event in the *orbiter* coordinate frame	2-68
2.2-27	Forward fuselage debris field	2-69
2.2-28	Payload bay door debris footprint relative to the main crew module debris field	2-70
2.2-29	Payload bay door debris footprint relative to the payload bay structure	2-70
2.2-30	Recovered X_o 576 bulkhead of the *Challenger* crew module showing a portion of the X_o 582 ring frame bulkhead	2-71
2.2-31	Close-up of the starboard sill gusset area	2-72
2.2-32	Forward fuselage skin still attached to the starboard hoist fitting	2-73
2.2-33	Thermal erosion on the aft end of the starboard-side skin, *Columbia* Reconstruction Database debris item no. 2436	2-74
2.2-34	Forward port hoist fitting, *Columbia* Reconstruction Database debris item no. 32038	2-74
2.2-35	Bay 1 midbody bottom skin, *Columbia* Reconstruction Database debris items nos. 2429 and 33852	2-75
2.2-36	Bay 1 midbody skin, *Columbia* Reconstruction Database debris item no. 88290	2-75
2.2-37	Bay 1 midbody skin, *Columbia* Reconstruction Database debris item no. 14908	2-76
2.2-38	Port and starboard x-links	2-77
2.2-39	View looking forward at the starboard x-link and side skin	2-77
2.2-40	Detail of thermal erosion on the upper bolts at the starboard x-link and skin	2-78
2.2-41	Weak links – sill joint gusset, starboard side	2-78
2.2-42	Structural loading from starboard x-link to sill longeron	2-79
2.2-43	Hot gas flow heats up starboard sill and the adjacent x-link	2-79
2.2-44	Forebody departed the midbody at the Catastrophic Event	2-80

2.2-45	How the port x-link failed, cross-section view looking down	2-81
2.2-46	Views of the port x-link with about a 0.10-inch yield on the outboard side of the clevis	2-81
2.2-47	*Columbia* Reconstruction Database debris item no. 2207, which includes the remainder of the lug fitting that supports the aft end of the port x-link	2-82
2.2-48	View of the port x-link and a portion of the X_o 582 ring frame bulkhead debris items, looking aft	2-83
2.2-49	View looking up and aft at the port x-link	2-83
2.2-50	Tunnel adaptor assembly and middeck access panel	2-84
2.2-51	Tunnel adapter assembly and the manufacturing access panel debris, view looking forward of "B" hatch opening	2-85
2.2-52	Tunnel adapter assembly debris showing that the bolt shanks failed mainly by tension and bending	2-85
2.2-53	Cross section through tunnel adapter assembly mounting bolt	2-86
2.2-54	Predicted motion based on debris evidence	2-86
2.2-55	Airlock in the middeck	2-87
2.2-56	View looking forward at the "B" hatch opening on the X_o 576 bulkhead	2-87
2.3-1	Cabin depressurization timeline	2-91
2.3-2	Plot from a cabin depressurization model showing a 12.4-inch-diameter hole above the middeck floor	2-92
2.3-3	Plot from a cabin depressurization model showing a 4.8-inch-diameter hole below the middeck floor	2-92
2.4-1	Five time-paired images covering under 2 seconds of time showing the Crew Module Catastrophic Event	2-96
2.4-2	Apache video of the Total Dispersal, spanning from GMT 14:01:06.73 to GMT 14:01:09.67	2-97
2.4-3	An analyzed frame from the Apache video	2-98
2.4-4	Debris field of the forward fuselage and crew module pressure vessel	2-100
2.4-5	Debris field of the nonstructural items from inside the crew module pressure vessel	2-102
2.4-6	Schematic of the orbiter three-paned window	2-102
2.4-7	Window numbering system	2-103
2.4-8	Recovered debris field for window glass	2-104
2.4-9	Debris field of the window panes, forward fuselage, and crew module pressure vessel	2-104
2.4-10	Debris field of the forebody components	2-105
2.4-11	Forward fuselage debris field	2-105
2.4-12	Debris field of the middeck and the flight deck	2-106
2.4-13	Middeck structural debris port vs. starboard	2-107
2.4-14	Flight deck panels by overhead, port, and starboard	2-107
2.4-15	Debris field of the airlock structure, flight crew equipment stowed in the airlock, and items bonded to airlock structure	2-108
2.4-16	Debris field of the forward fuselage, pressure vessel, and airlock structure	2-108
2.4-17	Debris field of the middeck, flight deck, and airlock	2-108
2.4-18	Forebody bulkheads	2-109
2.4-19	Debris field of the key forebody bulkheads	2-109
2.4-20	Debris field of the crew module bulkheads, middeck, and flight deck	2-110
2.4-21	Debris field of the avionics bays	2-110
2.4-22	Histogram of the forebody structural components, plus SPACEHAB	2-111

2.4-23	Nose cap *Columbia* Reconstruction Database debris item no. 1114 vs. the original nose cap with chin panel	2-112
2.4-24	Two large forward Reaction Control System debris pieces, *Columbia* Reconstruction Database debris item nos. 792 and 82061	2-112
2.4-25	Crew module x-links, *Columbia* Reconstruction Database debris item nos. 1678 and 1765	2-113
2.4-26	Starboard-side skin, view looking inboard, *Columbia* Reconstruction Database debris item no. 2436	2-114
2.4-27	Inboard side of the starboard-side skin, *Columbia* Reconstruction Database debris item no. 2436	2-115
2.4-28	View looking from the top at the y-links	2-116
2.4-29	The y-link debris, still attached to the mounting on the X_o 576 bulkhead, *Columbia* Reconstruction Database debris item nos. 72770 and 9280	2-116
2.4-30	The z-link debris photographs, *Columbia* Reconstruction Database debris item no. 53828	2-117
2.4-31	View of middeck floor and Volume E, looking down and aft	2-118
2.4-32	Volume E debris damage	2-118
2.4-33	Scenario showing how the crew module pressure vessel could impact the forward fuselage, and the middeck Volume E could impact the crew module pressure vessel, with resultant damage	2-119
2.4-34	Forward fuselage debris field showing tile, nose landing gear door structure, and star tracker door structure west of the main forebody debris field	2-120
2.4-35	Crew module pressure vessel skin lightweight design	2-121
2.4-36	Typical crew module pressure vessel skin in middeck, looking outboard	2-121
2.4-37	Recovered crew module pressure vessel base skin	2-122
2.4-38	Forward fuselage arrowhead panel and the forward two windows, facing in	2-122
2.4-39	Low-heat damage on the forward fuselage arrowhead panel, facing out, *Columbia* Reconstruction Database debris item no. 65049	2-123
2.4-40	Lower forward fuselage skin and frame debris	2-123
2.4-41	Lower forward fuselage skin panel near external tank attachment	2-124
2.4-42	View looking forward of the recovered crew module X_{cm} 200 bulkhead debris	2-125
2.4-43	Top view of the crew module middeck floor debris	2-126
2.4-44	Bottom view of the recovered middeck floor items from virtual reconstruction	2-126
2.4-45	Top view of the crew module flight deck debris	2-127
2.4-46	View looking aft of the recovered X_o 576 bulkhead debris	2-127
2.4-47	Middeck accommodations rack	2-128
2.4-48	Recovered middeck accommodations rack structure	2-129
2.4-49	Main middeck accommodations rack structure, as found	2-129
2.4-50	Crew escape pole in launch/landing position, middeck, looking aft	2-130
2.4-51	Port end of the crew escape pole, showing lanyard magazine, middeck, looking forward	2-130
2.4-52	Pole housing, main and extension poles	2-131
2.4-53	Port ends of housing, deployment spring, and main and extension poles	2-131
2.4-54	Crew escape pole in launch/landing position, middeck, looking aft, with direction of cantilever load noted from debris	2-132
2.4-55	Intact orbiter flight deck from the Shuttle Mission Simulator	2-132
2.4-56	Virtual reconstruction of the recovered *Columbia* flight deck panels	2-133
2.4-57	Port view looking aft, recovered *Columbia* crew module Windows 1, 2, and 3	2-133
2.4-58	Starboard view looking aft, recovered *Columbia* crew module Windows 4, 5, and 6	2-133

2.4-59	View looking forward, recovered *Columbia* crew module Windows 7, 9, and 10 and forward fuselage Window 8	2-134
2.4-60	Window fragment removed from the thermal pane frame from Window 8	2-135
2.4-61	*Endeavour*, OV-105, X_o 582 ring frame bulkhead arch with rollers	2-137
2.4-62	Nominal configuration (*Endeavour*, OV-105) of the two inner rollers and overhead windows	2-137
2.4-63	*Columbia* debris for same location	2-137
2.4-64	Titanium deposit on windows indicates forebody traveled backwards	2-138
3.1-1	Depiction of the flight deck seats	3-2
3.1-2	Depiction of the middeck seats	3-3
3.1-3	Pilot seat	3-4
3.1-4	Mission Specialist seat	3-4
3.1-5	Commander's seat and one of the rotational hand controllers, flight deck	3-5
3.1-6	Identified debris from the flight deck seats	3-6
3.1-7	Identified debris from the middeck seats	3-7
3.1-8	Debris from unknown seat locations	3-7
3.1-9	Recovery locations of seat structure debris	3-8
3.1-10	Comparison of seat debris from a flight deck Mission Specialist seat and a middeck Mission Specialist seat	3-9
3.1-11	Stable attitude of the lithium hydroxide door/seat 6/seat 7 combination, as indicated by deposition	3-10
3.1-12	Common seat failure locations	3-10
3.1-13	Example of an intact lithium hydroxide door with seats 6 and 7 attached	3-11
3.1-14	Recovered lithium hydroxide door with seats 6 and 7 legs attached	3-11
3.1-15	Failure force of seat forward leg and floor fitting vs. temperature	3-12
3.1-16	Example of a "broom-straw" fracture on a *Columbia* seat leg	3-13
3.1-17	Scanning electron microscope image of broom-straw fracture surface cross sections	3-13
3.1-18	Upper seatback debris item location	3-14
3.1-19	Upper seatbacks	3-15
3.1-20	Inertial reel mounting location	3-16
3.1-21	Seat 3 inertial reel, looking aft	3-17
3.1-22	Upper seatback inertial reel strap pass-through slots	3-18
3.1-23	Inertial reel melted strap material	3-19
3.1-24	Close-up of strap melt pattern	3-19
3.1-25	Example of linear waves on inertial reel straps	3-21
3.1-26	Seat strap failures: seat 1 and seat 2	3-24
3.1-27	Seat strap pass-through slot (intact seat shown)	3-24
3.1-28	Seat 3 inertial reel strap	3-25
3.1-29	Seat 4 inertial reel strap	3-25
3.1-30	Seat 6 inertial reel strap	3-26
3.1-31	Seat 7 inertial reel strap	3-26
3.1-32	Inertial reel mechanism	3-27
3.1-33	Inertial reel locking gear witness mark	3-28
3.1-34	Circular witness marks on inertial reels from a training seat, a flight (non-*Columbia*) seat, and a new (unused) flight-qualified inertial reel	3-29
3.1-35	Separation gap on the seat 4 inertial reel	3-29
3.1-36	Seat belt adjusters and restraint buckle	3-30
3.1-37	Recovered belt adjuster rear view	3-31
3.1-38	Fractured belt adjuster	3-31
3.1-39	Demonstration of the five-point seatbelt buckle	3-32
3.1-40	Recovered five-point seatbelt buckle	3-32

3.1-41	Indentations on latching pins, looking from center of buckle towards outer edge	3-32
3.1-42	Belt tongue positions	3-33
3.2-1	Crew worn equipment: advanced crew escape suit, harness, parachute, and survival gear	3-38
3.2-2	Example of neck ring	3-39
3.2-3	Example of wrist ring	3-39
3.2-4	Example of helmet	3-40
3.2-5	Example of neck ring latch	3-40
3.2-6	Example of gloves	3-40
3.2-7	Example of boots	3-41
3.2-8	Example of g-suit	3-41
3.2-9	Example of liquid cooling garment	3-41
3.2-10	Example of Army/Navy personal radio communications-112 radio	3-42
3.2-11	Example of parachute harness	3-42
3.2-12	Example of Emergency Oxygen System	3-43
3.2-13	Example of personal parachute assembly	3-43
3.2-14	Example of search and rescue satellite-aided tracking beacon	3-44
3.2-15	Seat 1 recovered crew worn equipment	3-48
3.2-16	Seat 2 recovered crew worn equipment	3-48
3.2-17	Seat 3 recovered crew worn equipment	3-49
3.2-18	Seat 4 recovered crew worn equipment	3-49
3.2-19	Seat 5 recovered crew worn equipment	3-49
3.2-20	Seat 6 recovered crew worn equipment	3-49
3.2-21	Seat 7 recovered crew worn equipment	3-50
3.2-22	Delamination and pitting damage	3-51
3.2-23	Helmet pressure visor thermal effects	3-52
3.2-24	Hold-down cable guide tube	3-53
3.2-25	Examples of intact suit-side and glove-side glove disconnects	3-54
3.2-26	Recovered suit-side, glove-side, and mated glove disconnects	3-55
3.2-27	Example of an intact dual suit controller	3-55
3.2-28	Recovered dual suit controller front and back	3-56
3.2-29	Recovered whole Emergency Oxygen System bottle/reducers and a bottle fragment	3-57
3.2-30	Seawater Activated Release System with Frost fittings (male/female) still mated, with a segment of harness strap attached	3-58
3.2-31	Directional melting on Seawater Activated Release System/ parachute harness strap	3-58
3.2-32	External and internal views of a recovered search and rescue satellite-aided tracking beacon	3-59
3.2-33	External and internal views of a recovered Army/Navy personal radio communications-112 radio	3-60
3.2-34	Relative location of helmets, search and rescue satellite-aided tracking beacons, and Army/Navy personal radio communications-112 radios	3-61
3.2-35	Relative locations of all crew escape equipment – flight deck crew members and middeck crew members	3-62
3.3-1	Loss of control/breakup cue card as flown on STS-107	3-66
3.3-2	*Columbia* recovered flight deck forward panels	3-68
3.3-3	Recovered R2 panel from *Columbia*	3-69
3.3-4	Recovered R2 panel from *Columbia* (after it was unfolded)	3-69
3.4-1	Depiction of the flight deck seats	3-74
3.4-2	Depiction of the middeck seats	3-74
3.4-3	Video frame-capture from the recovered middeck video	3-75
3.4-4	Video frame-capture from the recovered flight deck video	3-75
3.4-5	Location of the Commander’s seat and the rotational hand controller	3-76

3.4-6	Landing gear indicator panel, identical on both sides of the flight deck forward display panels	3-77
3.4-7	Sequence (1-second intervals) showing a simulation of orbiter loss of control pitch-up from GMT 13:59:37 to GMT 13:59:46	3-78
3.4-8	Depiction of the orbiter and crew member axes conventions	3-79
3.4-9	Representative accelerations from modeled motion analysis from loss of control to the Catastrophic Event	3-79
3.4-10	Location of the R2 panel	3-81
3.4-11	Recovered R2 panel from *Columbia* (after it was unfolded)	3-81
3.4-12	Operation of the lever-lock controller switches	3-81
3.4-13a	View of middeck floor and Volume E, looking aft	3-82
3.4-13b	Scenario showing how the crew module pressure vessel could impact the forward fuselage, and the middeck Volume E could impact the crew module pressure vessel, with resultant damage	3-82
3.4-14	[Redacted]	3-82
3.4-15	Cabin depressurization timeline	3-84
3.4-16	Representative acceleration profiles from the orbiter breakup (Catastrophic Event) to the orbiter forebody breakup (Crew Module Catastrophic Event) based on aerodynamic modeling	3-85
3.4-17	Detail of the five-point harness	3-86
3.4-18	Demonstration of an integrated seat/suit/crew member in entry configuration	3-87
3.4-19	Example of an intact, pristine advanced crew escape suit nonconformal helmet (not an STS-107 helmet)	3-87
3.4-20	[Redacted]	3-87
3.4-21	Ground location of the recovered flight crew equipment	3-88
3.4-22	[Redacted]	3-88
3.4-23	[Redacted]	3-89
4-1	Concentric layers of protection for space shuttle crew members	4-4
4-2	Management and leadership structure	4-5
4-3	Functional organization of the Spacecraft Crew Survival Integration Investigation Team	4-6
4-4	Visual representation of the breakup sequence, looking east	4-14
4-5	Orbiter and debris trajectories, looking down at Texas in a westerly direction	4-15
4-6	Crew module debris groups, looking south-southwest	4-15
4-7	Virtual reconstruction of the aft flight deck and middeck floor	4-16
4-8	Forward fuselage/crew module (forebody) configuration	4-17
4-9	Coefficients that were used to define the vehicle outer mode line shape	4-18
4-10	Computational fluid dynamics analysis of heating at the tip of the payload bay door roller for orthogonal geometry into the direction of travel at M = 10.5	4-20
4-11	Plot for a pair of number strings that was interpreted for three different latitude/longitude formats	4-25

Tables

2.1-1	Maximum Survivable Altitude of Crew Module	2-29
2.1-2	Material Properties Important to Thermal Analyses	2-41
2.2-1	Videos Used to Create the Debris Tree in Figure 2.2-9	2-52
2.2-2	Orbiter Breakup Event Sequence and Times Determined by Video	2-54
2.2-3	Release Times of Large Structures	2-56
2.2-4	Comparison of Video and Ballistic Sequence	2-57
2.2-5	Comparison of Orbiter Breakup Sequence from Video, Ballistic, and Cluster Analysis	2-65
2.3-1	Westernmost Crew Module Debris	2-89
2.4-1	Ballistic Timeline for Post-Crew Module Catastrophic Event Forebody Events	2-99
2.4-2	Individual Debris Objects for the Crew Module Post-Crew Module Catastrophic Event	2-100
2.4-3	Specified Orbiter Window Design Thicknesses	2-103
3.1-1	Upper Seatback Comparisons	3-15
3.1-2	Inertial Reel Strap Comparisons	3-22
3.2-1	Recovered Crew Worn Components	3-46
4-1	Principal Spacecraft Crew Survival Integration Investigation Team Members	4-6
4-2	Individuals Supporting the Principal Spacecraft Crew Survival Integration Investigation Team	4-7
4-3	Experts External to NASA	4-8
4-4	Data Types and Data Sources	4-10
4-5	Videos That Were Used	4-22

Acronyms and Abbreviations

A/G	air-to-ground
A/G1	air-to-ground 1
A/N PRC	Army/Navy personal radio communications
AC	alternating current
AC3	alternating current Bus 3
ACES	advanced crew escape suit
AFIP	Air Force Institute of Pathology
AOD	automatic opening device
AOS	acquisition of signal
APU	auxiliary power unit
ASCAN	astronaut candidate
BFS	backup flight software
BIP	bio-instrument pass-through
BLIMP-K	Boundary Layer Integral Matrix Procedure-Kinetic
BN	ballistic number
BRC	Biodynamic Research Corporation
Btu	British thermal unit
Btu/lbm	Btu per pound of mass
c.g.	center of gravity
CAIB	*Columbia* Accident Investigation Board
CAPCOM	capsule communicator
CCA	communications carrier assembly
C_D	coefficient of drag
CDR	Commander
CE	Catastrophic Event
CEE	crew escape equipment
CEL	Concept Exploration Laboratory
CES	Crew Escape System
CFD	computational fluid dynamics
CH4	Channel 4
CM	crew module
CMCE	Crew Module Catastrophic Event
CO_2	carbon dioxide
COMM	communications
CRD	*Columbia* Reconstruction Database
CRP	*Columbia* Research and Preservation
CRT	cathode ray tube
CSS	control stick steering
CSWG	Crew Survival Working Group
CTB	cargo transfer bag
CWE	crew worn equipment
D/O PREP	deorbit preparation
DAP	digital autopilot
DD.ddddd	degrees and decimal degrees
DD MM.mmm	degrees, minutes, and decimal minutes
DD MM SS.sss	degrees, minutes, seconds, and decimal seconds
DNA	deoxyribonucleic acid
DoD	Department of Defense

DP/dt	pressure change rate
DSC	dual suit controller
ECLSS	Environmental Control and Life Support System
EI	entry interface
EOS	Emergency Oxygen System
ET	external tank
EVA	extravehicular activity
FAA	Federal Aviation Administration
FBI	Federal Bureau of Investigation
FCS	Flight Control System
FF	forward fuselage
FOV	field of view
fps	feet per second
FREESTAR	Fast Reaction Experiments Enabling Science, Technology, Applications, and Research
FSP	Fault Summary Page
GMT	Greenwich Mean Time
GPC	general purpose computer
GPS	Global Positioning System
hh:mm:ss	hours:minutes:seconds
ICU	individual cooling unit
ISAG	Image Science and Analysis Group
ISS	International Space Station
JSC	Johnson Space Center
KEAS	knots equivalent air speed
KSC	Kennedy Space Center
LCAT	Large Core Arc Tunnel
LCG	liquid cooling garment
LED	light emitting diode
LIB	left inboard
LiOH	lithium hydroxide
LOB	left outboard
LOC	loss of control
LOS	loss of signal
LPU	life preserver unit
LRWG	Late Re-entry Working Group
MADS	Module Auxiliary Data System
MAGR	Miniaturized Airborne Global Receiver
MAP	middeck access panel
MAR	middeck accommodation rack
MCC	Mission Control Center
μm	micron
MMACS	Mechanical, Maintenance, Arm, and Crew Systems
MPS	Main Propulsion System
MS	Mission Specialist
MSID	measurement and stimulus indication
N_2	nitrogen
NBC	National Broadcasting Corporation
NET	no earlier than
NLT	no later than
NO	nitric oxide
NTSB	National Transportation Safety Board

O_2	oxygen
OEX	orbiter experiment
OMS	Orbital Maneuvering System
ORSAT	Object Reentry Survival Analysis Tool
OSL	off-scale low
PASS	Primary Avionics Software System
PGSC	payload general support computer
PLBD	payload bay door
PLT	Pilot
PPA	personal parachute assembly
$ppCO_2$	partial pressure of carbon dioxide
ppO_2	partial pressure of oxygen
PS	Payload Specialist
psf	pounds per square foot
psi	pounds per square inch
psia	pounds per square inch absolute
psid	pounds per square inch differential
QD	quick disconnect
RCC	reinforced carbon-carbon
RCS	Reaction Control System
RGPC	reconstructed general purpose computer
RHC	rotational hand controller
RPTA	rudder pedal transducer assembly
SARSAT	search and rescue satellite-aided tracking
SCSIIT	Spacecraft Crew Survival Integrated Investigation Team
SEAWARS	Seawater Activated Release System
SES	Shuttle Engineering Simulator
SFRM	space flight resource management
SINDA	Systems Improved Numerical Differencing Analyzer
SMPTE	Society of Motion Picture and Television Engineers
SMS	shuttle mission simulator
SORT	Simulation and Optimization of Rocket Trajectories
SSME	space shuttle main engine
SUPA	shuttle urine pretreat assembly
3-D	three-dimensional
TAA	tunnel adapter assembly
TCDT	terminal countdown demonstration test
TD	Total Dispersal thickness dimension
TDRS-W	Telemetry, Tracking, and Data Relay Satellite-West
TELCU	thermal electric liquid cooling unit
TET	tape elapsed time
TIG	time of ignition
TIM	Technical Interchange Meeting
TiO_2	titanium oxide
TPS	Thermal Protection System
TSUB-A	Telonics Satellite Uplink Beacon-A
TUC	time of useful consciousness
WCS	Waste Collection System
WSB	water spray boiler

Executive Summary

Background

NASA commissioned the Columbia Accident Investigation Board (CAIB) to conduct a thorough review of both the technical and the organizational causes of the loss of the Space Shuttle *Columbia* and her crew on February 1, 2003. The accident investigation that followed determined that a large piece of insulating foam from *Columbia*'s external tank (ET) had come off during ascent and struck the leading edge of the left wing, causing critical damage. The damage was undetected during the mission. The CAIB's findings and recommendations were published in 2003 and are available on the web at http://caib.nasa.gov/. NASA responded to the CAIB findings and recommendations with the Space Shuttle Return to Flight Implementation Plan.[1] Significant enhancements were made to NASA's organizational structure, technical rigor, and understanding of the flight environment. The ET was redesigned to reduce foam shedding and eliminate critical debris. In 2005, NASA succeeded in returning the space shuttle to flight. In 2010, the space shuttle will complete its mission of assembling the International Space Station and will be retired to make way for the next generation of human space flight vehicles: the Constellation Program.

The Space Shuttle Program recognized the importance of capturing the lessons learned from the loss of *Columbia* and her crew to benefit future human exploration, particularly future vehicle design. The program commissioned the Spacecraft Crew Survival Integrated Investigation Team (SCSIIT). The SCSIIT was asked to perform a comprehensive analysis of the accident, focusing on factors and events affecting crew survival, and to develop recommendations for improving crew survival for all future human space flight vehicles. To do this, the SCSIIT investigated all elements of crew survival, including the design features, equipment, training, and procedures intended to protect the crew. This report documents the SCSIIT findings, conclusions, and recommendations.

Results

One of the more difficult problems facing the SCSIIT was how to characterize events that occurred in an operating regime that was far outside the collective experience of aircraft accident investigation and without significant applicable test data. The investigation relied on data in the form of video, recovered debris, and medical findings, each supplemented with modeling and analyses when needed. The SCSIIT used these data to identify all events with lethal potential (even those that occurred after the crew was deceased) during entry so that threats to crew survival could be described and methodically approached in future designs. In the course of the investigation, five events with lethal potential were identified.

1. **Depressurization of the crew module at or shortly after orbiter breakup.**
   The pressure suit used by space shuttle crews on ascent and entry was not a part of the initial design of the orbiter. It was introduced in response to the *Challenger* accident. While it protects the crew from many contingency scenarios, there are several areas where integration difficulties diminish the capability of the suit to protect the crew. The *Columbia* depressurization event occurred so rapidly that the crew members were incapacitated within seconds, before they could configure the suit for full protection from loss of cabin pressure. Although circulatory systems functioned for a brief time, the effects of the depressurization were severe enough that the crew could not have regained consciousness. This event was lethal to the crew.

[1]NASA's Implementation Plan for Space Shuttle Return to Flight and Beyond, 10th Edition, June 3, 2005.

Key Recommendations

Space shuttle crew training should include greater emphasis on the transition between problem-solving and survival operations.

Future spacecraft must fully integrate suit operations into the design of the vehicle and provide features that will protect the crew without hindering normal operations.

2. **Exposure of unconscious or deceased crew members to a dynamic rotating load environment with a lack of upper body restraint and nonconformal helmets.**
When the orbiter lost control, the resultant motion was not lethal but did require crew members to brace against the motion. The forebody, which is made up of the crew module and forward fuselage, separated at orbiter breakup. The forebody continued to rotate. After the crew lost consciousness due to the loss of cabin pressure, the seat inertial reel mechanisms on the crews' shoulder harnesses did not lock. As a result, the unconscious or deceased crew was exposed to cyclical rotational motion while restrained only at the lower body. Crew helmets do not conform to the head. Consequently, lethal trauma occurred to the unconscious or deceased crew due to the lack of upper body support and restraint.

Key Recommendations

Crew procedures must be re-evaluated in light of the findings regarding the motion of the intact orbiter and the forebody after separation.

Future spacecraft should be evaluated for loss of control motion and dynamics for adequate integration into development, design, and crew training.

Future spacecraft seats and suits should be integrated to ensure proper restraint of the crew in off-nominal situations while not affecting operational performance. Future crewed spacecraft vehicle design should account for vehicle loss of control to maximize the probability of crew survival.

3. **Separation of the crew from the crew module and the seat with associated forces, material interactions, and thermal consequences.**
The breakup of the crew module and the crew's subsequent exposure to hypersonic entry conditions was not survivable by any currently existing capability. It was an extremely significant event, but it was very difficult to characterize because many events appeared to happen in a short period of time. The actual maximum survivable altitude for the crew module following a breakup of the orbiter is too complex to compute because it depends on the altitude and velocity at release as well as rotational dynamics that are understood only in a general way. The lethal-type consequences of exposure to entry conditions included traumatic injury due to seat restraints, high loads associated with deceleration due to a change in ballistic number, aerodynamic loads, and thermal events. Crew circulatory functions ceased shortly before or during this event. The ascent and entry suit had no performance requirements for occupant protection from thermal events. The only known complete protection from this event would be to prevent its occurrence.

Key Recommendation

Future vehicle design should incorporate an analysis for loss of control/breakup to optimize for the most graceful degradation of vehicle systems and structure to enhance chances for crew survival. Operational procedures can then integrate the most likely scenarios into survival strategies.

4. **Exposure to near vacuum, aerodynamic accelerations, and cold temperatures.**
The ascent and entry suit system is certified to a maximum altitude of 100,000 feet and velocity of 560 knots equivalent air speed. It is uncertain whether it can protect a crew member at higher altitudes and air speeds.

Key Recommendation

Crew survival suits should be evaluated as an integrated system to determine the various weak points (thermal, pressure, windblast, chemical exposure, etc.). Once identified, alternatives should be explored to strengthen the weak areas.

5. Ground impact.
The ascent and entry suit system provides protection from ground impact with a parachute system. The current parachute system requires manual action by a crew member to activate the opening sequence.

Key Recommendation

Future spacecraft crew survival systems should not rely on manual activation to protect the crew.

Improving Crew Survival Investigations

The SCSIIT also identified recommendations regarding crew survival investigations. These include that:

- Crew survival investigations should be given high priority for all future spacecraft mishaps. Medically sensitive data should always be protected to preserve the privacy of the victims and their families. Because there is a limited database of information, each accident provides crucial understanding of the environment and expanding the envelope for survival.
- Data management proved to be critical to the investigation in many areas, such as equipment identification marking, debris recovery ground coordinates, database documentation, and tracking versions of reports, briefings, and analyses. Preservation of debris and data that may be of value in future investigations should be standardized and continued.

The SCSIIT investigation was performed with the belief that a comprehensive, respectful investigation could provide knowledge that would improve the safety of future space flight crews and explorers. By learning these lessons and ensuring that we continue the journey begun by the crews of Apollo 1, *Challenger*, and *Columbia*, we help to give meaning to their sacrifice and the sacrifice of their families. It is for them, and for the future generations of explorers, that we strive to be better and go farther.

Introduction

Human space flight is still in its infancy; spacecraft navigate narrow tracks of carefully computed ascent and entry trajectories with little allowable deviation. Until recently, it remained the province of a few governments. As private industry and more countries join in this great enterprise, we must share findings that may help protect those who venture into space. In the history of NASA, this approach has resulted in many improvements in crew survival. After the Apollo 1 fire, sweeping changes were made to spacecraft design and to the way crew rescue equipment was positioned and available at the launch pad. After the *Challenger* accident, a jettisonable hatch, personal oxygen systems, parachutes, rafts, and pressure suits were added to ascent and entry operations of the space shuttle.

As we move toward a time when human space flight will be commonplace, there is an obligation to make this inherently risky endeavor as safe as feasible. Design features, equipment, training, and procedures all play a role in improving crew safety and survival in contingencies. In aviation, continual improvement in oxygen systems, pressure suits, parachutes, ejection seats, and other equipment and systems has been made. It is a core value in the aviation world to evaluate these systems in every accident and pool the data to understand how design improvements may improve the chances that a crew will survive in a future accident.

The *Columbia* accident was not survivable. After the *Columbia* Accident Investigation Board (CAIB) investigation regarding the cause of the accident was completed, further consideration produced the question of whether there were lessons to be learned about how to improve crew survival in the future.

This investigation was performed with the belief that a comprehensive, respectful investigation could provide knowledge that can protect future crews in the worldwide community of human space flight. Additionally, in the course of the investigation, several areas of research were identified that could improve our understanding of both nominal space flight and future spacecraft accidents.

This report is the first comprehensive, publicly available accident investigation report addressing crew survival for a human spacecraft mishap, and it provides key information for future crew survival investigations. The results of this investigation are intended to add meaning to the sacrifice of the crew's lives by making space flight safer for all future generations.

The Columbia *Accident Investigation Board Report*

The CAIB completed its investigation into the *Columbia* mishap and published Volume I of its report in August 2003. Five supporting volumes were subsequently completed and published. The CAIB Report provides a thorough study of the accident and its causes. Since the crew had no role in causing the accident, the CAIB Report contained limited discussion of crew-related events. Although the CAIB Report included no formal recommendations concerning crew survival, it did contain the following relevant observation:[2]

> Observation 10.2-1 *Future crewed-vehicle requirements should incorporate the knowledge gained from the* Challenger *and* Columbia *accidents in assessing the feasibility of vehicles that could ensure crew survival even if the vehicle is destroyed*.

[2]*Columbia* Accident Investigation Board Report, Volume I, Section 10.2, Crew Escape and Survival, August 2003.

Additionally, Appendix G12, Crew Survivability, page 355, Volume V, October 2003 added:

> *To enhance the likelihood of crew survivability, NASA must evaluate the feasibility of improvements to protect the crew cabin in existing orbiters.*
>
> *NASA should investigate techniques that will prevent the structural failure of the CM* [crew module] *due to thermal degradation of structural properties and determine their feasibility for application.*
>
> *Future crewed vehicles should incorporate the knowledge gained from the 51-L* [Challenger] *and STS-107 mishaps in assessing the feasibility of designing vehicles that will provide for crew survival even in the face of a mishap that results in the loss of vehicle.*
>
> *Crew procedures and techniques for use of CWE [crew worn equipment] should be standardized and complied with by all crewmembers.*

To address post-*Columbia* Return to Flight actions, the Space Shuttle Program approved the formation of a multidisciplinary Spacecraft Crew Survival Integrated Investigation Team (SCSIIT) in July 2004. The team's primary objective was to combine engineering and medical analyses to determine what happened to the crew module and the *Columbia* crew to enhance crew safety and survival for future human space flights. This effort built upon and extended the activities of the Crew Survival Working Group (CSWG), which was formed at the time of the CAIB investigation.

In many regards this investigation presented several challenges. First, space flight is a relatively new and rare experience and there have been only a few fatal mishaps. Consequently, there is no integrated or widely available body of information regarding the analysis of spacecraft accidents for crew survival. The environment of atmospheric entry is also unique when compared to aviation. The SCSIIT had to break new ground in conducting the investigation of a singular event in such a complex environment. The team had to modify existing models and tools, normally used for specific nominal situations in a predictive manner, to understand the mishap environment. Many of the technical tools and concepts used will be of great assistance to a future spacecraft accident investigator. With the proliferation of commercial and international human space flight activities, it is crucial that all participants begin to develop a more comprehensive process and database of information regarding spacecraft accident investigation.

Because of the nature of the *Columbia* accident, there are many unknowns associated with it. The SCSIIT attempted to address these unknowns through calculated judgment and some speculation. In the end, there were varying degrees of certainty and confidence. The word "probable" refers to events that the team was very confident occurred. "Likely" refers to events that the team is somewhat confident occurred, although supporting evidence may be less definite. When an event is described as "possible," it generally reflects a lack of data to confirm or refute the scenario but is still considered valuable to mention. The reconstruction of this accident relied on a wide array of data. This report reflects the final consensus reached by the investigators.

Summary of Conclusions and Recommendations

Lethal events

The SCSIIT framed its analysis by attempting to identify all of the potentially lethal events that occurred during the mishap, including those that occurred after the crew was deceased. This allowed the team to identify specific threats to crew survival at different phases of entry and address those threats in recommendations for future vehicles. In the course of the investigation, the SCSIIT identified five events with lethal potential. These events are summarized below, along with the findings and recommendations that accompany each one. Each event is discussed in detail in the body of the report.

1. **The first event with lethal potential was depressurization of the crew module, which started at or shortly after orbiter breakup.**
The majority of the SCSIIT findings related to the first lethal event were connected to the operational incompatibilities of the advanced crew escape suit (ACES) with the orbiter. The launch and entry suit was added in response to the *Challenger* accident, rather than as a part of the original vehicle design. The ACES was the successor to that suit. The suit protects the crew in many scenarios; however, there are several areas where integration difficulties diminish the capability of the suit to protect the crew. Integration issues include: the crew cannot keep their visors down throughout entry because doing so results in high oxygen concentrations in the cabin; gloves can inhibit the performance of nominal tasks; and the cabin stow/deorbit preparation timeframe is so busy that sometimes crew members do not have enough time to complete suit-related steps prior to atmospheric entry.

   As *Columbia* entered the atmosphere, one crew member was not yet wearing the ACES helmet and three crew members were not wearing gloves. Per nominal procedures, the crew wearing helmets had visors up. There was a period of about 40 seconds after the orbiter loss of control (LOC) but prior to depressurization when the crew was conscious and capable of action. Part of this short timeframe was undoubtedly employed in recognizing that a problem existed, as the indications of LOC developed gradually. The crew members could have closed their visors in this timeframe but did not. The SCSIIT attributed this to the training regimen, which separates vehicle systems training from emergency egress training and does not emphasize the *transition* between problem resolution and a survival situation. Once the cabin depressurization began, the rate of depressurization incapacitated the crew so quickly that even those crew members who had fully donned the ACES did not have time to lower their visors. Although circulatory systems functioned for a brief time, the crew could not have regained consciousness upon descent to lower altitudes due to the effects of the depressurization.

   ### *Key Recommendations*

   - Crew survival systems and procedures should be incorporated early into future spacecraft designs to ensure that they are compatible with nominal operations and that sufficient time exists to ensure all safety-critical equipment can be configured prior to entry interface.
   - The training program should be evaluated to determine how to best incorporate the transition from problem-solving to survival.
   - Future spacecraft crew survival systems should not rely solely on manual activation to protect the crew.

2. **The second event with lethal potential was unconscious or deceased crew members exposed to a dynamic rotating load environment with nonconformal helmets and a lack of upper body restraint.**
The orbiter lost control, probably when the hydraulic systems failed due to hot gas intrusion in the left wing. The resulting motion was not lethal but did require bracing by the crew. The forebody (crew module and forward fuselage) eventually separated and the crew module lost pressure at orbiter breakup. When it separated, the forebody began a multi-axis rotation at approximately 0.1 revolution/second. Loads due to deceleration significantly decreased at the moment of breakup due to the change in ballistic number, but began to climb as the forebody continued to decelerate.

   After the crew module depressurized and the crew lost consciousness, the seat inertial reel mechanisms failed to lock despite the off-nominal motion. The reels were not defective; they were simply not designed to lock under the conditions the forebody experienced. The upper harness straps failed at some point prior to the forebody breakup, causing the straps to recoil back into the inertial reel mechanism. Because the reel mechanisms did not lock, the unconscious or deceased crew members were exposed to cyclical rotational motion while their upper bodies were inadequately restrained. Helmets that did not conform to the head and the lack of upper body restraint resulted in injuries and lethal trauma.

Current emergency egress procedures for a vehicle LOC or breakup assume that the crew module will eventually stop rotating and will stabilize in a specific attitude. Aerodynamic analysis completed during this investigation shows that this is extremely unlikely. Further, the procedures are based on ascent conditions only.

Key Recommendations

- Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments. Inertial reels should be evaluated for appropriateness of design for off-nominal scenarios.
- Helmets should provide head and neck protection in off-nominal dynamic load conditions. The current space shuttle inertial reels should be manually locked at the first sign of an off-nominal situation.
- A team of crew escape instructors, flight directors, and astronauts should be assembled to assess orbiter procedures in the context of ascent, deorbit, and entry contingencies.
- Future spacecraft should be evaluated while still in the design phase for dynamics and entry thermal and aerodynamic loads during a vehicle LOC for adequate integration into development, design, and crew training.
- Future crewed spacecraft vehicle design should account for vehicle LOC contingencies to maximize the probability of crew survival.

3. The third event with lethal potential was separation from the crew module and the seats with associated forces, material interactions, and thermal consequences. This event is the least understood due to limitations in current knowledge of mechanisms at this Mach number and altitude. Seat restraints played a role in the lethality of this event.

The breakup of the crew module and resultant exposure of the crew to entry conditions was an extremely significant event but was very difficult to characterize since many related events occurred in a short period of time. The consequences of exposure to entry conditions included traumatic injury related to seat restraints, high loads associated with deceleration due to a change in ballistic number, aerodynamic loads, and thermal events. All crew were deceased before, or by the end of, this event. The ACES has no performance requirements for occupant protection from thermal events and may not provide adequate protection even for egress scenarios involving heat and flames. There is no known complete protection from the breakup event except to prevent its occurrence.

The actual maximum survivable altitude for the crew module following a breakup of the orbiter is too complex to compute because it depends on the altitude and velocity at release as well as rotational dynamics, which are understood only in a general way.

Key Recommendations

- Future vehicle design should incorporate analysis for LOC/breakup to optimize for the most graceful degradation to vehicle systems and structure to enhance chances for crew survival. Operational procedures can then integrate the most likely scenarios into survival strategies.
- Future spacecraft suits, seats, and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments.
- Crew survival systems should be evaluated as an integrated system that includes boots, helmet, seat restraints, etc. to determine the various weak points (thermal, pressure, windblast, chemical exposure, etc.). Once identified, alternatives should be explored to strengthen the weak areas. Materials with low resistance to chemicals, heat, and flames should not be used on equipment that is intended to protect the wearer from such hostile environments.

4. **The fourth event with lethal potential was exposure to near vacuum, aerodynamic accelerations, and cold temperatures.**
   The ACES system is certified to operate at a maximum altitude of 100,000 feet, and certified to survive exposure to a maximum velocity of 560 knots equivalent air speed. The operating envelope of the orbiter is much greater than this. The actual maximum protection environment for the ACES is not known.

   The recommendation to strengthen the weak areas of the suit system will increase the probability of survival through this type of event as well.

5. **The final event with lethal potential was ground impact.**
   The ACES system provides protection from ground impact with a parachute system. The current parachute system requires manual action by a crew member to activate the opening sequence.

   The earlier recommendation that future survival systems should not rely on manual activation will address this lethal event as well.

Crew survival accident investigation

Although this investigation was a follow-up to the actual mishap investigation, there were many findings, conclusions, and recommendations that apply to spacecraft accident investigations in general and crew survival investigations in particular. The recommendations address both NASA processes and investigation processes in general.

This crew survival investigation was difficult to do because of both the technical complexity and the sensitivity of the topic. Other Return to Flight activities took priority over the crew survival follow-up investigation, leading to resource issues for the SCSIIT.

Key Recommendations

- In the event of a future fatal spacecraft mishap, NASA should place a high priority on the performance of crew survival investigations.
- Medically sensitive and personal effects data should always be protected to preserve the privacy of the victims and their families. Issues surrounding public release of this type of sensitive information during a NASA accident investigation should be resolved and policies documented throughout the agency to ensure future crew survival investigations are performed.
- Stress debriefings and other counseling services should be available to those experiencing ongoing stress as a result of participating in the debris recovery and investigation.
- Data management proved to be critical to the investigation in many areas. Specifically, location of and access to debris recovery ground coordinates, database documentation, and configuration management of versions of reports, briefings, and analyses were all important. Many elements were highly successful, but improvements could be made. Global Positioning System coordinates for recovered items should be standardized.[3] Configuration control for documents was not initiated as early as it could have been. Additionally, *Challenger* supporting data were generally not cataloged by references to crew survival or the crew module. It was extremely difficult to find relevant data. *Challenger* debris is unpreserved and inaccessible for analysis. Report generation should start early in the investigation process to help provide consistency and documentation. Preservation of debris and data that may be of value in future investigations should be continued using the approach of the *Columbia* Research and Preservation Team. Accident investigation teams should develop standard templates across all areas of data management for the types of investigations performed for *Columbia*.

[3]STS-107 *Columbia* Reconstruction Report, NSTS-60501, June 30, 2003, p. 142.

- Configuration management documentation of seat and suit components had a significant impact on the investigation – positive when done well and negative when inadequate. Serialization and quality marking requirements and policies for space flight hardware should be developed to the lowest component level practical to aid in accident investigation.
- There were many findings relative to ground-based video of the entry and mishap. The video was a vital source of data for understanding the accident, especially after telemetry was no longer available. Video was used to timeline key events and to help understand the motion and trajectory of the objects of interest. One video that proved important had been mis-categorized initially, when the timeline of the accident was not yet understood, and was not used until very late in the investigation. After a mishap timeline has been established, videos should be re-reviewed to ensure relevant data are being used.

Other

Thermal analysis of some titanium components showed that entry heating alone was insufficient to cause the damage seen. Shock wave interactions can account for the damage to some extent, but arc jet testing showed that titanium combustion may also have played a role. In many other cases, evaluation of material performance in the low-pressure, high-temperature environment did not exist. Studies should be performed to further characterize the material behavior of titanium in entry environments to better understand optimal space applications of this material.

Summary

In summary, many findings, conclusions, and recommendations have resulted from this investigation that will be valuable both to spacecraft designers and accident investigators. This report provides the reader an expert level of knowledge regarding the sequence of events that contributed to the loss of *Columbia*'s crew on February 1, 2003 and what can be learned to improve the safety of human space flight for all future crews. It is the team's expectation that readers will approach the report with the respect and integrity that the subject and the crew of *Columbia* deserve.

Report Format

The *Executive Summary* highlights the intention of the investigation and key results.

Introduction explains the purpose and scope of the report and summarizes key results.

Conclusions and Recommendations contains a brief summary of all conclusions and recommendations.

Chapter 1 Integrated Story provides the sequence of events of the mishap that related to the crew.

1.1 Integrated Investigation Results brings together into one integrated story the results from the various aspects of the investigation.

1.2 Master Timeline provides a reference timeline of key events including vehicle configuration and status, crew activities, and changing circumstances.

Chapters 2 and 3 provide detailed insight into the examination, analysis, and understanding of evidence and results. Findings, conclusions, and recommendations are embedded in these chapters. The different sections address the accident from the individual subject matter perspective. By intent, these sections contain highly interrelated data and information that are shared or duplicated. This repetition provides mutually supportive information from the different aspects of the investigation. It is expected that future technical readers may only have interest in a few of these sections, so repetition was accepted for completeness so that each section could stand on its own. To aid future investigations, this report contains substantially more technical data and information than is normally contained in an aviation mishap report.

Chapter 2 Vehicle Failure Assessment describes the vehicle analyses. These were used to understand what happened to the orbiter and the crew module structures.

2.1 Motion and Thermal Analyses describes the analyses performed to understand the motion of the vehicle and crew module and the resultant loads acting upon the crew and structure. Thermal analyses rely heavily on trajectory assessments and, therefore, are also covered in this section.

2.2 Orbiter Breakup Sequence describes the analyses performed to understand the sequence in which the orbiter breakup occurred.

2.3 Crew Cabin Pressure Environment Analysis describes the integrated analyses used to understand the cabin pressure conditions of the crew module during the mishap.

2.4 Forebody Breakup Sequence describes the analyses performed to understand the sequence of the breakup of the forebody (crew module and forward fuselage) of the orbiter.

Chapter 3 Occupant Protection addresses crew and crew equipment special assessments.

3.1 Crew Seats and *3.2 Crew Worn Equipment* address the function and performance of the equipment intended to protect the crew in the experienced motion, load, and thermal environment.

3.3 Crew Training addresses procedures and preparations associated with examined events and activities.

3.4 Crew Analysis encompasses the awareness the crew had of events, crew actions in response to the events, and the events of lethal potential to which the crew was exposed.

Chapter 4 Investigative Methods and Processes explains the structure and makeup of the team, the approach taken to conduct the investigation, a description of the tools used, and the collection and management of data and information.

Future Work addresses suggestions for forward work.

Appendix A contains a tutorial on ballistic trajectories, providing more conceptual insight into a critical topic discussed extensively in the report.

Conclusions and Recommendations

The first event with lethal potential was depressurization of the crew module, which started at or shortly after orbiter breakup.

Conclusion L1-1. After loss of control at GMT[4] 13:59:37 and prior to orbiter breakup at GMT 14:00:18, the *Columbia* cabin pressure was nominal and the crew was capable of conscious actions. (p. 2-89, p. 3-82)

Conclusion L1-2. The depressurization was due to relatively small cabin breaches above and below the middeck floor and was not a result of a major loss of cabin structural integrity. (p. 2-93)

Conclusion L1-3. The crew was exposed to a pressure altitude above 63,500 feet, indicating that the cabin depressurization event occurred above this altitude. (p. 2-91, p. 3-83)

Conclusion L1-4. The crew was not exposed to a cabin fire or thermal injury prior to depressurization, cessation of breathing, and loss of consciousness. (p. 3-89)

Conclusion L1-5. The depressurization incapacitated the crew members so rapidly that they were not able to lower their helmet visors. (p. 2-90, p. 3-84)

Recommendation L1-1. Incorporate objectives in the astronaut training program that emphasize understanding the transition from recoverable systems problems to impending survival situations. (p. 3-66)

Recommendation L1-2. Future spacecraft and crew survival systems should be designed such that the equipment and procedures provided to protect the crew in emergency situations are compatible with nominal operations. Future spacecraft vehicles, equipment, and mission timelines should be designed such that a suited crew member can perform all operations without compromising the configuration of the survival suit during critical phases of flight. (p. 3-38, p. 3-86)

Recommendation L1-3/L5-1. Future spacecraft crew survival systems should not rely on manual activation to protect the crew. (p. 3-20, p. 3-44, p. 3-84)

Recommendation L1-4. Future suit design should incorporate the ability for crew members to communicate visors-down without relying on spacecraft power. (p. 3-82)

The second event with lethal potential was unconscious or deceased crew members exposed to a dynamic rotating load environment with nonconformal helmets and a lack of upper body restraint.

Conclusion L2-1. Between orbiter breakup and the forebody[5] breakup, the free-flying forebody was rotating about all three axes at approximately 0.1 rev/sec and did not trim into a specific attitude. (p. 2-23)

[4]Greenwich Mean Time.
[5]The orbiter forebody consists of the crew module, forward fuselage, forward Reaction Control System, nose cap, and nose landing gear.

Conclusion L2-2. The seat inertial reels did not lock. (p. 3-20)

Conclusion L2-3. Lethal injuries resulted from inadequate upper body restraint and protection during rotational motion. (p. 3-20, p. 3-87)

Recommendation L2-1. Assemble a team of crew escape instructors, flight directors, and astronauts to assess orbiter procedures in the context of ascent, deorbit, and entry contingencies. Revise the procedures with consideration to time constraints and the interplay among the thermal environment, expected crew module dynamics, and crew and crew equipment capabilities. (p. 3-67)

Recommendation L2-2. Prior to operational deployment of future crewed spacecraft, determine the vehicle dynamics, entry thermal and aerodynamic loads, and crew survival envelopes during a vehicle loss of control so that they may be adequately integrated into training programs. (p. 2-10, p. 2-29, p. 3-67)

Recommendation L2-3. Future crewed spacecraft vehicle design should account for vehicle loss of control contingencies to maximize the probability of crew survival. (p. 3-67)

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments. (p. 3-20, p. 3-53, p. 3-87, p. 3-88)

Recommendation L2-5. Incorporate features into the pass-through slots on the seats such that the slot will not damage the strap. (p. 3-24)

Recommendation L2-6. Perform dynamic testing of straps and testing of straps at elevated temperatures to determine load-carrying capabilities under these conditions. Perform testing of strap materials in high-temperature/low-oxygen/low-pressure environments to determine materials properties under these conditions. (p. 3-27)

Recommendation L2-7. Design suit helmets with head protection as a functional requirement, not just as a portion of the pressure garment. Suits should incorporate conformal helmets with head and neck restraint devices, similar to helmet/head restraint techniques used in professional automobile racing. (p. 3-53, p. 3-87)

Recommendation L2-8. The current shuttle inertial reels should be manually locked at the first sign of an off-nominal situation. (p. 3-21, p. 3-88)

Recommendation L2-9. The use of inertial reels in future restraint systems should be evaluated to ensure that they are capable of protecting the crew during nominal and off-nominal situations without active crew intervention. (p. 3-88)

The third event with lethal potential was separation from the crew module and the seats with associated forces, material interactions, and thermal consequences. This event is the least understood due to limitations in current knowledge of mechanisms at this Mach number and altitude. Seat restraints played a role in the lethality of this event.

Conclusion L3-1. Complete loss of hydraulic pressure to the aerosurfaces resulting from the breach in the left wing was the probable proximal cause for the vehicle loss of control. (p. 2-6)

Conclusion L3-2. The breakup of both *Challenger* and *Columbia* resulted in most of the X_o 582[6] ring frame bulkhead remaining with the crew module or forebody. (p. 2-84)

[6] X_o 582 refers to the location of the bulkhead in the orbiter coordinate frame. This bulkhead is immediately aft of the crew module.

Conclusion L3-3. The actual maximum survivable altitude for a breakup of the space shuttle is not known. (p. 2-29)

Conclusion L3-4. The seat restraint system caused lethal-level injuries to the unconscious or deceased crew members when they separated from the seat. (p. 3-88)

Recommendation L3-1. Future vehicles should incorporate a design analysis for breakup to help guide design toward the most graceful degradation of the integrated vehicle systems and structure to maximize crew survival. (p. 2-87, p. 2-139, p. 3-88)

Recommendation L3-2. Future vehicles should be designed with a separation of critical functions to the maximum extent possible and robust protection for individual functional components when separation is not practical. (p. 2-6)

Recommendation L3-3. Future spacecraft design should incorporate crashworthy, locatable data recorders for accident/incident flight reconstruction. (p. 2-36)

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments. (p. 3-53)

Recommendation L3-5/L4-1. Evaluate crew survival suits as an integrated system that includes boots, helmet, and other elements to determine the weak points, such as thermal, pressure, windblast, or chemical exposure. Once identified, alternatives should be explored to strengthen the weak areas. Materials with low resistance to chemicals, heat, and flames should not be used on equipment that is intended to protect the wearer from such hostile environments. (p. 3-46, p. 3-63)

The fourth event with lethal potential was exposure to near vacuum, aerodynamic accelerations, and cold temperatures.

Conclusion L4-1. Although the advanced crew escape suit (ACES) system is certified to operate at a maximum altitude of 100,000 feet and to survive exposure to a maximum velocity of 560 knots equivalent air speed, the actual maximum protection environment for the ACES is not known. (p. 3-46)

See **Recommendation L3-5/L4-1** above, which also addresses this event.

The final event with lethal potential was ground impact.

Conclusion L5-1. The current parachute system requires manual action by a crew member to activate the opening sequence. (p. 3-44)

See **Recommendation L1-3/L5-1** above, which also addresses this event.

CREW SURVIVAL INVESTIGATIONS FOR SPACECRAFT ACCIDENTS

Recommendation A1. In the event of a future fatal human space flight mishap, NASA should place high priority on the crew survival aspects of the mishap both during the investigation as well as in its follow-up actions using dedicated individuals who are appropriately qualified in this specialized work. (p. 4-5, p. 4-9)

Recommendation A2. Medically sensitive and personal debris and data should always be available to designated investigators but protected from release to preserve the privacy of the victims and their families. (p. 4-11)

Recommendation A3. Resolve issues and document policies surrounding public release of sensitive information relative to the crew during a NASA accident investigation to ensure that all levels of the agency understand how future crew survival investigations should be performed. (p. 4-11)

Recommendation A4. Due to the complexity of the operating environment, in addition to traditional accident investigation techniques, spacecraft accident investigators must evaluate multiple sources of information including ballistics, video analysis, aerodynamic trajectories, and thermal and material analyses. (p. 4-9)

Recommendation A5. Develop equipment failure investigation marking ("fingerprinting") requirements and policies for space flight programs. Equipment fingerprinting requires three aspects to be effective: component serialization, marking, and tracking to the lowest assembly level practical. (p. 3-35, p. 3-63)

Recommendation A6. Standard templates for accident investigation data (document, presentation, data spreadsheet, etc.) should be used. All reports, presentations, spreadsheets, and other documents should include the following data on every page: title, date the file was created, date the file was updated, version (if applicable), person creating the file, and person editing the file (if different from author). (p. 4-10)

Recommendation A7. To aid in configuration control and ensure data are properly documented, report generation must begin early in the investigation process. (p. 4-10)

Conclusion A8-1. Spacecraft accidents are rare, and each event adds critical knowledge and understanding to the database of experience. (p. 3-84, p. 4-11)

Recommendation A8. As was executed with *Columbia*, spacecraft accident investigation plans must include provisions for debris and data preservation and security. All debris and data should be cataloged, stored, and preserved so they will be available for future investigations or studies. (p. 3-85, p. 4-11)

Recommendation A9. Post-traumatic stress debriefings and other counseling services should be available to those experiencing ongoing stress as a result of participating in the debris recovery and investigation. Designated personnel should follow up on a regular basis to ensure that individual needs are being met. (p. 4-12)

Recommendation A10. Global Positioning System receivers used for recording the latitude/longitude of recovered debris must all be calibrated the same way (i.e., using the same reference system), and the latitude/longitude data should be recorded in a standardized format.[7] (p. 4-25)

Recommendation A11. All video segments within a compilation should be categorized and summarized. All videos should be re-reviewed once the investigation has progressed to the point that a timeline has been established to verify that all relevant video data are being used. (p. 2-49, p. 4-23)

[7]STS-107 *Columbia* Reconstruction Report, NSTS-60501, June 30, 2003, p. 142.

ADDITIONAL CONCLUSIONS AND RECOMMENDATIONS

Conclusion A13-1. Titanium may oxidize and combust in entry heating conditions dependent on enthalpy, pressure, and geometry. (p. 2-45)

Conclusion A13-2. The heating from a Type IV shock-shock[8] impingement and titanium combustion (in some combination) likely resulted in the damage seen by the forward payload bay door rollers and the x-links.[9] (p. 2-45)

Recommendation A13. Studies should be performed to further characterize the material behavior of titanium in entry environments to better understand optimal space applications of this material. (p. 2-46)

[8]This refers to a specific type of intersecting hypersonic shock waves and is discussed in Section 2.1.
[9]The x-links are fittings that attach the crew module to the forward fuselage of the orbiter (see Section 2.1).

Chapter 1 – Integrated Story

1.1 *Integrated Investigation Results*
1.2 *Master Timeline*

1.1 Integrated Investigation Results

1.1.1 Events with lethal potential

There were five events identified with lethal potential to the crew.

The first event with lethal potential was depressurization of the crew module, which started at or shortly after orbiter breakup.

The second event with lethal potential was unconscious or deceased crew members exposed to a dynamic rotating load environment with nonconformal helmets and a lack of upper body restraint.

The third event with lethal potential was separation from the crew module and the seats with associated forces, material interactions, and thermal consequences. This event is the least understood due to limitations in current knowledge of mechanisms at this Mach number and altitude. Seat restraints played a role in the lethality of this event.

The fourth event with lethal potential was exposure to near vacuum, aerodynamic accelerations, and cold temperatures.

The final event with lethal potential was ground impact.

1.1.2 Integrated summary of events

This section provides an integrated summary of key events during the *Columbia* mishap as they relate to the crew and orbiter forebody (figures 1.1-1 and 1.1-2).[1] Figures 1.1-3 and 1.1-4 show depictions of the flight deck and middeck seats.

[1]The orbiter forebody consists of the crew module, forward fuselage, forward Reaction Control System (RCS), nose cap, and nose landing gear.

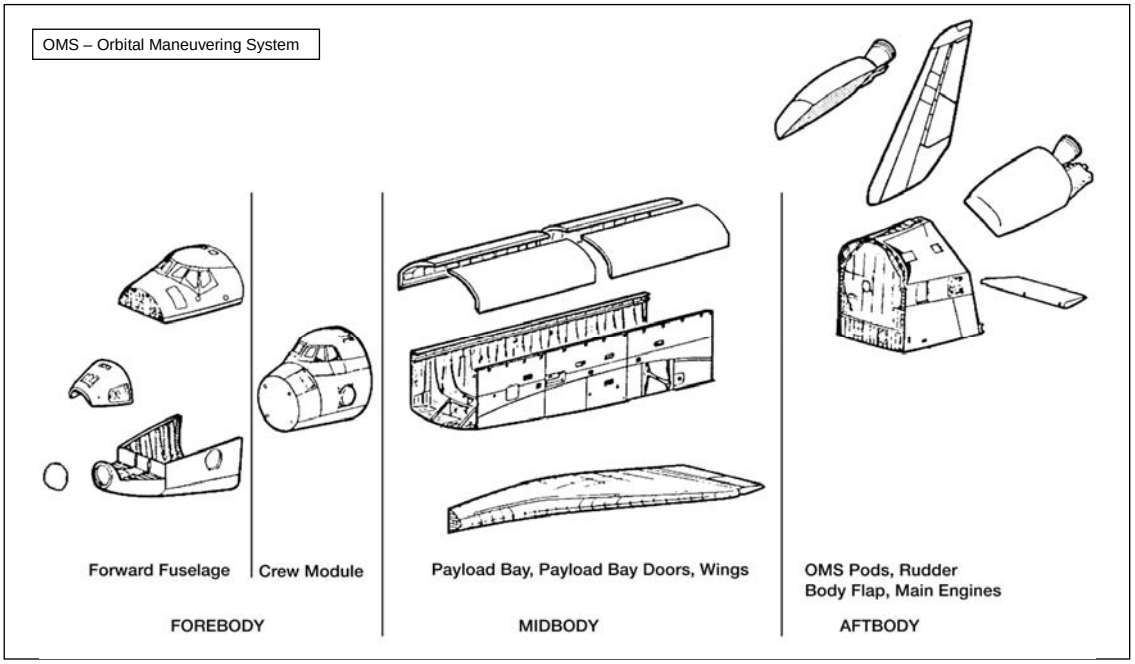


Figure 1.1-1. ***Depiction of the orbiter forebody, midbody, and aftbody elements.***

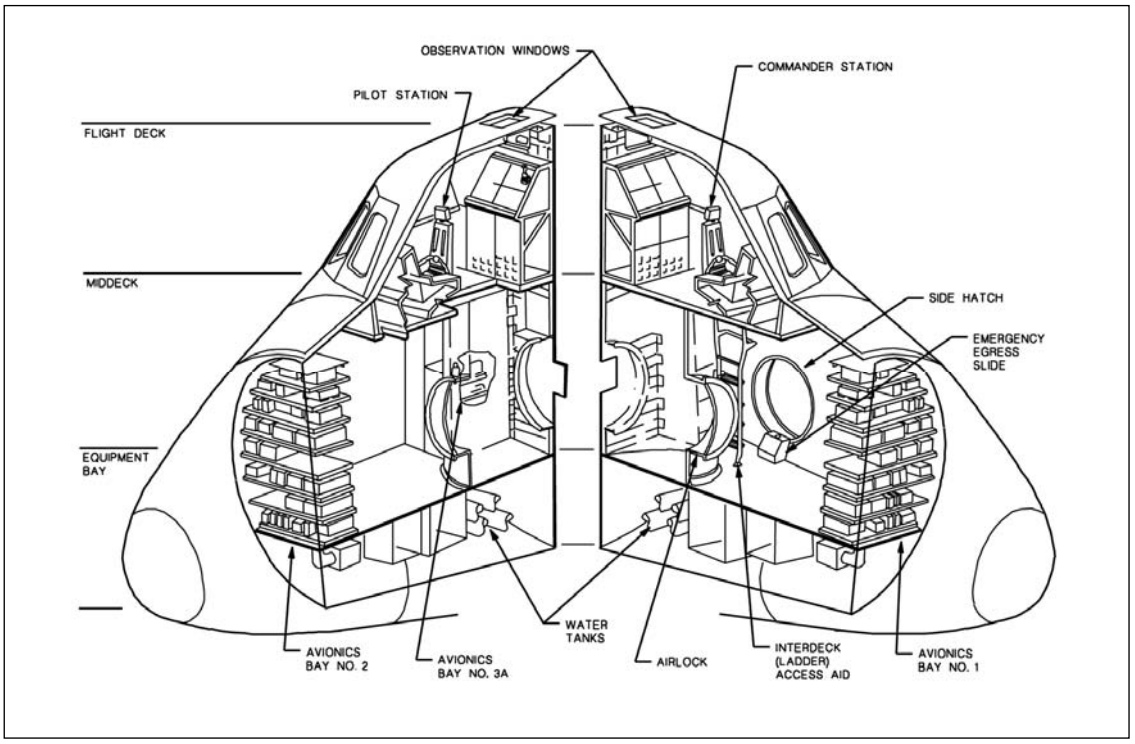


Figure 1.1-2. ***Depiction of the crew module flight deck and middeck within the forebody.*** **[Adapted from the Shuttle Operations Data Book]**

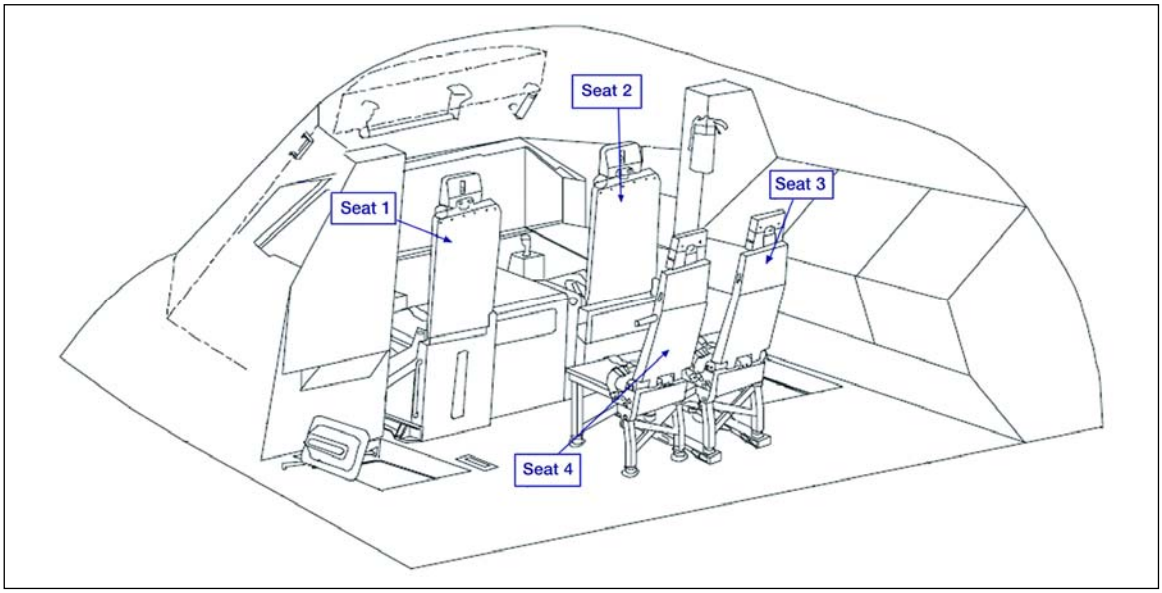


Figure 1.1-3. ***Depiction of the flight deck seats*****.**

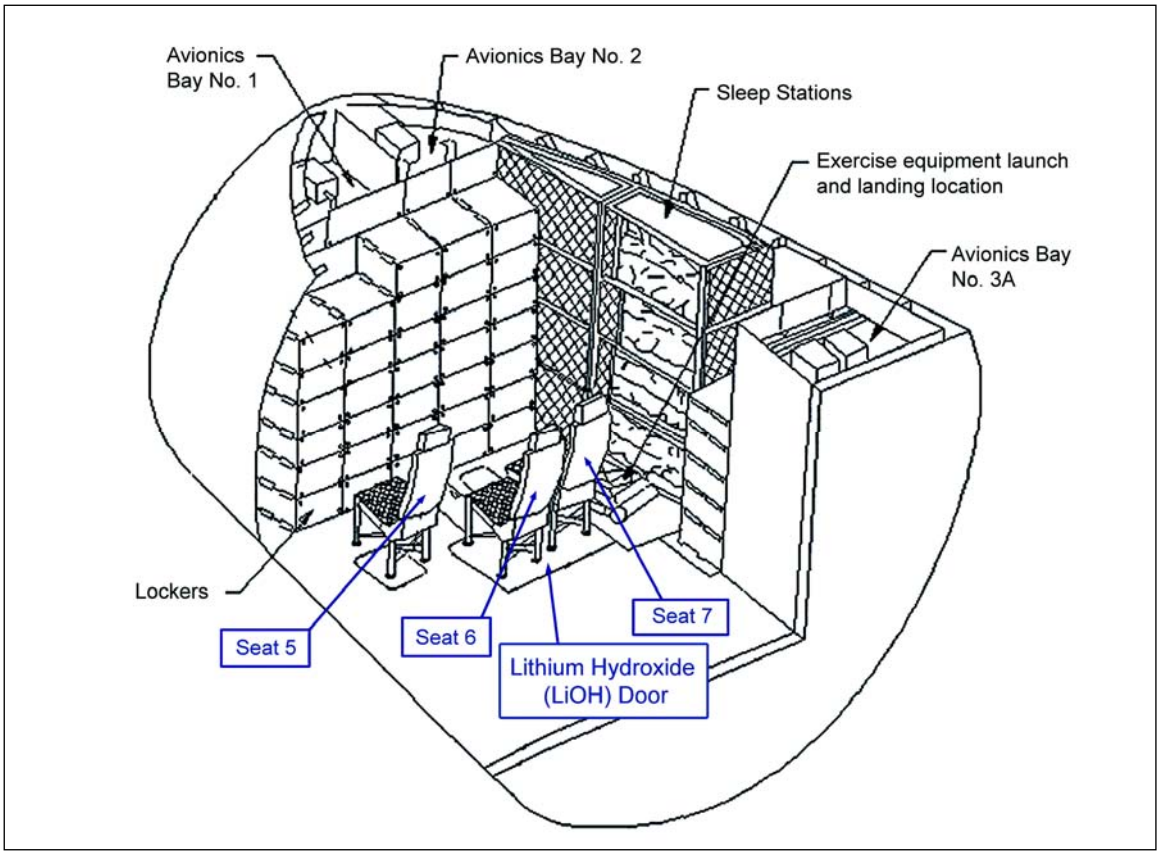


Figure 1.1-4. ***Depiction of the middeck seats*****. [Adapted from the Shuttle Crew Operations Manual]**

This timeline begins at Greenwich Mean Time (GMT) 09:15:30 (entry interface (EI)–16119 seconds) and ends at GMT 14:35:00 (EI+3051) (by which time most debris items had impacted the ground). This timeline overlaps with the latter portion of the timeline of the *Columbia* Accident Investigation Board (CAIB) Report.[2] The cause of the mishap will not be discussed because it was fully covered in the CAIB Report.

This timeline is divided into six phases, based on key events. Each phase of the timeline is addressed in sequence.

- Phase 1 [GMT 09:15:30 (EI–16119) to GMT 13:44:09 (EI)]: From the beginning of the deorbit preparation portion of the mission to EI. The deorbit preparation timeline begins 4 hours prior to the deorbit burn. After the burn, the orbiter descends in altitude until atmospheric drag effects become noticeable, roughly at EI. EI is defined as the time the orbiter descends through an altitude of 400,000 feet. At EI, *Columbia* was approximately 4,300 nautical miles from the landing site, traveling in excess of Mach 24.
- Phase 2 [GMT 13:44:09 (EI) to GMT 13:59:32 (EI+923)]: From EI to loss of signal (LOS). LOS is the loss of voice and real-time data transmissions from *Columbia*.
- Phase 3 [GMT 13:59:32 (EI+923) to GMT 14:00:18 (EI+969)]: From LOS to the Catastrophic Event (CE). The CE is defined as the initiation of the orbiter breakup into the primary subcomponents of the forebody, midbody,[3] and aftbody.[4] The CAIB timeline ends with the CE.
- Phase 4 [GMT 14:00:18 (EI+969) to GMT 14:00:53 (EI+1004)]: From the CE to the Crew Module Catastrophic Event (CMCE). The CMCE is defined as the initiation of the forebody breakup.
- Phase 5 [GMT 14:00:53 (EI+1004) to GMT 14:01:10 (EI+1021)]: From the CMCE to Total Dispersal (TD). TD is defined as the time when the crew module was substantively broken down into subcomponents and was no longer visible on ground-based videos.
- Phase 6 [GMT 14:01:10 (EI+1021) to approximately GMT 14:35:00 (EI+3051)]: From the TD to ground impact of the crew remains and the majority of the crew module debris.

Figure 1.1-5 shows the overall graphical timeline. This timeline represents the best fit to known and inferred data, but it is subject to some inherent uncertainty.

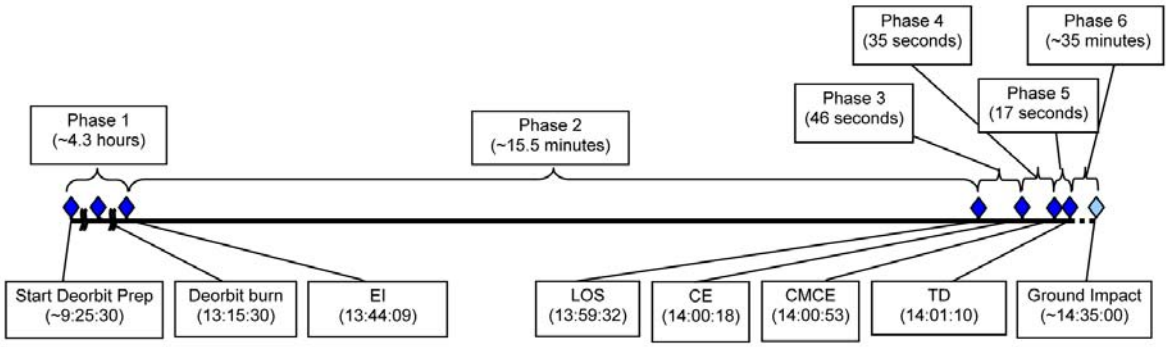


Figure 1.1-5. *Overall timeline.*

[2]*Columbia* Accident Investigation Board Report, Volume I, August 2003, pp. 40-41.
[3]The midbody consists of the payload bay and wings.
[4]The aftbody consists of the aft fuselage structure and internal components, the main engines, the left and right OMS pods (including the aft RCS), and the vertical tail.

The early portion of the timeline was developed from objective data such as on-board and downlinked vehicle instrumentation data, on-board video data recovered from the debris, and air-to-ground crew communications. As the timeline progresses into the later phases, the analysis increasingly relied on derived data such as results from ballistic analyses, thermal analyses, ground-based video, and recovered debris. All available data sets were integrated for this analysis.

The environmental conditions experienced by the crew are the focus for each phase. The conditions of interest are atmospheric pressure, thermal situation, and acceleration.[5] The data and methods used to understand the environment are described individually for each phase, and known and inferred events are integrated with the environmental information into a time-based sequence.

1.1.2.1 *Phase 1: Deorbit preparation to entry interface*
[GMT 09:15:30 (EI–16119) to GMT 13:44:09 (EI)]
4 hours, 28 minutes, 39 seconds in duration

This section discusses events affecting the crew from the deorbit preparation timeframe (4 hours prior to the deorbit burn) until EI, about 4 hours and 20 minutes in total. Figure 1.1-6 shows a timeline of this phase. All times are in GMT.

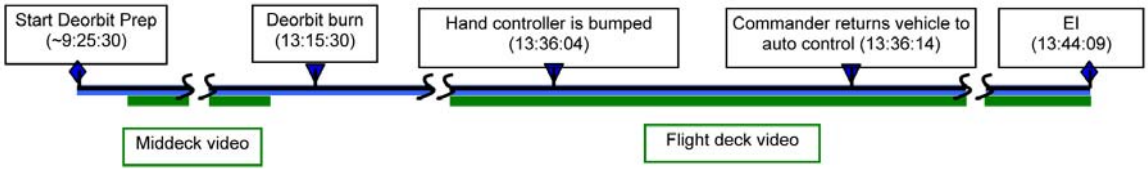


Figure 1.1-6. *Phase 1 timeline with key events*. Green bars represent times when video data are available. Blue bar represents when voice and telemetry transmissions are available (throughout this phase).

The principal sources of data for this phase are orbiter transmissions and recovered on-board video of middeck and flight deck activities. Transmissions include vehicle general purpose computer (GPC)-generated telemetry and audio transmissions made by the crew. The recovered middeck video shows the crew involved in deorbit preparation checklist activities about 2 hours prior to EI from approximately GMT 11:40:00 (EI–7449) to GMT 12:10:00 (EI–5649). The recovered flight deck video shows the flight deck crew seated and preparing for entry. This video was time-synchronized with audio transmissions and crew keystrokes recorded on the ground. The time duration of the video spanned across EI, showing activities from GMT 13:35:34 (EI–515) to GMT 13:48:45 (EI+276).

The crew performed cabin stow activities the day prior to entry and on the morning of entry day. Items were stowed and secured to prevent articles from coming loose in the cabin during entry. Objects aboard the orbiter are stowed in lockers, or are bagged and strapped down in the airlock and in the SPACEHAB module. The only "loose" items on the middeck or flight deck are clips, kneeboards, checklists, timers, writing instruments, drink bags used for fluid loading, and some crew escape equipment (CEE) prior to donning.[6] Any items that are not worn are restrained with VELCRO® or tethers.

The recovered middeck and flight deck videos show that all seats in the crew module were installed and the escape pole was in the process of being installed (figure 1.1-7). Recovered debris analysis shows that the pole installation was completed.

[5]Discussions of acceleration will use the Aerospace Medical Association convention. References to the standard gravitational acceleration of the Earth will use the lowercase "g." References to acceleration acting on objects and crew members in multiples of the Earth's gravitational acceleration will use the uppercase "G."
[6]CEE consists of things such as the g-suit, advanced crew escape suit (ACES), parachute, etc. See Section 3.2 Crew Worn Equipment for details.

The cabin stow and deorbit preparation portion of a shuttle mission is a busy period; according to many experienced crew members, shuttle crews often struggle to complete all actions in the time allotted, giving priority to time-critical orbiter systems activities. It is an accepted operational practice for a crew to reorder the tasks as necessary. The middeck video, which ended approximately an hour before the deorbit burn, indicates that the *Columbia* crew members were using their discretion to order their tasks. As a result, the middeck video cannot be precisely time-synchronized using either the published checklist or the crew-specific plan.

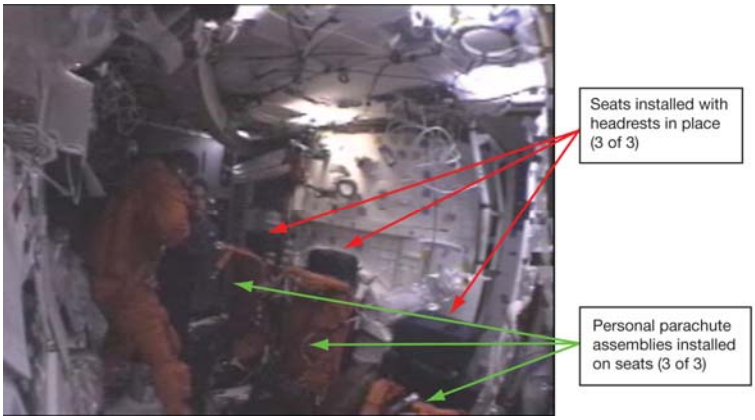


Figure 1.1-7. ***This early image from the recovered middeck video shows progress of the crew in deorbit preparation tasks.***

At 45 minutes prior to time of ignition (TIG)–45 minutes), the Commander (CDR) and Pilot (PLT) began working tasks in the entry checklist. By TIG–30 minutes, the rest of the crew should have completed items in the deorbit preparation checklist and transitioned to the entry checklist.

Deorbit burn occurred at GMT 13:15:30 (EI–1719/TIG+0). The burn was nominal, and *Columbia* began entry into the Earth's atmosphere. Per the checklist, a few tasks remain to be completed after the burn, including stowing the last laptop computer, which requires a crew member to be out of the seat. Crew equipment configuration items on the entry checklist (all crew members seated and strapped in, helmets and gloves donned, and suit pressure checked) were not entirely completed prior to EI. At least one crew member was not wearing the helmet and several were not wearing gloves.

The flight deck video shows that conditions on the flight deck were nominal during the entire time of the video recording. The video shows the flight deck crew finishing most checklist tasks close to the planned times. However, one flight deck crew member did not yet have gloves in place in time for the ACES pressure check. One event of note occurred at GMT 13:36:04 (EI–485/TIG+1234) when the CDR bumped the rotational hand controller (RHC) accidentally. Movement of the RHC out of the centered position caused the digital autopilot (DAP) to "downmode" from the "Auto" mode to "Inertial" mode. When this occurred, a "DAP DOWNMODE RHC" caution and warning message was displayed, the INRTL button on the C3 panel was illuminated, and a tone, which can be heard in the recovered flight deck video, was annunciated. An immediate reactivation of the autopilot was performed by the CDR. The capsule communicator (CAPCOM) in the Mission Control Center (MCC) then requested the CDR to enter "another Item 27," which is a command to fully recover the vehicle attitude from the bumped RHC.

Bumping of the RHC is a relatively common occurrence by either the PLT or the CDR because the ACES is bulky and the area near the controls is confined. Such RHC bumps with prompt recovery represent a very low hazard to the crew. The original design specifications of the orbiters were for a shirtsleeve environment (i.e., no special clothing needed to be worn). Although pressure suits have been worn during launch and entry since the *Challenger* accident, no modifications were made to displays and controls to accommodate the ACES.

1.1.2.2 *Phase 2: Entry interface to loss of signal* [GMT 13:44:09 (EI) to GMT 13:59:32 (EI+923)] 15 minutes, 23 seconds in duration

This section discusses events affecting the crew from EI (GMT 13:44:09) to LOS (the last audio and real-time telemetry transmission received from *Columbia*) at GMT 13:59:32 (EI+923).[7] This section discusses *Columbia*'s entry and the minimal indications available to *Columbia*'s crew and the MCC that *Columbia*'s structure was compromised. Ground-based video of the orbiter's flight is first available in this phase. This phase was approximately 15 minutes long. Figure 1.1-8 shows the timeline and key events for Phase 2.

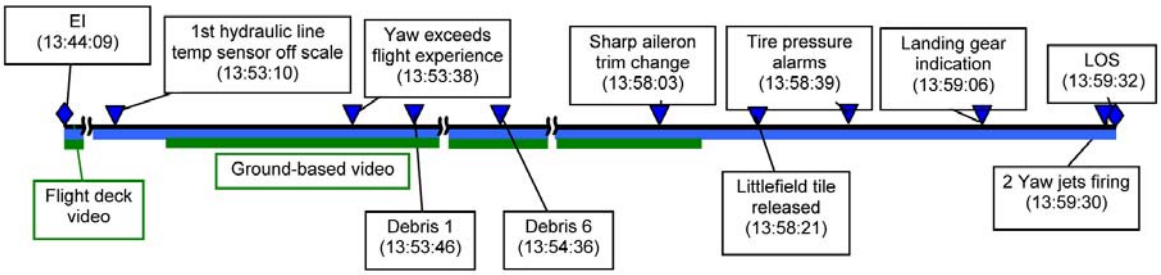


Figure 1.1-8. *Phase 2 timeline with key events*. Green bars represent times when video data are available. Blue bar represents when the Modular Auxiliary Data System/orbiter experiment (MADS/OEX) recorder data, and voice and telemetry transmissions are available (throughout this phase).

The sources of information that provided data for the investigation of Phase 2 are: orbiter transmissions; the MADS/OEX recorder; recovered flight deck video; ground-based video; and recovered debris. The MADS/OEX recorder was an on-board data collection recording system located in the crew module. Recorded data parameters consisted of structural temperature, strain, and accelerations from sensors located throughout the orbiter and concentrated in the left wing. This system was unique to *Columbia*. The MADS/OEX system did not display data to the crew or transmit telemetry to the MCC. Consequently, no MADS/OEX data were available in real time. The MADS/OEX recorder was recovered intact in the debris field and the data were recovered. The recovered flight deck video, beginning in Phase 1 at GMT 13:35:34 (EI–515), contains data through GMT 13:48:45 (EI+276), 4 minutes into this phase. Ground-based imagery recorded the entry of *Columbia* from the coast of California to the final breakup over Texas, with a gap in coverage from eastern New Mexico to western Texas.

Crew cabin pressure and the thermal environment inside the cockpit were nominal throughout this phase. At EI, atmospheric drag on the orbiter began to gradually increase. The initial roll and subsequent roll reversals caused accelerations of up to 0.8 G during entry. During this period of a shuttle mission, crew members typically experience heaviness, dizziness, and sometimes stomach awareness or mild nausea. Based upon telemetry, accelerations were nominal despite the damage-induced aerodynamic changes to the orbiter. The orbiter was shedding debris during at least part of this phase, although changes in mass properties were small and no detectable load spikes were noted by the MCC, reported by the crew, or found in post-mishap analysis data. No anomalous orbiter systems conditions were displayed to the crew until the end of this phase.

At EI, the vehicle also began to experience the thermal effects of the Earth's atmosphere. Shock waves due to the vehicle's hypersonic velocities and the frictional effect of the atmosphere began to heat the orbiter's surface. Temperatures on the surface of the orbiter during entry vary by location, with the nose and leading edge of the wings experiencing temperatures greater than 2,800°F (1,538°C). The Thermal Protection

[7] Although LOS is defined here as the loss of audio and real-time data transmissions, some instrument-based telemetry was received shortly into Phase 3; this will be discussed in the next section.

System (TPS) of the orbiter consists of reinforced carbon-carbon (RCC) panels, tile, and thermal blankets, and is designed to protect the orbiter's structure from this nominal entry heating.

The CAIB concluded that the breach in the TPS on the leading edge of the left wing allowed hot gas to penetrate the wing and to work its way aft and inboard, toward the midbody fuselage and left main landing gear wheel well (figure 1.1-9). CAIB analysis showed that this eventually caused significant internal damage to *Columbia*'s left wing, changing the wing's aerodynamic properties. There were no indications to the crew of this ongoing damage. Throughout phase 2, the orbiter Flight Control System (FCS) corrected for the damage-induced yaw and roll moments, and control of the orbiter was maintained.[8]

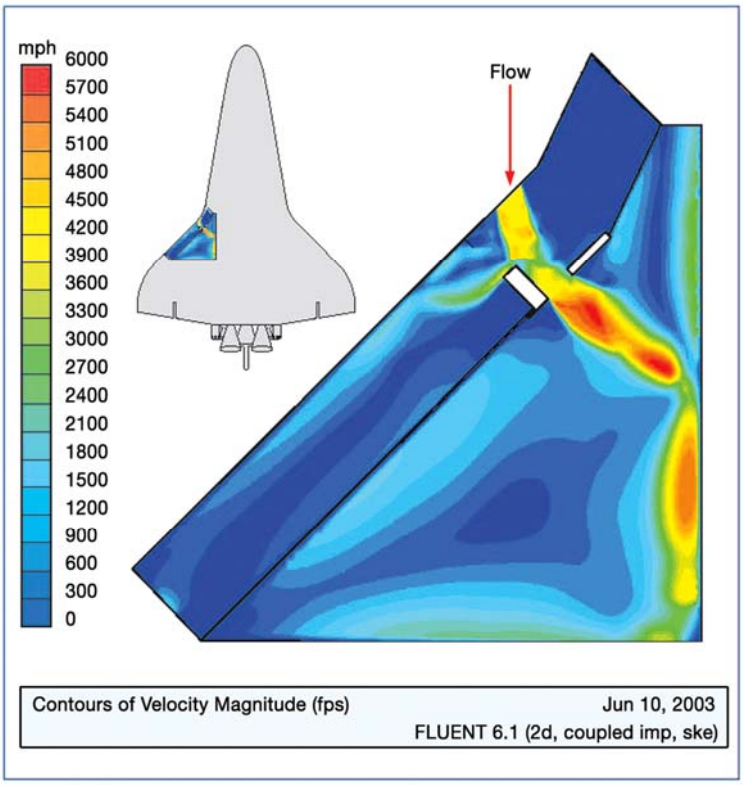


Figure 1.1-9. ***The effects of wing Thermal Protection System damage as reported by the* Columbia *Accident Investigation Board.***

The recovered flight deck video shows the CDR requesting a suit pressure integrity check from the other crew members. Suit pressurization checks for the CDR, PLT, and Mission Specialist (MS) 4 were observed in the video and validated with telemetry of the oxygen (O_2) system. After the ACES pressure checks, the crew members turned off the flow of O_2 to the ACES and opened their visors. The O_2 must be turned off because the suit vents O_2-enriched air into the cabin with the visor down and O_2 flowing. Venting O_2-enriched air eventually creates an increased concentration of O_2, leading to an increased fire hazard. To prevent carbon dioxide buildup inside the helmet, visors are returned to the open position. Open visors have the added benefit of improved crew comfort and communication. Telemetry data were consistent with one or two more suit pressure integrity checks occurring after the end of the flight deck video. It could not be determined which crew member(s) performed these checks. The video, which ended at GMT 13:48:45 (EI+276), shows the four flight deck crew members suited, seated, and strapped in with helmets donned. All except one had fully donned and connected gloves. The recovered flight deck video indicates that the crew was not aware of any problems.

At GMT 13:49:32 (EI+323), a nominal roll to the right was completed for energy management. At GMT 13:50:53 (EI+404), *Columbia* started the expected 10-minute window of nominal peak heating.[9]

At GMT 13:51:46 (EI+457), the inertial sideslip (yaw) angle began a negative trend (yaw to the left), although the angle remained within previous flight experience for almost 2 minutes (figure 1.1-10). At GMT 13:52:05 (EI+476), the yaw moment changed due to increased drag from the damaged left wing; the orbiter's FCS commanded the aileron trim to compensate (figure 1.1-11). Neither the yaw moment change nor the aileron trim change was obvious to either the MCC or the crew as an off-nominal condition, although post-accident analysis concluded that this was the first indication of the orbiter's response to the

[8] *Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 78.
[9] *Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 38.

changing aerodynamic properties brought about by the left wing damage. Otherr post-mishap analysis determined that damage inside the wing began no later than GMT 13:52:17 (EI+488).[10]

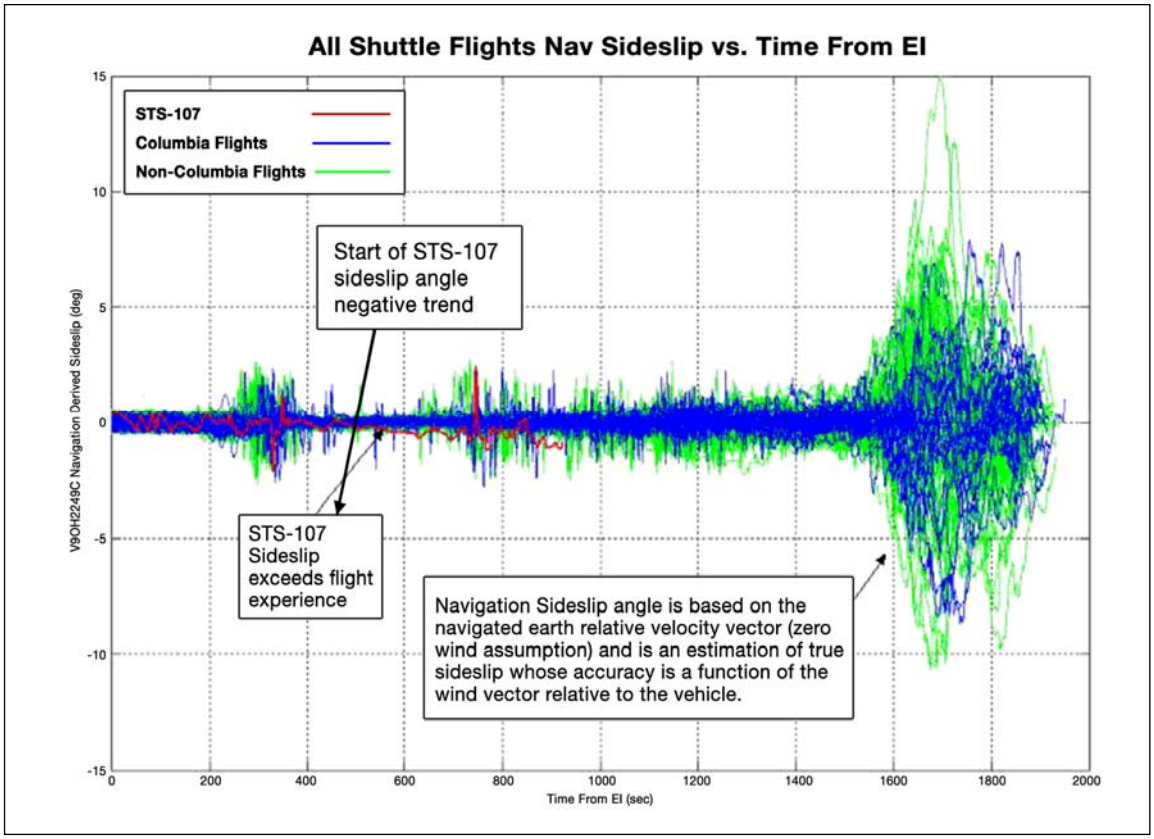


Figure 1.1-10. ***Negative trend starts at GMT 13:51:46 (EI+457) and exceeds flight experience at GMT 13:53:38 (EI+569).***[11]

[10]*Columbia* Accident Investigation Board Report, Volume I, August 2003, pp. 68 and 71.
[11]Integrated Entry Environment Team Final Report, May 30, 2003, Figure 6.6-2, p. 30; taken verbatim from document.

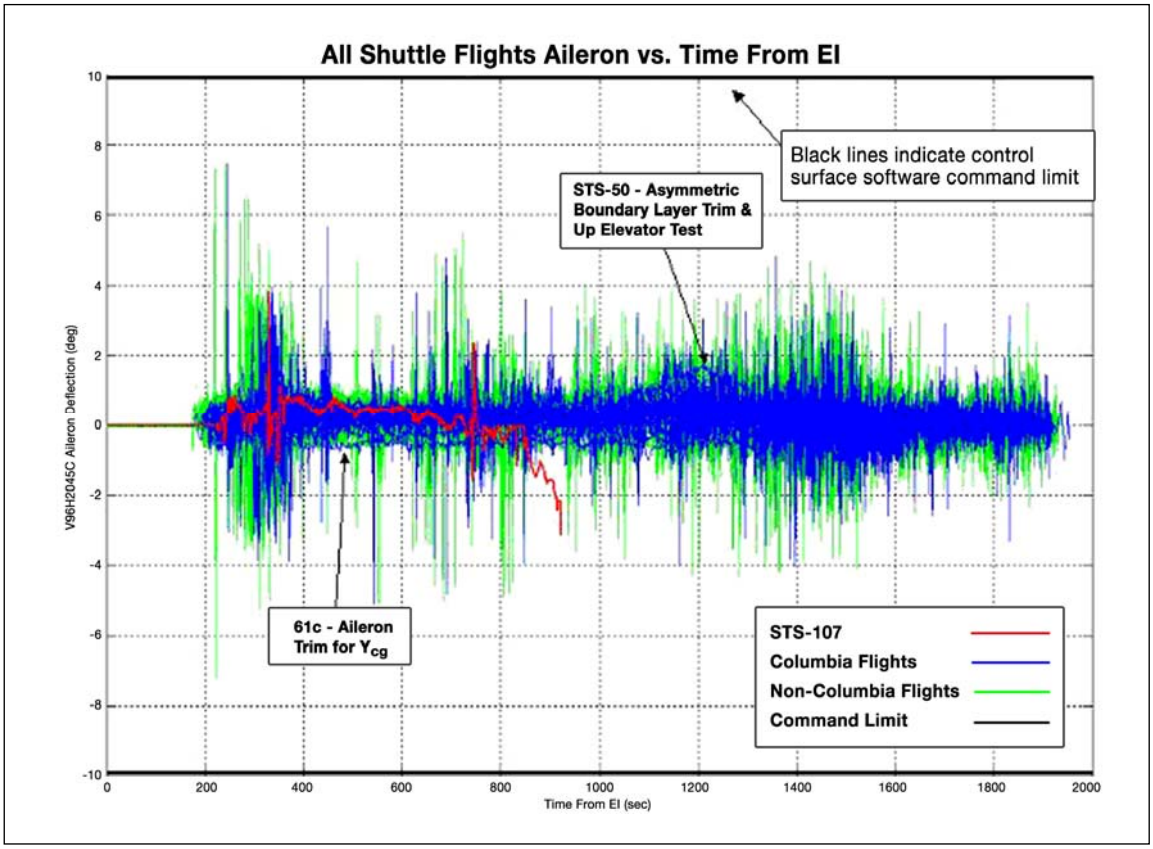


Figure 1.1-11. ***The sharp divergence from previous flight experience starts at approximately GMT 13:58:19 (EI+850).***[12]

The left main landing gear brake line in the inboard sidewall of the wheel well began to show an off-nominal temperature rise rate at GMT 13:52:17 (EI+488).

MCC personnel became aware of an off-nominal flight condition when four hydraulic return line temperature sensors in the left wing went off-scale low from GMT 13:53:10 (EI+541)[13] to GMT 13:53:36 (EI+567). A sensor suddenly going off-scale usually indicates a failure of the sensor or the wiring. Loss of an individual sensor for various reasons has occurred in previous missions. However, the simultaneous failure of multiple sensors from separate redundant systems was an event outside previous flight experience. These temperature data were not available to the crew and the crew was not notified of these indications. The loss of sensors generated concern in the MCC, and investigation by the flight control team began immediately.

Ground-based video coverage of *Columbia* was acquired by videographers unassociated with NASA at GMT 13:53:15 (EI+546).[14] Figure 1.1-12 is the first ground-based image of *Columbia* acquired. The bright spot circled is the "orbiter envelope" and is the nominally produced hot gas and plasma that surround the orbiter during entry. The actual shape of the orbiter (and most debris) cannot be seen in any non-telescopic-based video; only the surrounding hot gas/plasma envelope is visible.

[12]Integrated Entry Environment Team Final Report, May 30, 2003, Figure 6.6-1, p. 30; taken verbatim from document.
[13]*Columbia* Accident Investigation Board Report, Volume I, August 2003, description of these sensor failures.
[14]All video frames of the orbiter prior to GMT 13:59:32 (EI+923) were processed by the STS-107 Image Analysis Team. Details of video processing can be found in the *Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, October 2003. All videos are assumed to have an approximately 1-second error.

Columbia crossed the California coastline at GMT 13:53:26 (EI+557).

At GMT 13:53:38 (EI+569), the sideslip angle (left yaw) exceeded all previous flight experience (figure 1.1-10).

The first known debris shedding event on entry was identified as Debris 1 and most likely originated from the left wing. Debris 1 becomes visible on ground-based video at GMT 13:53:46 (EI+577). Later ballistic analysis estimated a release time of GMT 13:53:44.8 (EI+575).[15] Luminosity measurements and calculated rates of deceleration were used to determine that the mass was < 8 lbs.[16] Figure 1.1-13 shows the orbiter and the debris.

Figure 1.1-12. ***This is the first frame of ground-based video. Columbia*** **is circled. Time 1 (TM1) shows the Greenwich Mean Time. The Pacific Standard Time displayed on the lower portion of the image is inexact.**

The Mechanical, Maintenance, Arm, and Crew Systems (MMACS) officer in the MCC notified the Flight Director of the off-scale low hydraulic line temperature sensors at GMT 13:54:24 (EI+615).

The brightest debris shedding event that occurred in this phase, Debris 6, is first visible on video at GMT 13:54:36 (EI+627) (figure 1.1-14). Ballistic estimates determined that the actual release time was 4 seconds earlier. Luminosity measurements and calculated rates of deceleration were used to determine that the mass was probably a few hundred pounds.[17] There were no data from sensors, instrumental indications, or apparent crew recognition of this debris loss.

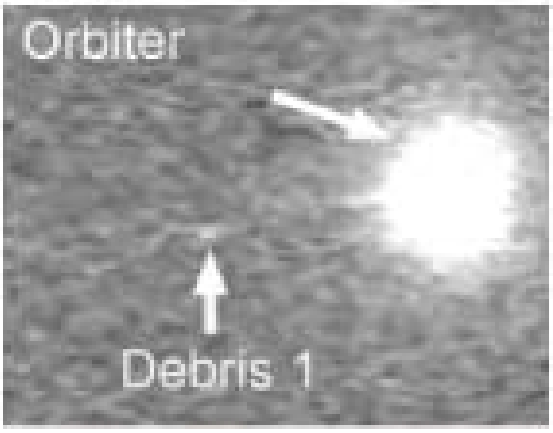


Figure 1.1-13. ***Video capture of the first observed incident of debris being shed.*** **The orbiter is traveling from left to right in this image.**

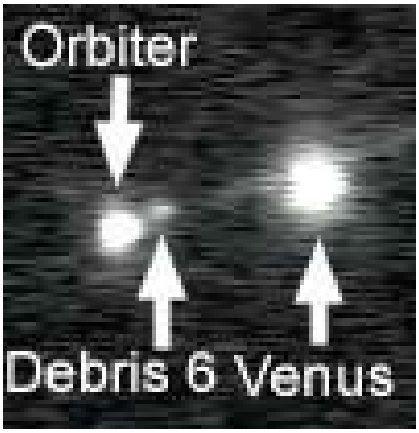


Figure 1.1-14. ***The brightest debris event known to have occurred prior to loss of signal.*** **The orbiter is traveling from right to left.**

[15]All ballistic analysis release times have a ±5-second uncertainty.
[16]*Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, October 2003, p. 110.
[17]*Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, October 2003, p. 110.

At approximately GMT 13:54:30 (EI+621), cabin O_2 partial pressure and cabin pressure telemetry indicated signatures consistent with additional ACES pressurization events, indicating that the crew was continuing suit activities.

The first planned roll reversal was initiated from right wing low to left wing low at GMT 13:56:30 (EI+741).

At approximately GMT 13:58:03 (EI+834), the aileron trim begins to diverge sharply from the expected values to counteract the increasing adverse moments due to the left wing damage (figure 1.1-15).

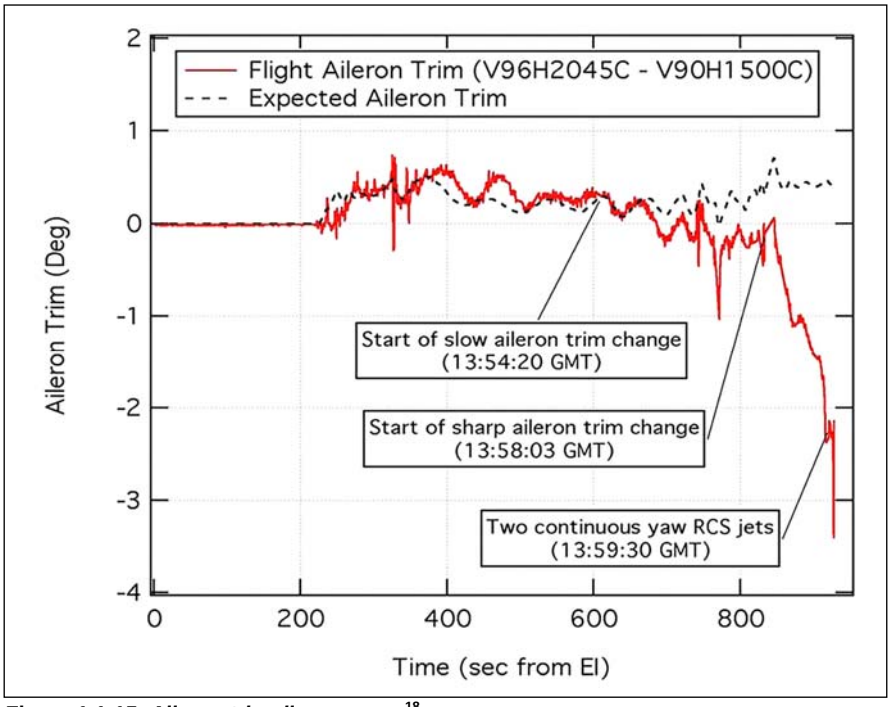


Figure 1.1-15. *Aileron trim discrepancy*.[18]

Western ground-based video coverage ends at GMT 13:58:12 (EI+843).

Ballistic analysis indicates that the westernmost piece of recovered *Columbia* debris was released at GMT 13:58:21 (EI+852). This debris, a tile from the left wing upper surface located just inboard of RCC panels 8 and 9, was found in Littlefield, Texas[19] (figure 1.1-16).

Figure 1.1-16. *The Littlefield tile*. [Picture from the *Columbia* Reconstruction Database, debris item no. 14768]

[18]Integrated Entry Environment Team Final Report, May 30, 2003, Figure 6.3-4, p. 20.
[19]STS-107 *Columbia* Reconstruction Report, NSTS-60501, June 2003, pp. 21 and 121.

Columbia's crew received the first indication of a problem at GMT 13:58:39 (EI+870) when the first of four fault messages was annunciated on the on-board Backup Flight Software monitor. These messages were accompanied by an audible tone. The fault messages indicated a loss of pressure on the left main landing gear tires. These indications also were presented to the flight control team in the MCC. The CDR and PLT called up the fault page for these messages and reviewed the information. The failure the crew saw would be familiar, although slightly different from what they saw in training. One of the failure scenarios the crew practiced during training was a circuit breaker trip that resulted in one-half of the tire pressure sensors being disabled. A circuit breaker trip would disable some sensors for all of the tires (left main gear, right main gear, and nose gear), but the failure signature during the accident involved all the tire pressure sensors on the left main gear only. At GMT 13:58:48 (EI+879), the crew began a call to the MCC but the call was broken and not repeated. Brief interruptions of communications often occur due to the tracking and data relay satellite antenna pointing angles changing relative to the orbiter's transceivers. This specific dropout of communication was not unexpected.

At GMT 13:59:06 (EI+897), 10 seconds after the fourth of four tire pressure fault messages, telemetry indicated the "LEFT MAIN GEAR DOWN" lock sensor transferred to "ON." Other sensors indicated that the landing gear door was still closed and the landing gear was locked in the up (stowed) position. These mixed signals caused the left landing gear position indicator to display a "barber pole" (figure 1.1-17), which indicates an indeterminate landing gear position. Post-accident analysis of the data and recovered debris indicates that the left landing gear was locked in the "up" position and the landing gear door was closed. The signal indicating that the gear was down was a false signal that was likely triggered by damage to the sensor system (sensor, wiring harness, etc.). Based on training experience, the crew was probably attempting to diagnose the situation given that it involved the same landing gear as the tire pressure messages and indicated a potential landing gear deployment problem.

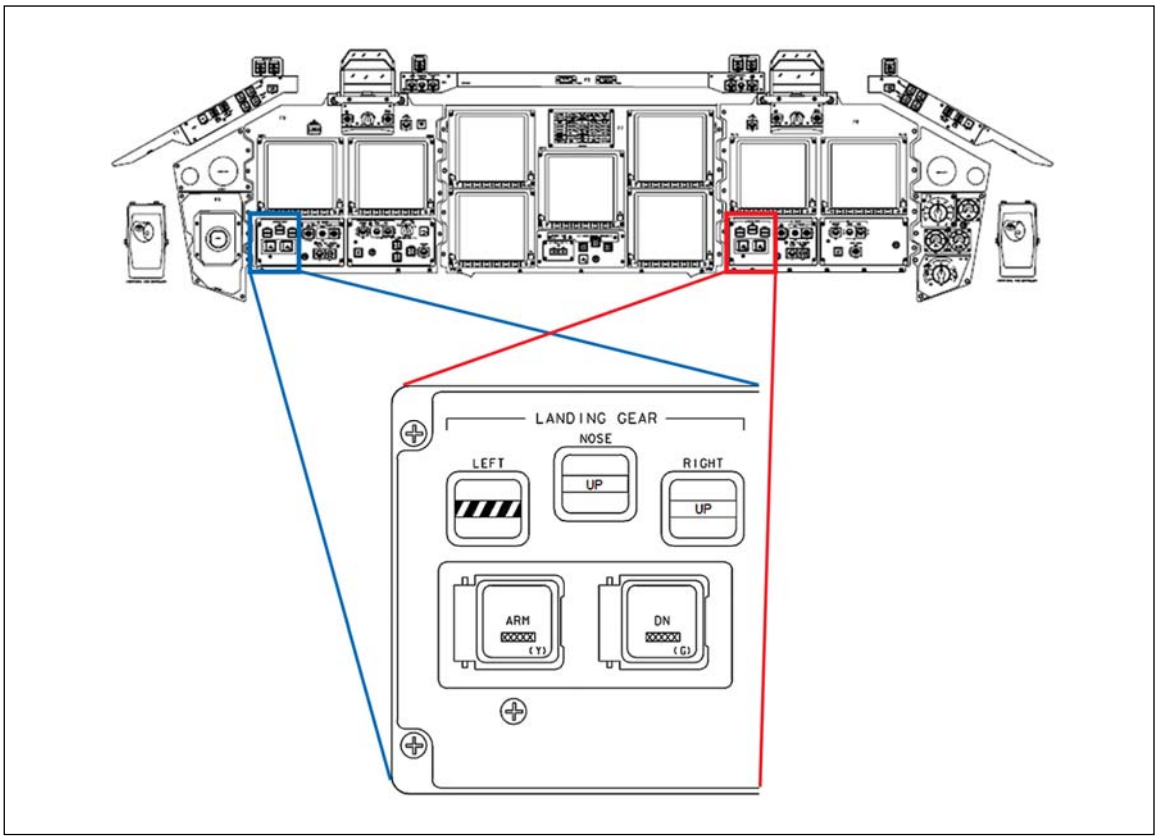


Figure 1.1-17. ***Landing gear indicator panel, identical on both sides of the flight deck forward display panels*****. Left indicator showing "barber pole" (indeterminate position). [Adapted from the Space Shuttle Systems Handbook]**

At GMT 13:59:29 (EI+920), the orbiter yaw and roll rates exceeded the ability of the aileron trim to compensate for the changing drag of the deformed left wing. One second later, the R2R and R3R RCS jets[20] activated. Typically, RCS jets pulse throughout entry, adjusting the orbiter's flight path as needed. RCS jets had been pulsing nominally until this time when R2R and R3R began firing continuously as the orbiter attempted to counteract the increased left wing drag and resulting yaw moment. A small light on a panel in front of the CDR would have become illuminated continuously (figure 1.1-18). Experience shows that this jet fire light is not easily noticed.

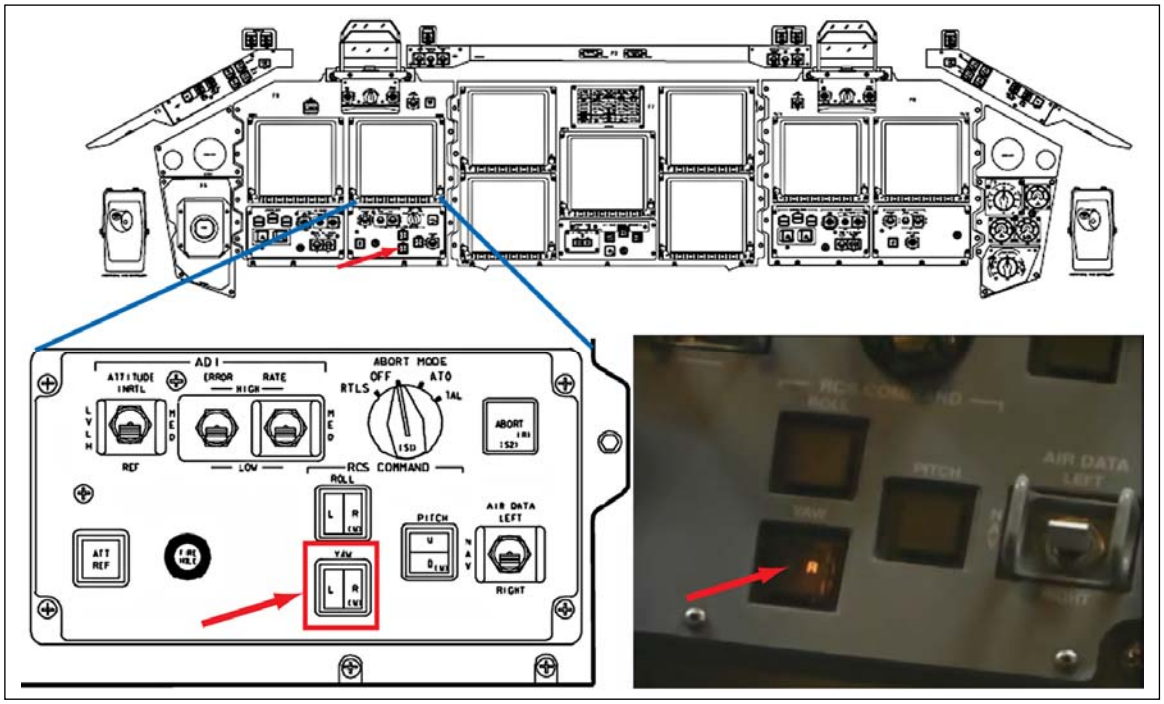


Figure 1.1-18. *Reaction Control System thruster status display*. [Pictures (top and bottom left) adapted from the Space Shuttle Systems Handbook; picture (bottom right) from the Shuttle Engineering Simulator]

At GMT 13:59:32 (EI+923), the crew acknowledged a call from the MCC but the crew's response was interrupted in mid-sentence ("Roger, uh …"[21]). This was the final call heard from *Columbia*. This is also the time of LOS, when all audio and real-time data to the MCC from *Columbia* was lost. A short dropout (seconds) was expected at this time based on pre-mission analysis as the orbiter switched from one communication satellite to another. The CAPCOM replied to the partial transmission to let the crew know that the flight controllers saw the tire pressure fault messages and that the MCC did not understand the last transmission.[22] The MCC personnel recognized that there were problems occurring with *Columbia*, but the telemetry signatures were such that these personnel were unable to complete analysis of the wide-ranging (and seemingly unrelated) problems before contact was lost.

There were no indications to the crew and the MCC that the loss of audio communications and real-time data was more than a brief condition. To all on-board appearances, *Columbia* only had a potential issue with landing gear deployment; a non-trivial event, but the crew had time to troubleshoot the problem. Changing drag on the left wing was just beginning to develop into a potentially recognizable problem.

[20]Right-firing RCS jets on the right OMS pod.
[21]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 43.
[22]The CAPCOM continued to attempt to contact the crew on different radio frequencies to re-establish voice communications.

Based on seat debris and medical findings, at the end of this phase one middeck crew member had not fully ingressed the seat yet, although the action may have been in progress. This crew member was responsible for completing post-deorbit burn tasks and was assigned to be the last to ingress the seat.

1.1.2.3 *Phase 3: Loss of signal to Catastrophic Event* [GMT 13:59:32 (EI+923) to GMT 14:00:18 (EI+969)] 46 seconds in duration

This section discusses key events that affected the crew from LOS at GMT 13:59:32 (EI+923) to the CE, which began at 14:00:18 (EI+969) (figure 1.1-19). During this period of time at about GMT 13:59:37 (EI+928), loss of control (LOC) of *Columbia* occurred. LOC marks the beginning of the transition from controlled flight to an uncontrolled ballistic entry. This phase is 46 seconds long.

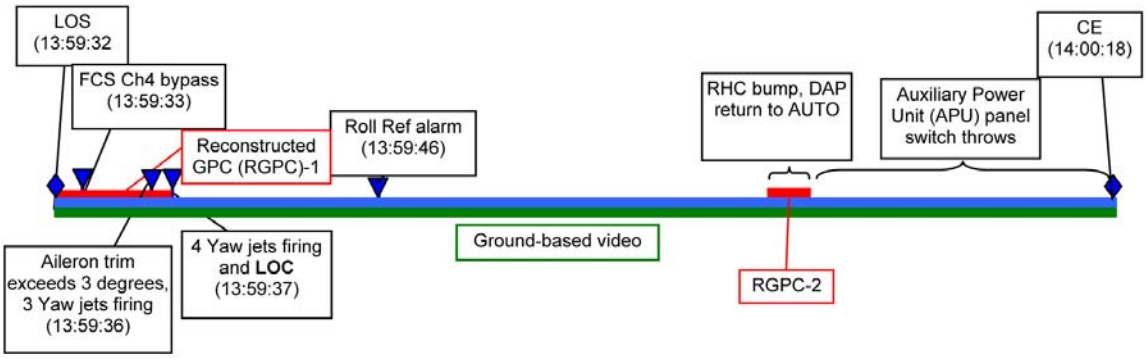


Figure 1.1-19. *Phase 3 timeline with key events*. Real-time voice and telemetry transmissions were not available in this phase or subsequent phases. The green bar represents time when video data are available; the blue bar represents when the Modular Auxiliary Data System/orbiter experiment recorder data are available (both available throughout this phase). Red bars represent times when reconstructed general purpose computer data were available.

Available instrumentation data (recorded and recovered) become scarce in Phase 3 (and nonexistent in subsequent phases). The sources of data used to reconstruct conditions in Phase 3 are: reconstructed telemetry; MADS/OEX recorder; ground-based videos; recovered debris; and aerodynamic and ballistics analyses. Reconstructed general purpose computer (RGPC) data were data that were recorded at the telemetry receiving ground station at White Sands, New Mexico, but not transmitted to the MCC in real time due to quality filtering. After the accident, the data were retrieved and manually reconstructed.[23] The RGPC data include real-time parameter data (such as pressures, temperatures, and switch positions), time-stamped alert messages, and non-time-stamped alert messages. Ground-based video was re-established starting at approximately GMT 13:59:32.5 (EI+923), at about LOS. For the first 16 seconds of this phase, a single video supplies coverage. Additional video coverage begins at GMT 13:59:48 (EI+939). Recovered *Columbia* debris was used for reconstruction via visual inspection, material sampling, and ballistic analysis. This led to conclusions regarding thermal events and material loads. Ballistic analysis was performed on select debris items to help understand the events and their sequence.

At GMT 13:59:33 (EI+924), data from RGPC-1 showed the primary software system annunciated that FCS Channel 4 had been automatically bypassed out of the control loop. This bypass occurred because of a failed wire bundle and resulted in a Master Alarm. The Master Alarm was annunciated visually and aurally. While there is no crew action associated with this frequently trained FCS fault message other than to perform a message reset, the crew may have called up a display to analyze the failure. Crews are trained to troubleshoot systems errors, and this crew would have been evaluating this new message along with the previous tire pressure and landing gear down-lock indications to assess whether there was a common

[23] *Columbia* Accident Investigation Board Report, Volume II, Appendix D.9, Data Review and Timeline Reconstruction Report, October 2003.

system fault that could account for all of these messages. It is unknown whether the increasing aileron trim and thruster firings were noticed by the flight deck crew members.

At GMT 13:59:36 (EI+927), the third RCS yaw jet, R4R, began firing continuously and aileron trim exceeded 3 degrees. There is no alarm associated with a deviating trim condition, and the crew is not expected to monitor the trim during this period of entry. At GMT 13:59:37 (EI+928), the fourth and last RCS yaw jet, R1R, began firing continuously.

To summarize, in the minute prior to LOC, the crew received several indications of various vehicle systems problems:

1. 58 seconds prior: the first of four tire pressure alert messages was displayed.
2. 31 seconds prior: left main landing gear indicator transitioned to an indeterminate state (no annunciated alarm).
3. 7 seconds prior: pulsing RCS yaw light became constant as two RCS jets began firing continuously (no annunciated alarm).
4. 4 seconds prior: FCS channel bypass message and Master Alarm.
5. 0.6 second prior: aileron trim exceeded 3 degrees (no annunciated alarm).

Ground-based video data show a brightening event at GMT 13:59:37 (EI+928).

RGPC-1 ends[24] at GMT 13:59:37.4 (EI+928) with an approximately 25-second gap in data until RGPC-2 data begins at GMT 14:00:02.660 (EI+953). RGPC-2 data include messages generated during the 25-second gap; some of the messages do not have time tags, and some message time tags are corrupted.

Vehicle LOC probably occurred at GMT 13:59:37 (EI+928).[25] The CAIB Report concluded that "During re-entry this breach in the TPS allowed superheated air to penetrate through the leading edge insulation and progressively melt the aluminum structure of the left wing, resulting in a weakening of the structure until increasing aerodynamic forces caused loss of control, failure of the wing, and breakup of the orbiter."[26] An in-depth review of the data by the Integrated Entry Environment team provided further insight into the probable sequence of events. The RGPC-2 data showed that a "ROLL REF" alarm was recorded at GMT 13:59:46 (EI+937), only 9 seconds after the end of RGPC-1. A ROLL REF alarm generally indicates that the drag of the orbiter has exceeded the entry drag profile. The Integrated Entry Environment team concluded that the most credible scenario that could cause this message within 9 seconds would be from a pitch deviation rather than a roll deviation (which would be expected with increasing drag on the wing).[27] A complete loss of hydraulics would cause the elevons and body flap to move to a floating position, resulting in an uncontrolled pitch-up. RGPC-2 data (approximately 25 seconds later) showed that the hydraulics systems failed, but no time signature was available to confirm when the loss occurred. Video data supported this time for LOC. The Spacecraft Crew Survival Integration Investigation Team (SCSIIT) concluded that the LOC occurred as a result of the loss of hydraulics at GMT 13:59:37 (EI+928). The loss of hydraulics likely occurred when all three redundant hydraulic systems lost pressure due to breaches in the hydraulic lines from thermal damage in the left wing. A visual simulation of the pitch-up associated with this type of LOC is shown in figure 1.1-20.

[24]Although data were received after GMT 13:59:37 (EI+928), none could be reconstructed until RGPC-2. See *Columbia* Accident Investigation Board Report, Volume II, Appendix D.19, Qualification and Interpretation of Sensor Data from STS-107, October 2003.
[25]This LOC time (GMT 13:59:37 (EI+928)) occurs 42 seconds earlier than that concluded in the *Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 73.
[26]*Columbia* Accident Investigation Board Report, Volume I, Executive Summary, August 2003, p. 9.
[27] EG-DIV-08-32- Integrated Entry Team Report, Appendix G - Post-LOS Analysis.

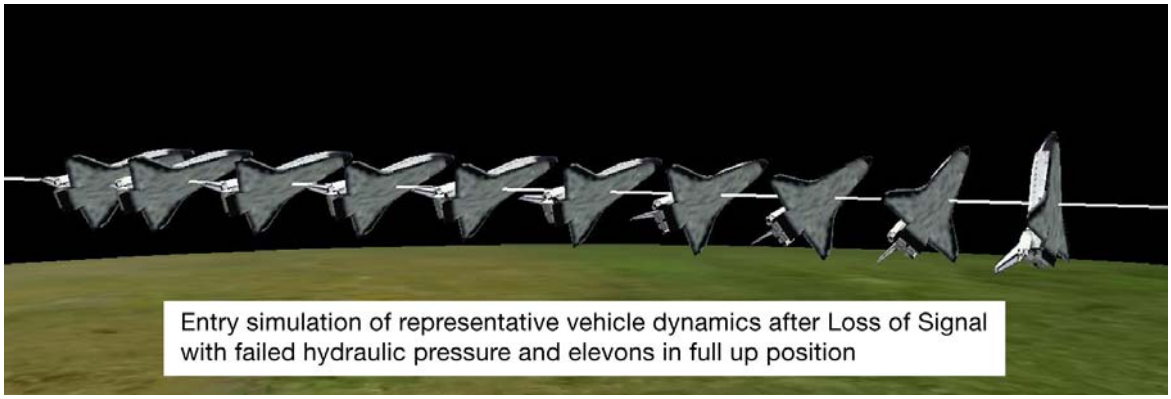


Figure 1.1-20. ***Sequence (1-second intervals) showing a simulation of orbiter loss of control pitch-up from GMT 13:59:37 (EI+928) to GMT 13:59:46 (EI+937).*** **White line indicates vehicle trajectory relative to the ground.**

The LOC event marked the beginning of the transition from a controlled gliding trajectory into an uncontrolled ballistic trajectory. Once attitude control was lost, orbiter heating, lift, and drag were dictated predominantly by ballistic number.[28] The out-of-control orbiter configuration had significantly more drag than the nominal entry configuration, and the trajectory became steeper. Changes in the flight profile are recognizable in the ground-based video as changes in the orbiter's trail and in the brightness of the visual signal ("brightening events").

Video imagery shows a dynamically changing orbiter trail after GMT 13:59:37 (EI+928) with a braided or corkscrew appearance, implying motion of the orbiter. However, the specific attitude of the orbiter cannot be derived from ground-based imagery. Brightening events, objects separating, "puffs," and splitting of the trail are all seen in the video during this timeframe. Ballistic analysis of debris could not positively correlate a specific orbiter source to shedding events seen in the video. However, it is known that the left wing and the left OMS pod were being compromised.[29] Figure 1.1-21 shows video frames from GMT 13:59:35.5 (EI+925) through GMT 13:59:43.5 (EI+934) and displays some of these dynamic changes, although they are much more clearly seen in the video.

[28]Ballistic number is affected by the coefficient of drag of an object (which changes with its velocity), its weight, and the area presented to the velocity vector. A low ballistic number indicates high drag. The initial trajectory of an object with a low ballistic number is steeper than the trajectory of an object with a high ballistic number. See Section 2.1.
[29]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 68.

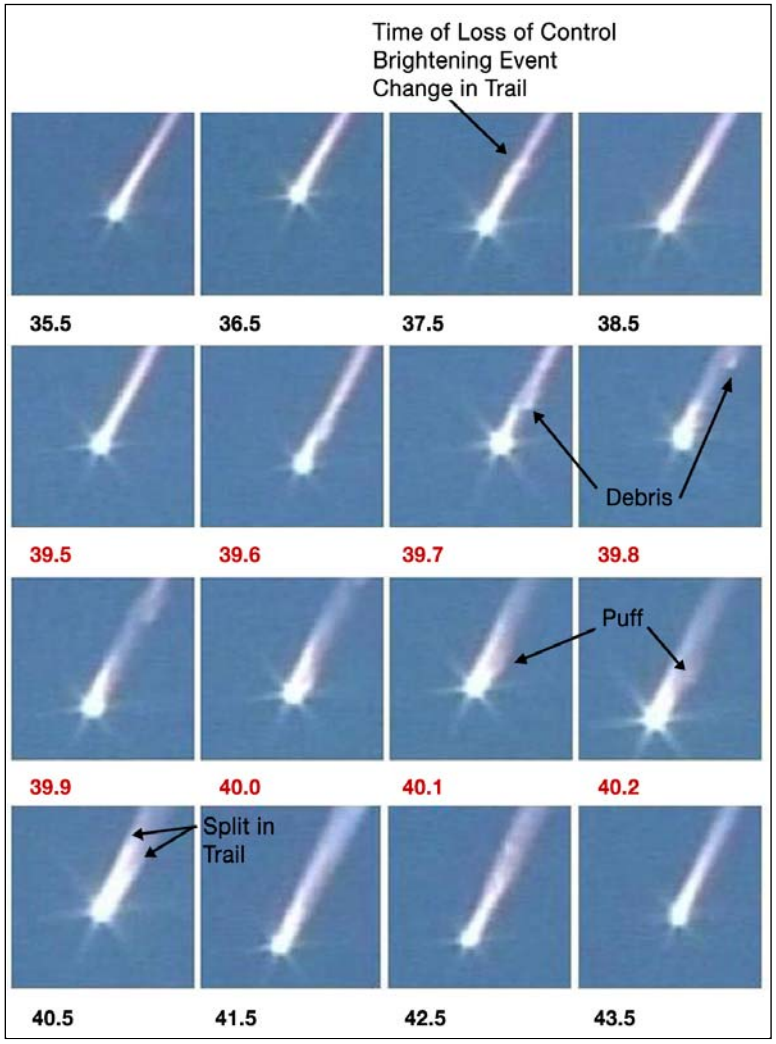


Figure 1.1-21. ***Video frame captures from ground-based video, beginning at GMT 13:59:35.5 (EI+926) and ending at GMT 13:59:43.5 (EI+937).*** **The numbers below each frame indicate the seconds after GMT 13:59:00. The frames in the first and last rows are 1 second apart. The frames in the second and third rows are 0.1 second apart.**

For the crew, the first strong indications of the LOC would be lighting and horizon changes seen through the windows and changes on the vehicle attitude displays. Additionally, the forces experienced by the crew changed significantly and began to differ from the nominal, expected accelerations. The accelerations were translational (due to aerodynamic drag) and angular (due to rotation of the orbiter). The translational acceleration due to drag was dominant, and the direction was changing as the orbiter attitude changed relative to the velocity vector (along the direction of flight).

Results of a shuttle LOC simulation show that the motion of the orbiter in this timeframe is best described as a highly oscillatory slow (30 to 40 degrees per second) flat spin, with the orbiter's belly generally facing into the velocity vector. It is important to note that the velocity vector was still nearly parallel to the ground as the vehicle was moving along its trajectory in excess of Mach 15. The crew experienced a swaying motion to the left and right (Y-axis) combined with a pull forward (X-axis) away from the seatback. The Z-axis accelerations pushed the crew members down into their seats. These motions might induce nausea, dizziness, and disorientation in crew members, but they were not incapacitating. The total acceleration experienced by the crew increased from approximately 0.8 G at LOC to slightly more than 3 G by the CE (figure 1.1-22).

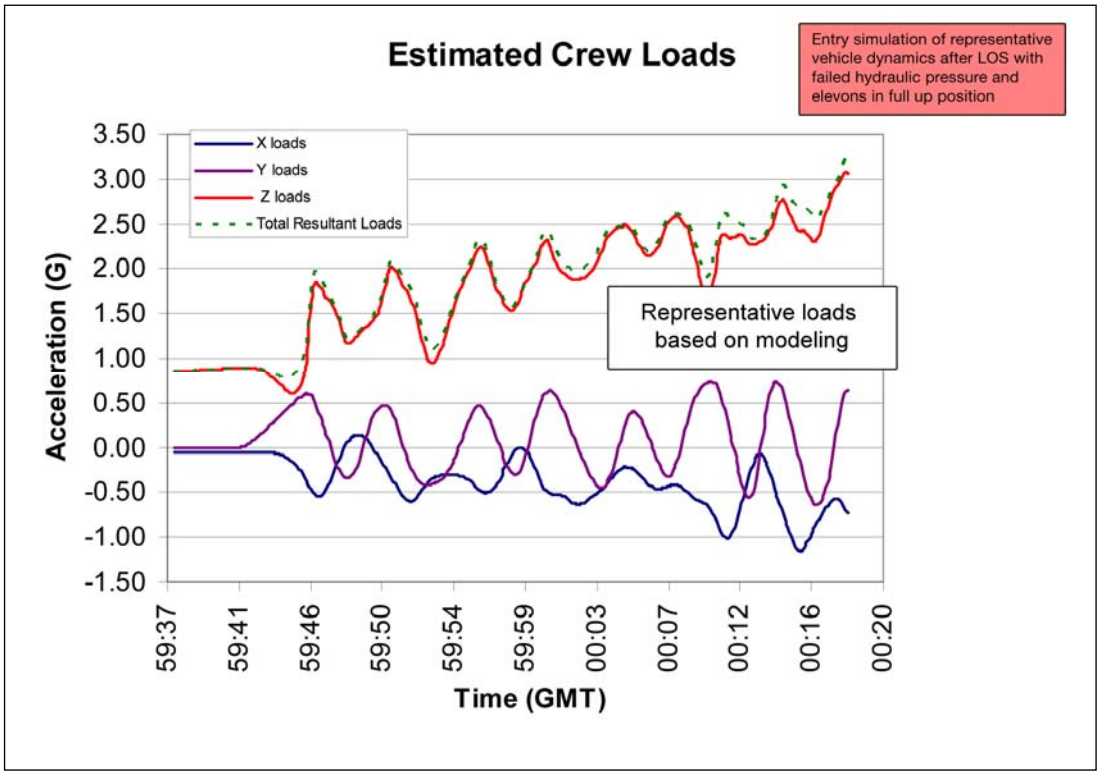


Figure 1.1-22. ***Resultant acceleration (G) vs. time prior to Catastrophic Event in the*** **crew** ***coordinate frame from the aerodynamic simulation model.***

The onset of this highly oscillatory flat spin likely resulted in the need for crew members to brace as they attempted to diagnose and correct the orbiter systems. As mentioned in the previous phase discussion, one middeck crew member had not completed seat ingress and strap-in at the beginning of this phase. Seat debris and medical analyses indicate that this crew member was not fully restrained before loss of consciousness. Only the shoulder and crotch straps appear to have been connected. The normal sequence for strap-in is to attach the lap belts to the crotch strap first, followed by the shoulder straps. Analysis of the seven recovered helmets indicated that this same crew member was the only one not wearing a helmet. Additionally, this crew member was tasked with post-deorbit burn duties. This suggests that this crew member was preparing to become seated and restrained when the LOC dynamics began. During a dynamic flight condition, the lap belts hanging down between the closely space seats would be difficult to grasp due to the motion of the orbiter, which may be why only the shoulder straps were connected.

At GMT 13:59:46 (EI+937), ground-based video indicates that a bright piece of debris was released followed by a second piece 2 seconds later. This second piece of debris[30] separated from the orbiter's trail and decelerated slowly, remaining visible for more than 37 seconds before dispersing into significantly fainter pieces. Ballistic analyses of ground debris indicate that pieces of the left OMS pod were being shed starting at about GMT 13:59:49 (EI+940).

RGPC-2 data show a message reset sometime between GMT 13:59:37.4 (EI+928) and GMT 14:00:05 (EI+956). This action is a nominal crew response to a fault message and requires a crew member to manually acknowledge the message by keyboard entry on the center panel. RGPC-2 data indicate that the RHC was moved beyond neutral sometime between GMT 14:00:01.7 (EI+952) and GMT 14:00:03.6 (EI+954), triggering a "DAP DOWNMODE RHC" message at GMT 14:00:03.637 (EI+954). This message, which is identical to the DAP DOWNMODE message that occurred at GMT 13:36:04 (EI-485) in the first phase,

[30]Identified as Debris D in the CAIB timeline.

was likely due to an RHC bump due to the oscillatory motion of the orbiter. At GMT 14:00:03.678 (EI+954), the orbiter autopilot was returned to the AUTO mode. Returning the DAP to AUTO requires either the CDR or the PLT to press a button located on the glare shield. These actions indicate that the CDR or the PLT was still mentally and physically capable of processing display information and executing commands, and that the orbiter dynamics were still within human performance limitations.

RGPC-2 data show normal crew module temperature and pressure through the end of the period of reconstructed data [GMT 14:00:04.826 (EI+956)].

The RGPC-2 data show normal Freon flow through the radiators on the inside of the payload bay doors (PLBDs). This indicates that the radiators and PLBDs also retained structural integrity up to this point. Based on structural analysis, it is likely that the PLBDs were compromised prior to CE, after the end of RGPC-2 data. Loss of the PLBDs reduced the structural strength of the orbiter midbody and allowed hot gas to impinge upon the sills in the payload bay.

The RGPC-2 data also indicate that while all three auxiliary power units (APUs)[31] were running, all three hydraulic systems had zero pressure and zero quantities in the reservoirs. With the loss of hydraulic pressures and the vehicle LOC, the crew likely assumed a generic problem with the APUs. A crew module panel was recovered with switch configurations indicating an attempt by the PLT to recover the hydraulic systems and hydraulic pressure by performing steps to initiate a restart of two of the three APUs. Switches for the same two of the three system hydraulic circulation pumps were also in the "On" position. While turning on the hydraulic circulation pump is not on the emergency checklist, it nonetheless can provide some limited hydraulic pressure and shows good systems knowledge by the crew members as they worked to attempt to restore orbiter control. These switch positions were not reflected in RPGC-2 data and, therefore, must have occurred after GMT 14:00:05 (EI+956).

Although the orbiter continued to shed debris, ground-based video from GMT 14:00:09 (EI+960) to GMT 14:00:18 (EI+969) shows a thin, relatively consistent trail, suggesting that the conditions remained steady for a short period of time (~9 seconds). Aerodynamic modeling indicates that this was a time of growing stresses on the orbiter and increasing Gs on the crew.

1.1.2.4 *Phase 4: Catastrophic Event to Crew Module Catastrophic Event*
[GMT 14:00:18 (EI+969) to GMT 14:00:53 (EI+1004)]
35 seconds in duration

This section discusses events affecting the crew from the CE at GMT 14:00:18 (EI+969) to the CMCE at GMT 14:00:53 (EI+1004) (figure 1.1-23). Separation of the forebody from the midbody and aftbody occurred at or just after CE. This phase lasted 35 seconds.

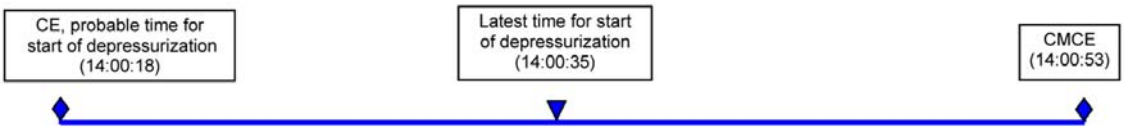


Figure 1.1-23. ***Phase 4 timeline with key events*.**

No telemetry or orbiter systems data are available during this phase. An on-board Global Positioning System (GPS) data recorder stopped at GMT 14:00:18.7 (EI+969) and the MADS/OEX recorder tape spool stopped at GMT 14:00:19 (EI+970) when the forebody lost power.[32] The sources of data that are available for reconstruction of events include ground-based video, recovered debris, medical findings, and modeling.

[31]The APUs drive the hydraulic pumps.
[32]The fuel cells, which provide all orbiter electrical power, are located in the midbody. Separation of the midbody from the forebody resulted in a total loss of power in the forebody.

Analysis of ground-based video identified specific events such as debris shedding and luminosity changes. A major luminosity event and orbiter trail characteristic change occurred at GMT 14:00:18.3 (EI+969) and is identified as the initiation of a major structural breakup called the CE. The aftbody and the forebody were identifiable as separate objects by GMT 14:00:25 (EI+976). Triangulating the video data provided relative motion, which was analyzed for an estimate of the deceleration and rotation rates of the forebody.

Columbia debris analysis consisted of five different methods: cluster analysis (plots of the ground location of recovered debris sorted by origination point on the orbiter), visual observations of debris, material analyses of melted deposits on select debris, and ballistic and thermal analysis of select debris. See Chapter 2.

Modeling was performed for various properties of the forebody. Aero thermal modeling provided estimated heat exposure. Aerodynamic modeling was used to evaluate the possible stable modes of the forebody. Aerodynamic modeling also provided estimates of G-loads, which increased as the forebody decelerated. The effects of the changing G-loads on both the forebody and the crew were analyzed. A combined environmental and structural analysis was performed to understand the effects of depressurization due to a single hole (or multiple holes equivalent to the same cross-sectional area) and the subsequent delta-pressure effects on the crew module structure.

At GMT 14:00:18 (EI+969), video showed that a significant event (the CE) occurred to the orbiter (figure 1.1-24). The GPS Miniaturized Airborne Global Receiver (MAGR) experiment, which was located in the middeck and powered by a fuel cell in the payload bay, experienced a loss of power at the CE. Less than 1 second later, the MADS/OEX recorder, which was also located in the crew module and similarly powered from the payload bay, also experienced a total power loss. The conclusion was that the forward and midbody orbiter segments separated at the CE. The CE is actually the start of a period of several seconds in which the orbiter underwent a major structural breakup. At GMT 14:00:25 (EI+976), there were visual indications that the orbiter was in multiple pieces. Ballistics analysis and structural debris analysis supports this period as the breakup event.

Figure 1.1-24. *The Catastrophic Event is depicted in these three frames of video that cover 0.1 second*. There is no change in the magnification/zoom factor. The third frame represents GMT 14:00:18.3 (EI+969).

The CAIB Report, based on data provided by the Crew Survival Working Group (CSWG), concluded that "Separation of the crew module/forward fuselage assembly from the rest of the orbiter likely occurred immediately in front of the payload bay (between X_o 576[33] and X_o 582 ring frame bulkheads)."[34] However, the SCSIIT's subsequent in-depth review of the debris field showed that significant portions of the X_o 582 ring frame bulkhead were found intermingled with the crew module debris field. Structural analysis led to the conclusion that the forebody separated from the midbody aft of the X_o 582 ring frame bulkhead. An aerodynamic simulation indicates that the structural operating load limits for the orbiter were not exceeded, indicating thermal degradation likely played a role in the failure.

The simulation also indicates that the crew module attachment fittings' (the x-links, y-links, and z-link) load limits were not exceeded. Although the exact sequence is not known, structural debris analysis suggests that the initial failure occurred immediately aft of the starboard x-link, where it attached to the payload bay

[33]The X_o terminology refers to the X position in inches in the orbiter coordinate frame, where the X-axis runs the length of the orbiter from fore to aft. See Section 2.1 for a visual graphic of the orbiter coordinate frame.
[34]*Columbia* Accident Investigation Board Report, Volume 1, August, 2003, p. 77.

sill, and was likely from a combination of thermal degradation and structural loads. The forebody probably separated from the midbody from starboard to port. Based on structural evidence and the debris field, the crew module remained with the forward fuselage indicating that the links attaching the two structures remained intact. The remaining orbiter structure separated into aft and midbody/right wing segments. See Chapter 2. Figure 1.1-25 shows the recovered x-links.

Figure 1.1-25. ***Port x-link, debris item no. 1678*** **(top),** ***and starboard x-link, debris item no. 1765*** **(bottom),** ***from the*** **Columbia** ***Reconstruction Database.***

The resulting jerk acting on the crew module attach fittings as the forebody separated from midbody structure caused motion of the crew module within the forward fuselage shell. The crew module pressure vessel impacted the forward fuselage, which apparently resulted in damage to the crew module pressure vessel, internal crew module structure, and forward fuselage structure. Debris evidence shows that internal damage occurred to some volumes and lockers on the middeck in close proximity to the pressure vessel shell. At least one crew module pressure vessel breach occurred in the lower equipment bay or middeck area (figure 1.1-26), probably at or near the time of the CE, but definitely not later than GMT 14:00:35 (EI+986) ±5 sec. This time is based on the ballistic analysis of a recovered mission patch confirmed to have come from inside the crew module from one of the volumes (Volume E) that suffered damage.

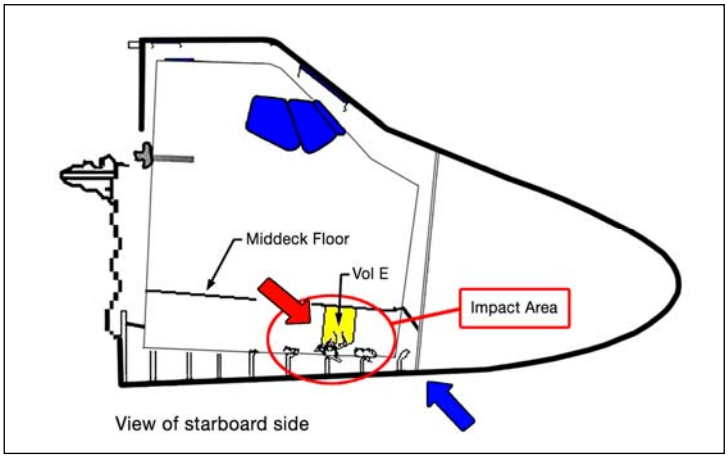


Figure 1.1-26. ***Scenario showing how the crew module pressure vessel could impact the forward fuselage, and the middeck Volume E could impact the crew module pressure vessel, with resultant damage.***

The start of crew cabin depressurization can be narrowed to a range of 17 seconds, from between GMT 14:00:18 (EI+969) to GMT 14:00:35 (EI+986) ±5 sec (see Section 2.3). Crew module debris items recovered west of the main crew module debris field were 8 in. in diameter or smaller, were not comprised of crew module primary structure, and originated from areas above and below the middeck floor. This indicates that the crew module depressurization was due to multiple breaches (above and below the floor), and that these breaches were initially small. Another crew module breach possibly occurred at the starboard x-link area, but no significant flight deck debris is seen west of the CMCE-related debris, suggesting that this breach occurred later rather than earlier in the timeline.

When the forebody separated from the midbody, the crew members experienced three dramatic changes in their environment:

1. all power was lost,
2. the motion and acceleration environment changed; and
3. crew cabin depressurization began within 0 to 17 seconds.

With the loss of power, all of the lights and displays went dark (although each astronaut already had individual chem-lights activated). The intercom system was no longer functional and the orbiter O_2 system was no longer available for use, although individual, crew worn Emergency Oxygen System (EOS) bottles were still available.

As the forebody broke free from the rest of the orbiter, its ballistic number underwent a sharp change from an average ballistic number of 41.7 pounds per square foot (psf) (out of control intact orbiter) to 122 psf (free-flying forebody). The aerodynamic drag of the forebody instantaneously decreased, resulting in a reduction in the translational deceleration from approximately 3.5 G to about 1 G (figure 1.1-27).

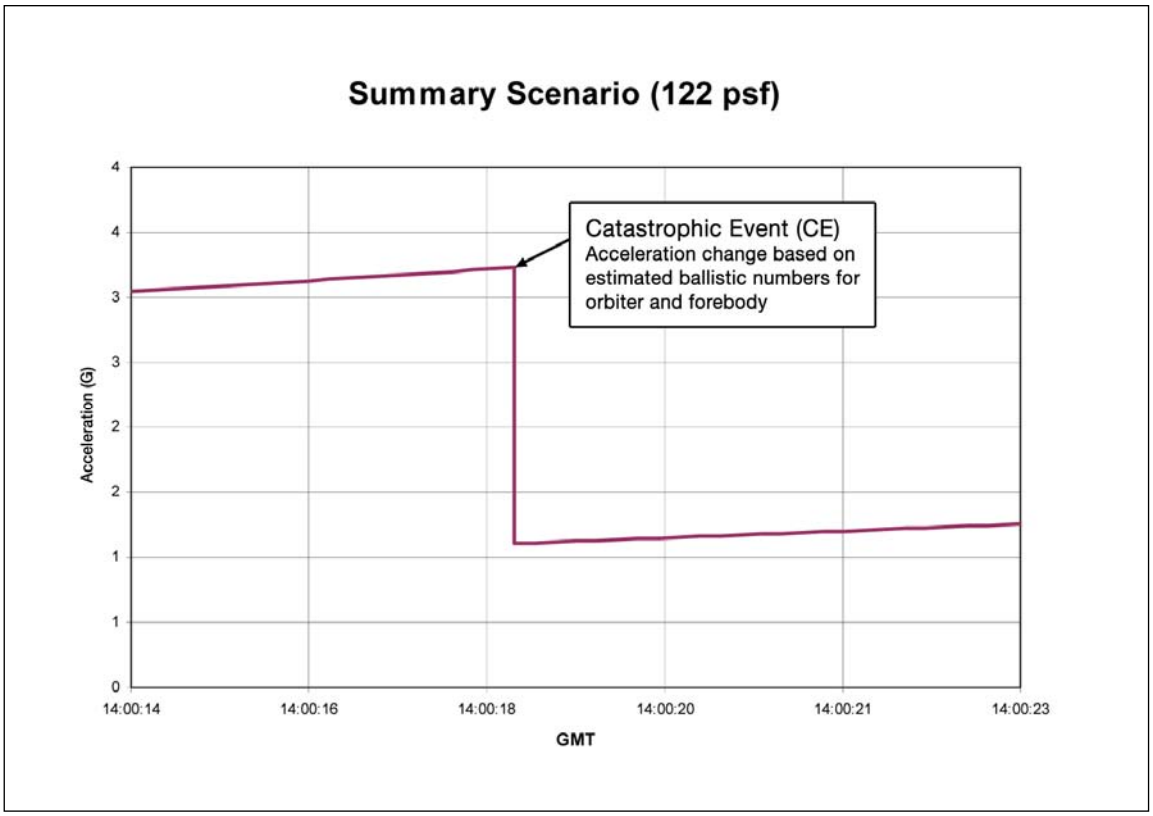


Figure 1.1-27. ***Estimated change in total G experienced by the forebody due to a change in ballistic number from the orbiter breakup.***

The asymmetrical starboard to port separation of the forebody from the midbody would have induced rotation in the forebody, introducing new angular accelerations. The angular accelerations acting on the forebody at this time are impossible to accurately characterize due to inadequate data, but they likely changed significantly due to the abrupt change in the center of axis of rotation when the center of gravity (c.g.) changed from X_o 1075.5 to X_o 470.8 (figure 1.1-28).

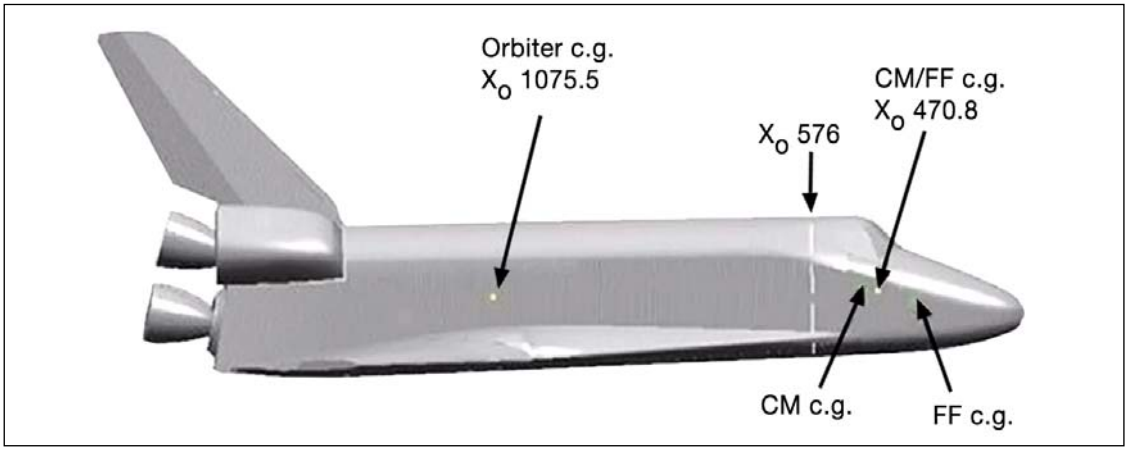


Figure 1.1-28. ***X-axis center-of-gravity locations for the intact orbiter, the crew module, the forward fuselage, and the forebody (crew module plus forward fuselage).*** **The X_o 576 is the aft bulkhead of the crew module.**

Also, as the forebody broke from the vehicle, the crew module moved within the forward fuselage shell resulting in transient rates of change (described earlier). Although it is probable that momentary sharp changes in acceleration caused high instantaneous G-loads, medical evidence indicates that the crew cabin pressure and load environment at the CE were still within human limitations for survival.

Effects of cabin depressurization on the crew would depend on the rate of depressurization. Existing CEE is capable of protecting the crew from rapid decompression via pressure suit, helmet, and either the orbiter O_2 or an individual EOS for a limited time. However, recovered crew equipment shows crew visors were in the nominal (up) position rather than emergency configuration (down and locked). Inspection of the wrist and glove rings showed that the glove wrist rings were not attached to the suit for two crew members on the middeck and one crew member on the flight deck, and one crew member had not yet donned the helmet. The change (from the crew's vantage point) from a nominal entry profile to the LOC and subsequent separation of the forebody from the orbiter all occurred in approximately 40 seconds. Experience shows that this is not sufficient time to don gloves and helmets.

Histological (tissue) examination of all crew member remains showed the effects of depressurization. Neither the effects of CE nor the accelerations immediately post-CE would preclude the crew members who were wearing helmets from closing and locking their visors at the first indication of a cabin depressurization. This action can be accomplished in seconds. This strongly suggests that the depressurization rate was rapid enough to be nearly immediately incapacitating. The exact rate of cabin depressurization could not be determined, but based on video evidence complete loss of pressure was reached no later than (NLT) GMT 14:00:59 (EI+1010), and was likely much earlier. The medical findings show that the crew could not have regained consciousness after this event. Additionally, respiration ceased after the depressurization, but circulatory functions could still have existed for a short period of time for at least some crew members.

The first event with lethal potential was depressurization of the crew module, which started at or shortly after orbiter breakup. Existing crew equipment protects for this type of lethal event, but inadequate time existed to configure the equipment for the environment encountered.

After the CE, the forebody was exposed to a high thermal environment as it decelerated and descended into an increasingly dense atmosphere. TPS tile and blankets on the forward fuselage protected the crew module,

but the aft bulkhead was unprotected. However, debris field analysis indicates that the aft bulkhead remained intact until crew module breakup. The volume between the forward fuselage and the crew module had openings to the environment, which could result in the entry of heated gas. Crew module breaches could allow the entry of this gas into the crew module after the dynamic pressure outside the crew module exceeded atmospheric pressure inside the crew module. A few molten globules of metal were found on recovered seat harness straps, indicating the presence of heated metal inside the crew module while the unconscious or deceased crew members were still restrained in their seats. Although the timing of the deposition cannot be determined precisely, it may have been very close to the crew module breakup. There is no evidence to suggest that the overall crew module internal structure temperature was severe, but local hot spots may have existed near breaches.

The orbiter had substantial rates of rotation in all axes of rotation when RGPC-2 data ended at GMT 14:00:04 (EI+955). The orbiter breakup at the CE imparted motion to the forebody, and the forebody began rotating after it broke free from the vehicle. Aerodynamic modeling indicates that the free-flying forebody would not achieve a stable attitude. Videos of the forebody show brightening and dimming, implying rotational motion. Triangulation analysis of the forebody in video showed a slow wobble motion in all three axes, also supporting rotation or tumbling. Thermal damage seen on external portions of the forebody indicates intermittent exposure to heat. Based on the wobble motion, rotation rates gradually increased with an estimated initial average rate of 0.1 revolution per second (36 degrees per second) around a changing body axis (see Section 2.1). This rate is not extreme, and even peak rates toward the end of this phase result in angular accelerations of less than 2 G. Translational deceleration due to aerodynamic drag also increased, up to approximately 3 G at the CMCE. The loss and redistribution of mass as forward fuselage structure failed and separated would affect the rotation rate. Modeling suggests that the rate could continue to build, up to 0.5 revolution per second, although this was not verified with video data. The crew members seated farthest from the crew module c.g. experienced the highest angular accelerations due to the greater distance (moment arm) from the center of rotation. This acceleration was in addition to the translational accelerations and, depending on the attitude of the rotating forebody, the accelerations experienced by the crew members could vary from about –1 G to +5 G.

Under rapidly changing accelerations, the design intention is that inertial reels on the seat restraint shoulder harnesses will lock, and remain locked, until manually disengaged by the crew member. To lock, the inertial reel mechanism used on orbiter seats requires 1.78 G to 2 G of strap acceleration, in a direction orthogonal to the mechanism (straight out of the seatback). The abrupt dynamics associated with the CE would be expected to have locked the inertial reel. In the subsequent multi-axial rotating environment experienced during this phase, it is expected that the unconscious or deceased crew periodically would arrive at a posture allowing harness retraction. The harness would then remain retracted if the inertial reels had locked. However, seat analysis shows that several of the shoulder harness restraints failed with the inertial reel straps partly or fully extended, and other inertial reel straps were extended at some point during this phase (see Section 3.1). Either the acceleration on the straps was insufficient to lock the harness, the loading was not orthogonal (preventing harness retraction), ACES equipment blocked the strap retraction slot, or some combination of all three effects occurred. The net effect was that the crew members had no upper body restraint and were restrained solely by their lap belts.

The combination of the lack of upper body restraint and a helmet that, by design, does not internally conform to the head while exposed to cyclical motion resulted in lethal mechanical injuries for some of the unconscious or deceased crew members. The circulatory system of most of the unconscious or deceased crew was still functioning at the time of these lethal injuries. If the harnesses had been locked or the crew had been conscious and able to brace, the injuries likely would not have been lethal.

The second event with lethal potential was unconscious or deceased crew members exposed to a dynamic rotating load environment with nonconformal helmets and a lack of upper body restraint.

Existing seat and helmet design did not protect the crew from this lethal event.

Crew module structure temperature increased during this phase, resulting in a corresponding reduction in structural strength. The increase in loads due to the increasing deceleration and increasing rotational rate and thermal degradation resulted in eventual structural failure.

In summary, in the 35 seconds from the CE to the CMCE, the forebody detached, the crew module breached and depressurized, and the forebody experienced increased heating and began to structurally degrade.

Figure 1.1-29 shows an overall summary of accelerations, heat rates, and trajectory. *Information for post-CMCE (GMT 14:00:53 (EI+1004) and beyond) is predicted data for an **intact crew module** and **is not representative** of actual trajectory, accelerations, and thermal environment for the crew or for individual components of the crew module.*

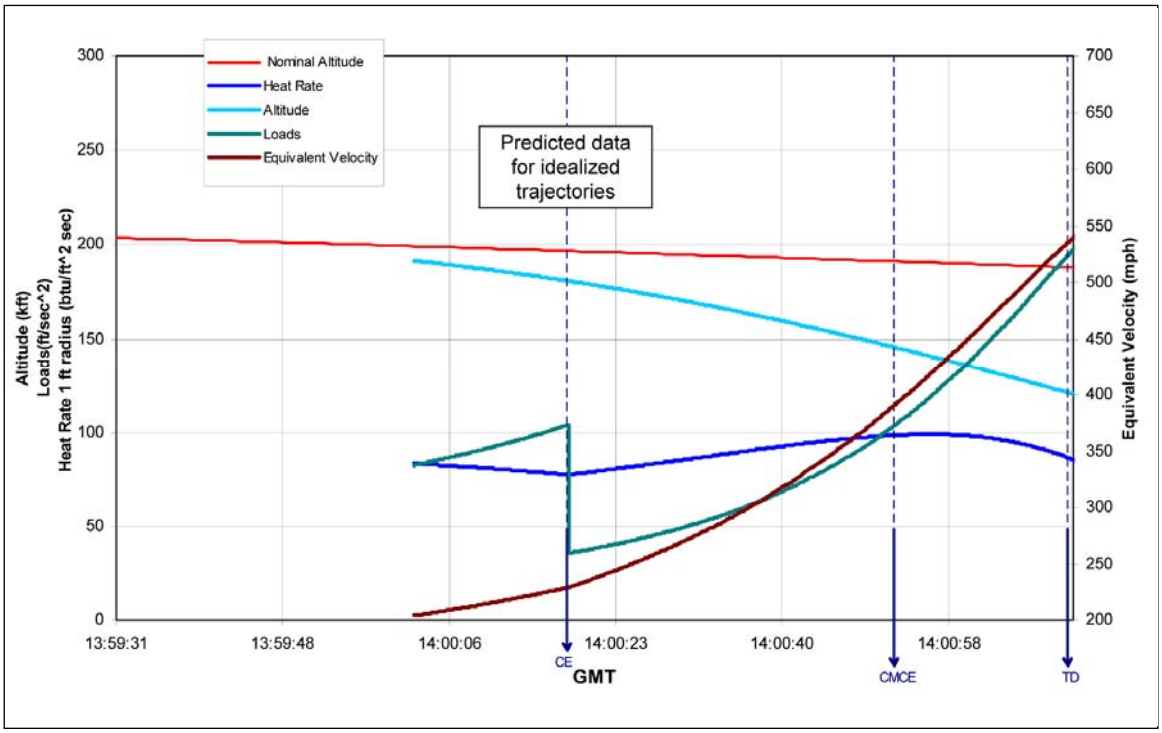


Figure 1.1-29. ***Predicted acceleration and thermal data for an idealized trajectory of the intact orbiter with a ballistic number of 42 pounds per square foot, and post-Catastrophic Event free-flying forebody with a ballistic number of 122 pounds per square foot.***

1.1.2.5 *Phase 5: Crew Module Catastrophic Event to Total Dispersal*
[GMT 14:00:53 (EI+1004) to GMT 14:01:10 (EI+1021)]
17 seconds in duration

This section discusses events affecting the crew from the CMCE at GMT 14:00:53 (EI+1004) to TD at GMT 14:01:10 (EI+1021). This phase lasted 17 seconds.

Three sources of data were available for analysis of Phase 5: video, debris, and medical findings.

Video data show that a significant forebody brightening event began at GMT 14:00:53 (EI+1004). By GMT 14:01:10 (EI+1021), the forward fuselage, crew module, and trailing debris clearly originating from the forebody are no longer visible in the video. In all videos, the last of this debris disappears from sight while in the middle of the field of view (FOV) rather than leaving the FOV, indicating either speed and/or size decreased such that its brightness, which was created by frictional heating from drag, was below the sensitivity of the video camera.

Debris analysis consisted of five techniques: cluster analysis, visual observations, materials analyses, ballistics, and thermal modeling predictions (see Section 2.4). Materials analysis was performed on select items to evaluate thermal exposure (see Sections 2.1 and 3.2). Materials evaluations were performed on the seats, some crew equipment, and some forebody structure. Additionally, some thermal modeling was done to estimate peak thermal temperatures after separation for various items. The model results were then compared to the actual debris appearance.

Based on video and ballistic evidence, at GMT 14:00:53 (EI+1004) a significant breakup event began. This event is designated as the CMCE.

The CMCE started with the separation of the forward fuselage from the crew module, exposing the entire crew module to the thermal effects of entry. The main forebody debris field included all recovered crew module pressure vessel structure, almost 90 percent of recovered forward fuselage structure, and around 90 percent of the crew module contents. This indicates that the failures of the forward fuselage and crew module were closely associated. Ballistic analysis confirmed this assessment.

The video recorded from an Apache helicopter operating in the area of Ft. Hood, Texas shows a significant event of two objects with similar luminosity and ballistic number separating simultaneously from the forebody (figure 1.1-30). The remaining central object maintained integrity for several more seconds in the video. Shortly after these items peeled away, the remaining object began to lose large pieces of structure.

The conclusion was drawn that these two objects were most likely the upper and lower forward fuselage sections, leaving the crew module (the central object) intact but no longer protected. Within seconds, the crew module began to lose structural integrity as well.

Forward fuselage debris shows localized thermal damage and very little evidence of debris-debris interaction. Large portions of structure were recovered intact. Material deposition on the interior of the forward fuselage debris was not significant. Reconstruction of the forward fuselage debris supports a structural failure from starboard-to-port and forward-to-aft.

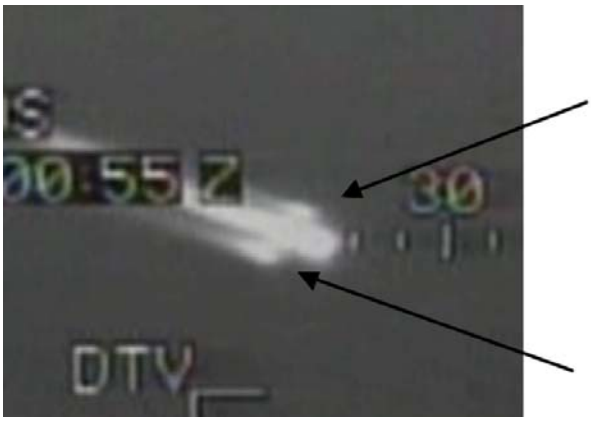


Figure 1.1-30. ***Video frame from the Apache video at GMT 14:00:55 (EI+1006) showing two similar luminosity objects separating from the forebody.***

The crew module breakup was rapid (<15 seconds). The range of ballistic numbers of the debris items resulted in quickly diverging individual trajectories such that very little debris-debris interaction occurred. Cluster analysis of the debris field shows that the crew module forward X_{cm}[35] 200 bulkhead debris is farther west than the crew module aft X_o 576 bulkhead debris. This indicates that the failure of the forward bulkhead happened prior to the failure of the aft bulkhead. The middeck floor debris field begins at the same longitude as the forward bulkhead, suggesting that the failures were nearly simultaneous. Debris evidence from the crew module structure suggests a starboard-to-port breakup of the middeck area, which probably included the forward bulkhead. Cluster analysis and evidence of significant heating of the flight deck floor and the flight deck seats indicates that the flight deck was intact for a short period of time (probably less than 5 seconds) after separation from the middeck. Cluster analysis indicates that the airlock stayed with the flight deck, possibly connected by the aft bulkhead.

[35]The X_{cm} terminology refers to the X position in inches in the crew module coordinate frame, where the X-axis runs the length of the crew module from fore to aft. The crew module coordinate frame axes are coincident with the orbiter frame axes, but with a different X-axis origin.

There is no evidence of an explosion or a fire. Analysis of thermal vectors on numerous debris items showed multiple independent heat vectors across the structure. For example, many recovered middeck floor panels were nearly pristine with paint still visible, while floor panels from immediately adjacent locations had melted materials deposited on them and other signs of high thermal exposure (figure 1.1-31). After breakup, individual items experienced their own trajectories and heat exposure. This heat exposure can vary enormously with ballistic number and other effects such as shadowing from other debris items and orientation of the item into the heat vector. The lack of consistent directional heating vectors on crew module debris suggests heating was due to individual item trajectories and random exposure during breakup rather than a major breach resulting in directional heating.

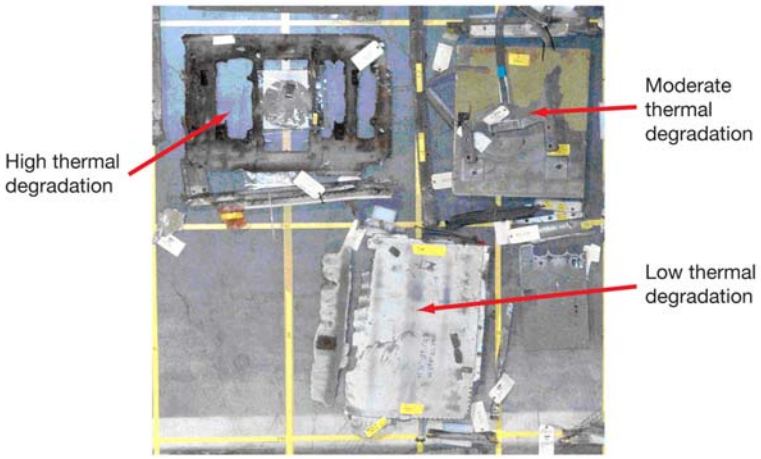


Figure 1.1-31. ***Middeck floor debris in original relative orientation showing varying thermal exposure*.**

The exact time and sequence that the crew and seats separated from the crew module is unknown. A comprehensive evaluation of ballistic analysis of debris, crew member remains, and crew worn equipment indicates that the middeck crew remains were separated from the crew module prior to the flight deck crew remains, supporting the conclusion that the flight deck stayed intact a few seconds longer than the middeck.

The dynamic pressure environment exposure caused the mechanical failure of the crew suits (common to high-speed accidents, but somewhat unexpected given the aerodynamic pressure of only 450 to 550 psf). The suit is designed to maintain structural integrity when exposed to a windblast that is up to 560 knots equivalent air speed (KEAS) (806 psf). This assumes that the helmet visor is down. The helmet visors being in the up position is the most likely explanation for the hastened disruption of the suits. Although suit disruption was primarily due to aerodynamic (mechanical) loads, the thermal environment and atomic oxygen in the atmosphere may have been a contributing factor.

The third event with lethal potential was separation from the crew module and the seats with associated forces, material interactions, and thermal consequences. This event is the least understood due to limitations in current knowledge of mechanisms at this Mach number and altitude. Seat restraints played a role in the lethality of this event. Although the seat restraints played a significant role in the lethal-level mechanical injuries, there is currently no full range of equipment to protect for this event. This event was not survivable by any means currently known to the investigative team. All circulatory functions had ceased by the end of this phase.

Whether an item separated from the crew module or the crew module lost significant mass, an instantaneous change in ballistic number occurred and resulted in varying deceleration and thermal profiles. The accelerations varied from over 30 G for a short duration (less than 5 seconds) to over 10 G for up to 20 seconds. The range of ballistic numbers of debris generated a range of thermal conditions. Objects with higher ballistic numbers take longer to decelerate, and experience longer periods of heating and lower G-spikes (see Section 2.1). Video shows a large deceleration of the crew module relative to the main engines at GMT 14:01:08 (EI+1019). The visual object actually represents a cloud of multiple objects that experienced deceleration at varying rates. Several seconds elapsed before objects of varying ballistic numbers separated visually from each other, creating the impression of a solid object in the video for a few seconds. This is consistent with a gradually expanding breakup caused by items having a wide range of ballistic numbers and deceleration trajectories, resulting in a widely spread debris cloud.

TD was complete by GMT 14:01:10 (EI+1021).

1.1.2.6 *Phase 6: Total Dispersal to ground impact*
[GMT 14:01:10 (EI+1021) to approximately 14:35:00 (EI+3051)]
approximately 34 minutes in duration

This section discusses events from TD (GMT 14:01:10 (EI+1021)) to the ground impact of debris. Heavy items impacted the ground much sooner than lighter objects but traveled much farther from the point of separation. It was calculated that a small fragment of cloth with a ballistic number of .5 psf would impact the ground around 33 minutes after separation. Based on this calculation, all crew module debris was likely on the ground (including very lightweight objects) by GMT 14:35:00 (EI+3051).

Three sources of data were available for Phase 6: video, debris analysis, and medical findings.

Very limited video data were available as the crew module rapidly disappeared from the FOV as it dispersed into smaller and smaller debris. The smaller size and loss of heating as the debris decelerated reduced the ability to detect items in the video. Debris analysis consisted of four techniques: visual observations, ballistics analysis, cluster analysis, and materials analysis (see Chapter 4).

The forebody breakup event occurred between 145,000 feet and 105,000 feet at an ambient pressure of approximately 0.03 pound per square inch (psi). After the deceleration peak, the overall deceleration would stabilize to 1 G at terminal velocity. At ground level, the ambient absolute pressure condition was approximately 14.7 psi and the temperature was 59°F (15°C).

The fourth event with lethal potential was exposure to near vacuum, aerodynamic accelerations, and cold temperatures. Current crew survival equipment is not certified to protect the crew above 100,000 feet, although it may potentially be capable of protecting the crew.

At the altitude the deceased crew departed from the crew module, the environmental risks include lack of O_2, low air pressure, high thermal exposure as a result of deceleration from high Mach numbers, and exposure to cold temperatures. Existing shuttle CEE is certified to protect a crew member exposed to an atmospheric/altitude environment up to 100,000 feet. Anecdotal evidence from the survival of the pilot of an SR-71 mishap[36] suggests that an intact, pressurized suit similar to the ACES can also protect a crew member at least up to speeds of Mach 3.

Shuttle crew members carry a personal O_2 supply that provides O_2 independent of the orbiter supply. This system can provide enough O_2 for a crew member to reach the ground from altitudes much greater than 100,000 feet, so it is not the limiting factor in the system.

The ground impact without parachute protection generated a very large instantaneous G event.

The final event with lethal potential was ground impact. Existing shuttle CEE protects for ground impact with a parachute. However, the crew must manually initiate the parachute opening sequence, or the parachute must be used in conjunction with the crew escape pole of the shuttle to initiate the parachute automatic opening sequence.

[36]Department of the Air Force, SR-71 Aircraft Mishap Report, January 25, 1966.

1.2 Master Timeline

The SCSIIT Master Timeline was developed as a tool to aid the SCSIIT with the investigation of what happened to the crew of STS-107. The SCSIIT Master Timeline began with the tailoring of the CAIB Master Timeline, Revision 15 to highlight crew-related events. Additional events were added from various sources, including:

- recorded telemetry
- RGPC data
- MADS/OEX recorder data
- MAGR data
- recovered on-board videos
- ground-based videos
- air-to-ground audio
- forensic analysis of medical findings
- engineering forensic analysis of vehicle and CEE debris
- ballistic analysis of vehicle and CEE debris

The timeline is divided into six phases:

- Phase 1: From the deorbit preparation checklist timeline initiation to EI. The deorbit preparation checklist timeline begins 4 hours prior to the deorbit burn. EI is defined as the time at which an altitude of 400,000 feet was reached. [GMT 09:15:30 (EI–16119 seconds) – GMT 13:44:09 (EI)]
- Phase 2: From EI to LOS. LOS is the time of the loss of voice and real-time data from *Columbia*. [GMT 13:44:09 (EI) – GMT 13:59:32 (EI+923)]
- Phase 3: From LOS to the CE. The CE is defined as the initiation of the orbiter breakup into the primary subcomponents of the forebody, midbody and aftbody. The CAIB timeline ends with the CE. [GMT 13:59:32 (EI+923) – GMT 14:00:18 (EI+969)]
- Phase 4: From the CE to the CMCE. The CMCE is defined as the initiation of the forebody breakup. [GMT 14:00:18 (EI+969) – GMT 14:00:53 (EI+1004)]
- Phase 5: From the CMCE to TD. TD is defined as the time at which the crew module was broken down into its subcomponents. [GMT 14:00:53 (EI+1004) – GMT 14:01:10 (EI+1021)]
- Phase 6: From the TD to ground impact of the crew and the bulk of the crew module debris. [GMT 14:01:10 (EI+1021) – approximately GMT 14:35:00 (EI+3051)]

All events are presented in GMT. In addition, events prior to the TIG of the deorbit burn also include the time prior to TIG. After TIG, the events include the time from TIG and the time to EI. After EI is reached, all events are presented in just GMT and EI.

Five symbols are used in the timeline to aid the reader in scanning for events of a certain category. The symbols are:

– indicates a crew-related event.
– indicates an event that is based/observed on video footage.
– indicates a vehicle-related event.
– indicates a vehicle-related event that occurred at the time of the separation of the forebody (crew module and forward fuselage) from the midbody.
– indicates a vehicle-related event that occurred after the separation of the forebody from the midbody.

The timeline does not include every event. For a comprehensive listing of events, the reader should consult sources such as:

- *Columbia* Accident Investigation Board Master Timeline, Revision 15
- *Columbia* Accident Investigation Board Report, Volume II, Appendix D.19, Qualification and Interpretation of Sensor Data from STS-107
- OEX Data Evaluation of End of Mission Data for STS-107, Vehicle Data Mapping Team, presentation, 8-22-2003, Rev B
- STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, 3-10-2003

1.2.1 Phase 1: Deorbit preparation to entry interface
GMT 09:15:30 (EI–16119) through GMT 13:44:09 (EI)

TIME		EVENT
09:15:30 (TIG[1]–04:00:00[1])		**Deorbit Preparation.** The crew begins working items on the deorbit preparation (D/O PREP) checklist at TIG–4 hours per pre-mission planning.
11:11:18 (TIG–02:04:12)		**OPS 301.** The crew, per nominal procedures, manually enters the OPS 301 command to initiate the Pre-Deorbit Coast Major Mode software. This is the first entry-phase software sequence in preparation for entry. [*Telemetry, Tracking, and Data Relay Satellite-West (TDRS-W) data*]
~11:40:00 (TIG–01:35:30)		**Recovered Middeck Video Begins.** Approximately 30 minutes of video (without audio) were recovered from a camera in the middeck. On the video, the crew is shown working through D/O PREP checklist items. Times are approximate due to the lack of audio and visual cues to synchronize activities seen on the video with GMT (figure 1.2-1).

[1]TIG is time of ignition and refers to the start time of the deorbit burn. TIG–hh:mm:ss is the time before the burn begins in hours (hh), minutes (mm), and seconds (ss). TIG+hh:mm:ss is the time after the burn begins.

TIME **EVENT**

Figure 1.2-1. ***Start of the recovered middeck video*****.**

The following was observed at the start of the video (figure 1.2-2):

- All middeck seats (seat 5, seat 6, and seat 7) are installed on the middeck floor.
- Personal parachute assemblies for MS3/Seat 5, MS1/Seat 6, and Payload Specialist 1 (PS1)/Seat 7 are positioned on seatbacks.
- MS3/Seat 5 has already donned the ACES (excluding gloves and helmet) and parachute harness.
- MS1/Seat 6 and PS1/Seat 7 have yet to don the ACESs.
- The escape pole is still stowed on the middeck ceiling (on-orbit location).

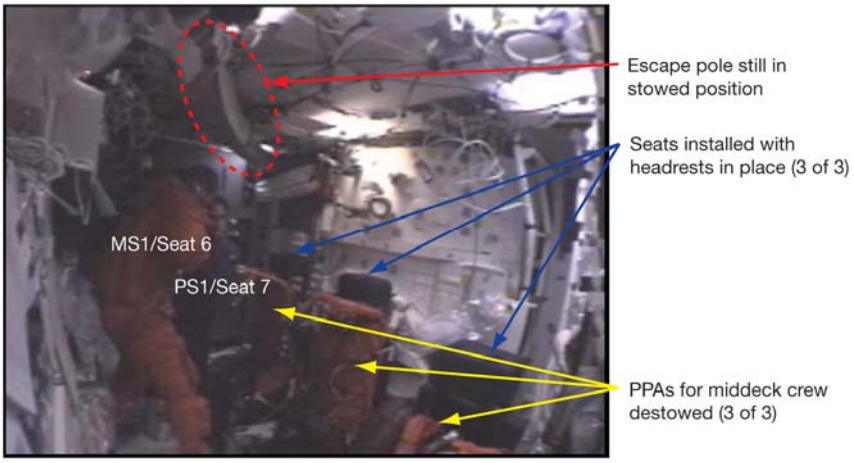


Figure 1.2-2. ***Middeck configuration during duration of the recovered video*****.**

TIME		EVENT
~11:48:56 (TIG–01:26:34)		CDR/Seat 1 dons ACES (excluding gloves and helmet) and parachute harness and goes to flight deck.
		MS3/Seat 5 and PS1/Seat 7 are observed fluid-loading in preparation for return to 1 G.
		MS1/Seat 6 dons ACES (excluding gloves and helmet) and parachute harness.
		PS1/Seat 7 and MS1/Seat 6 pass helmets in helmet bags to flight deck crew members.
12:10:00 (TIG–01:05:30)		**Recovered Middeck Video Ends.** The video ends with PS1/Seat 7, MS3/Seat 5, and MS1/Seat 6 beginning to remove the escape pole from the ceiling to install it. The D/O PREP checklist calls for the pole to be installed as one of many activities in the Entry Cabin Configuration block, which starts at TIG–03:50:00. NOTE: Evaluation of the debris reveals that the pole was installed in the launch/entry position, so the crew completed the installation (figure 1.2-3).

Figure 1.2-3. *End of the recovered middeck video*. [Mission Specialist 3/Seat 5 in foreground]

TIME		EVENT
12:53:04 (TIG–00:22:26)		**OPS 302.** Per nominal procedures, the crew manually enters the OPS 302 command to initiate the *Deorbit Execute* Major Mode software. [*Telemetry, TDRS-W data*]
13:10:00 (TIG–00:05:00)		The MCC gives the "GO" for deorbit burn.[2]
13:10:30 (TIG–00:05:00)		**TIG–5**. TIG refers to the time of the planned ignition of the OMS engines (referred to as the deorbit burn) to reduce the orbiter's velocity enough to result in entry into the atmosphere. This is a benchmark time to make sure the crew starts the APU in time for the deorbit burn and is also the time at which the last MS is to be seated. [*Based on TIG event in Master Timeline, Rev. 15 Baseline*]
13:10:39 (TIG–00:04:51)		**APU 2 Start – Low Press.** Three APUs provide pressure to the orbiter hydraulic systems (engine gimbals, elevons, and body flap). Only one APU is used to support the deorbit burn. [*Master Timeline, Rev. 15 Baseline*]
13:15:30 (TIG–00:00:00)		**TIG: Deorbit Burn Begins.** This is the beginning of the deorbit burn using the OMS engines. [*Master Timeline, Rev. 15 Baseline*]
13:18:08 (EI–1561)		**Deorbit Burn Ends.** This is the end of the deorbit burn. [*Master Timeline, Rev. 15 Baseline*]
13:20:21 (EI–1428)		**OPS 303.** Per nominal procedures, the crew manually enters the OPS 303 command to initiate the *Pre-entry Monitor* Major Mode software.
13:26:09 (EI–1080)		**Forward RCS Dump Start.** This is a nominal operation to deplete the forward RCS fuel and oxidizer tanks in preparation for entry. [*Master Timeline, Rev. 15 Baseline*]
13:27:12 (EI–1017)		**Forward RCS Dump Complete.** [*Master Timeline, Rev. 15 Baseline*]
13:31:25 (EI–764)		**APU 1 Start.** APU 1 is started per the deorbit procedures. [*Master Timeline, Rev. 15 Baseline*]
13:31:29 (EI–760)		**APU 3 Start.** APU 3 is started per the deorbit procedures. [*Master Timeline, Rev. 15 Baseline*]
13:31:57 (EI–732)		APU 1 is performing nominally. [*Master Timeline, Rev. 15 Baseline*]

[2]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 38.

TIME		EVENT
13:31:59 (EI–730)		APU 2 is performing nominally. [*Master Timeline, Rev. 15 Baseline*]
13:32:01 (EI–728)		APU 3 is performing nominally. [*Master Timeline, Rev. 15 Baseline*]
13:32:11 (EI–718)		PLT/Seat 2: "Houston, here comes SSME HYD repress." This is part of the procedure to use the hydraulic system to move the space shuttle main engines (SSMEs) to the desired position for entry and landing. [*SCSIIT air-to-ground 1 (A/G1) Tape Elapsed Time (TET)*[3] *02:20:26*]
13:32:18 (EI–711)		CAPCOM: "And we're ready, Willie. No deltas." This message informs the crew that there were no changes to the planned procedure. [*SCSIIT A/G1 TET 02:20:33*]
13:32:22 (EI–707)		PLT/Seat 2: "Copy, no deltas." [*SCSIIT A/G1 TET 02:20:37*]
13:35:16 (EI–533)		CAPCOM: "And *Columbia*, Houston. The HYD fluid thermal conditioning will not be required today. We'll meet you on the cards." [*SCSIIT A/G1 TET 02:23:31*]
13:35:26 (EI–523)		CDR/Seat 1: "And we copy, Houston. HYD fluid thermal conditioning not required, and we copy going to the cards." [*SCSIIT A/G1 TET 02:23:41*]
13:35:32 (EI–517)		CAPCOM: "And, Rick, don't want to lead you astray, and don't forget about the stuff on page 3-44." [*SCSIIT A/G1 TET 02:23:47*] Page 3-44 is part of the entry checklist; it contains the steps for enabling the RHCs by turning on the flight controller power and activating the entry video camera system. The last step has the crew go to the Entry Maneuvers cue card.
13:35:34 (EI–515)		**Recovered Flight Deck Video Begins.** The first audio on the tape is of the CAPCOM completing the sentence recorded on the A/G audio at approximately GMT 13:35:32 (EI-517): "…and don't forget about the stuff on page 3-44." [*SCSIIT A/G1 TET 02:23:51*] The PLT/Seat 2 is shown adjusting g-suit setting (figure 1.2-4).

[3]TET is the tape elapsed time from the start of the audio recording file.

TIME	EVENT

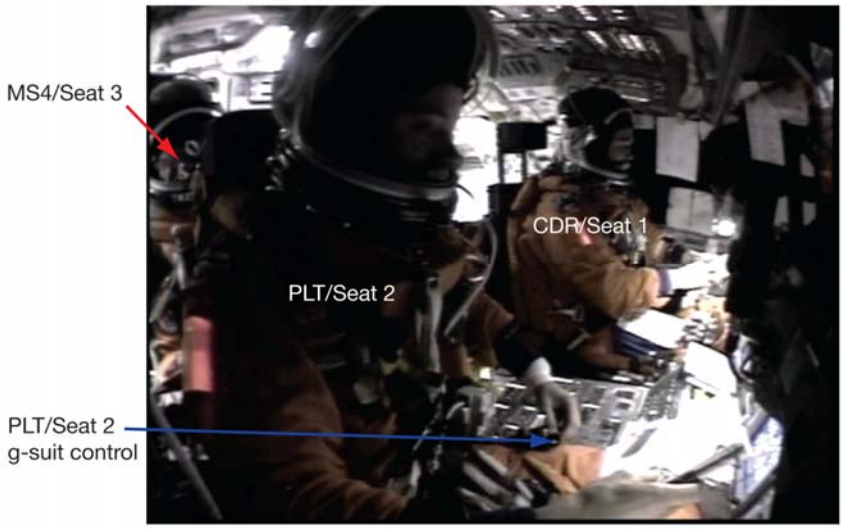


Figure 1.2-4. *Pilot/Seat 2 adjusting g-suit setting*.

All visible crew members (CDR/Seat 1, PLT/Seat 2, and MS4/Seat 3) are fully suited except for gloves and are strapped in. All helmet visors are OPEN per nominal procedure.

CDR/Seat 1: "Right, we're checking that. We've got the flight controller power on. We're working through the rest of it as well. Thanks." [*SCSIIT A/G1 TET 02:23:57*]

CAPCOM: "Sounds good."

13:36:02 (EI–487) MS4/Seat 3 is shown starting to don gloves (figure 1.2-5).

Figure 1.2-5. *Mission Specialist 4/Seat 3 donning glove*.
[Glove is within dotted circle]

TIME	EVENT
13:36:04 (EI–485)	Between GMT 13:36:04 (EI-485) and GMT 13:36:06 (EI-483), the CDR/Seat 1 is performing entry preparation actions that lead to the RHC being bumped (figure 1.2-6).

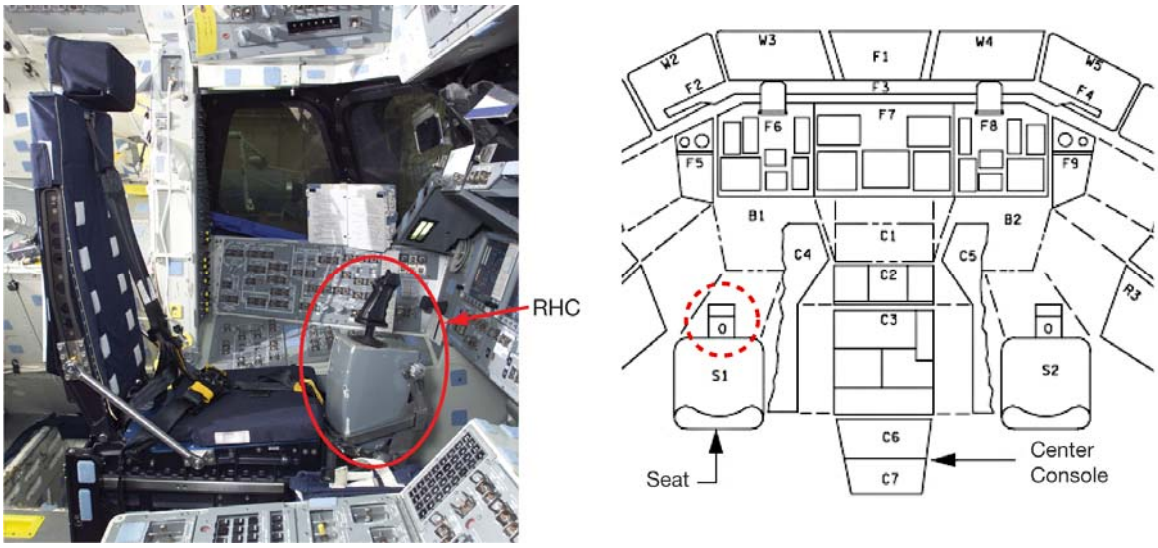


Figure 1.2-6. ***Location of Commander's seat and rotational hand controller*****. [Left picture from the Shuttle Training Simulator looking from starboard to port; right picture adapted from the Space Shuttle Systems Handbook looking from aft to forward]**

13:36:07 (EI–482)

DAP DNMODE RHC. Primary Avionics Software System (PASS) DAP DOWNMODE RHC (time from telemetry analysis). The message indicates that the RHC movement in the previous event was sufficient to mode the DAP out of AUTO into Inertial mode. When this occurs, a "DAP DOWNMODE RHC" caution and warning message is displayed, the INRTL button on the C3 panel is illuminated (see arrow in figure 1.2-7), and a tone, which can be heard in the recovered video tape, is annunciated (figure 1.2-7).

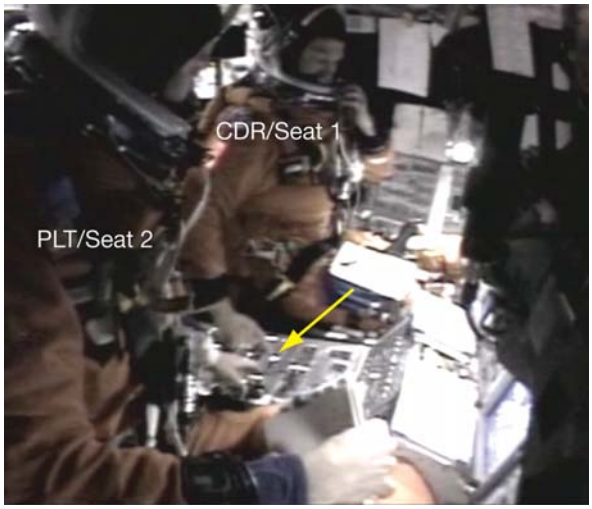


Figure 1.2-7. ***Digital autopilot button on the C3 panel is illuminated*****.**

TIME	EVENT
13:36:14 (EI–475)	CDR/Seat 1 responds to the DAP DOWNMODE RHC message by pressing the illuminated AUTO button on the C3 panel to restore the DAP to AUTO (figures 1.2-8 and 1.2-9).

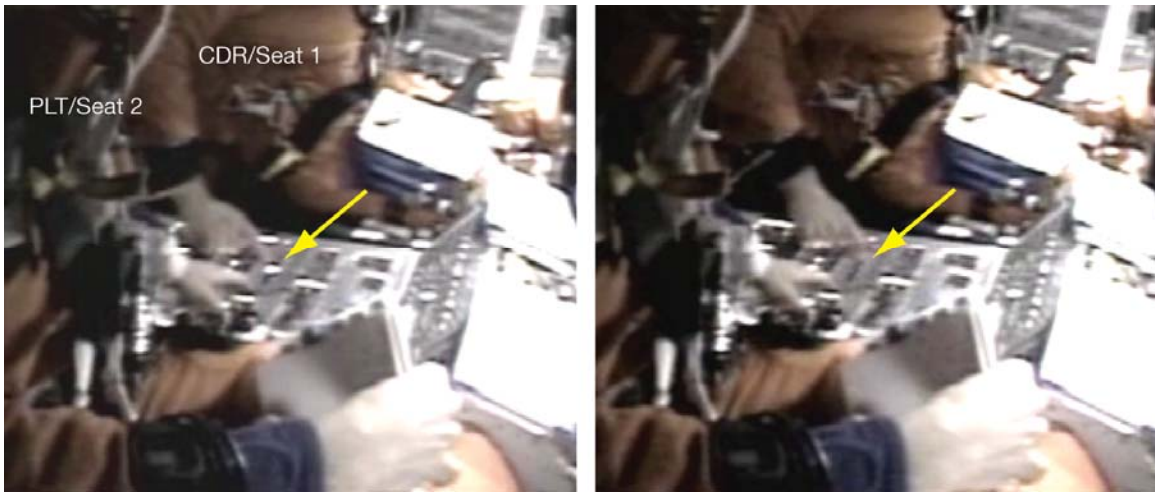


Figure 1.2-8. ***Digital autopilot is no longer in AUTO, INRTL light is ON*** **(left figure)*****; Commander/Seat 1 restores digital autopilot to AUTO, INRTL light is OFF*** **(right figure).**

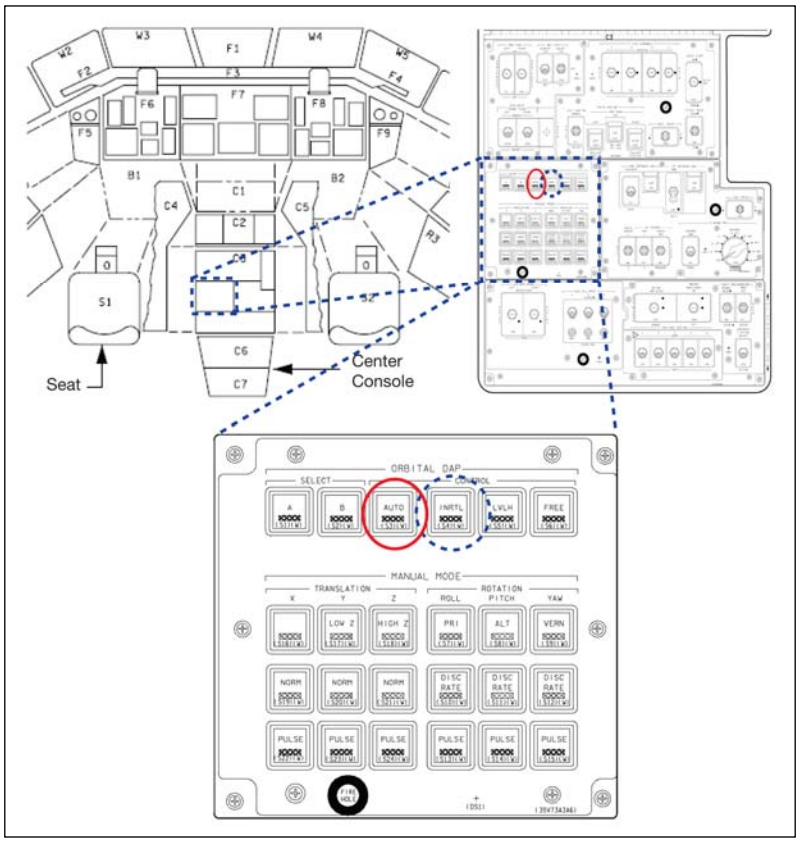


Figure 1.2-9. ***Location of the "INRTL" button*** **(blue circle)** ***that illuminated when the digital autopilot moded out of AUTO, and the "AUTO" button*** **(red circle)** ***that the Commander pressed to restore the digital autopilot to AUTO.*** **[Adapted from the Space Shuttle Systems Handbook]**

TIME		EVENT
13:37:31 (EI–398)		CAPCOM: "*Columbia*, Houston for Rick. We'll take another ITEM 27 please." This is required to resume the maneuver (to the EI–5-minute attitude) that was interrupted by the bumped RHC. The CDR/Seat 1 acknowledges the request. [*A/G recording and recovered flight deck video*]
13:37:39 (EI–390)		ITEM 27. CDR/Seat 1 manually inputs the ITEM 27 command using the keypad on the C2 panel. This fully recovers the vehicle from the bumped RHC. [*From video and telemetry*] (figures 1.2-10 and 1.2-11)

Figure 1.2-10. *Commander/Seat 1 inputting an ITEM 27*.

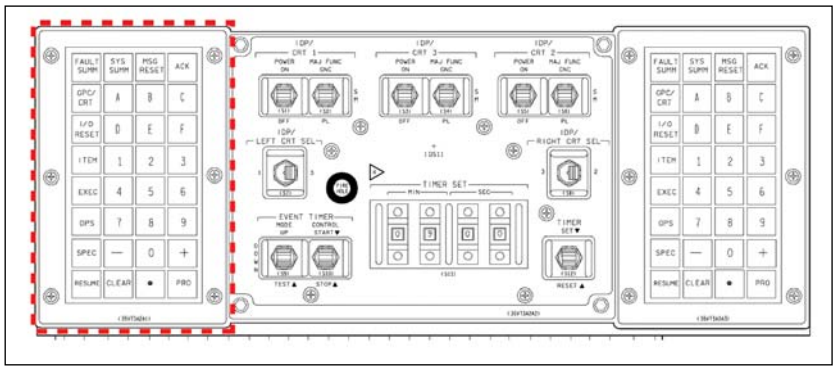


Figure 1.2-11. *Entry keypad* (red dashes) *on the C2 panel*. [Adapted from the Space Shuttle Systems Handbook]

13:37:44 (EI–385)		CDR/Seat 1: "And thanks for that, Houston. We gave you an ITEM 27. We bumped the stick earlier."
		CAPCOM: "Not a problem, Rick."
13:38:50 (EI–319)		CDR/Seat 1 enters the OPS 304 command into the queue. OPS 304 is the *Entry* Major Mode software.
13:38:56 (EI–313)		CDR/Seat 1: "And, Houston, we'll get the 304 at 5 minutes."
13:39:04 (EI–305)		CAPCOM: "Rick, we're ready for OPS 304."
13:39:09 (EI–300)		**OPS 304.** The CDR/Seat 1 executes OPS 304. [*Flight deck video, Telemetry, TDRS-W data – cathode ray tube (CRT) 1, Master Timeline, Rev. 15*]
13:40:12 (EI–237)		PLT/Seat 2 gloves are observed ON and MATED (figure 1.2-12).

TIME	EVENT

Figure 1.2-12. ***Pilot/Seat 2 observed with gloves ON and MATED*.**

1.2.2 Phase 2: Entry interface to loss of signal
[GMT 13:44:09 (EI) through GMT 13:59:32 (EI+923)]

13:44:09 (EI)

EI. This is the point where the orbiter is considered to be first encountering the atmosphere. (GPS derived) [*Master Timeline, Rev. 15 Baseline*]

Alt = 400,000 feet [per definition of EI]
Mach = 24.57 (*Master Timeline, Rev. 15 Baseline*]
Qbar = ~0.01 psf [modeling]

13:44:15 (EI+006)

CDR/Seat 1 states, “Just past EI.”

13:44:58 (EI+049)

CDR/Seat 1 request for everyone to check suit pressure integrity.

13:45:13 (EI+064)

CDR/Seat 1 observed with helmet visor down and latched in preparation for the suit pressure check and the communication check. MS2/Seat 4 visor observed OPEN and both gloves OFF (figure 1.2-13).

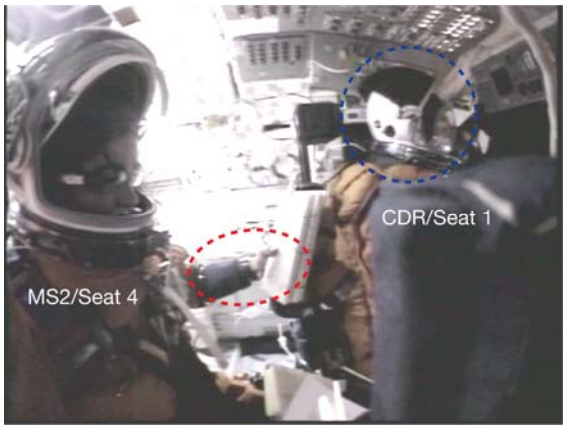


Figure 1.2-13. ***Commander/Seat 1 visor down and latched* (blue dashed circle) *and Mission Specialist 2/Seat 4 with helmet on and left glove off* (red dashed circle).**

TIME		EVENT
13:45:24 (EI+075)		**Suit Pressure/Communications (COMM) Check.** CDR/Seat 1, PLT/Seat 2, and MS4/Seat 3 complete COMM check. Visors are DOWN and locked during this check. Analysis of the O_2 supply pressure telemetry identified the O_2 supply pressure drop from the CDR/Seat 1, PLT/Seat 2, and MS4/Seat 3 suit pressure check.
		During the suit pressure integrity check, the CDR/Seat 1, PLT/Seat 2, and/or MS4/Seat 3 microphone activated the intercom system. Breathing is heard for about 20 seconds. CDR/Seat 1 comments, "It's noisy in there, isn't it?"
13:45:50 (EI+101)		MS4/Seat 3 states going visor UP after suit pressure check.
13:46:18 (EI+129)		PLT/Seat 2 is observed with visor OPEN.
13:46:48 (EI+159)		Mach = 24.66 [*Master Timeline, Rev. 15 Baseline*] Qbar = 0.5 psf [*Master Timeline, Rev. 15 Baseline*]
13:47:13 (EI+184)		**Accelerometer Bit Flip.** PLT/Seat 2 states that he observed a bit-flip on the accelerometer, indicating that the entry deceleration loads are starting to build as expected and were finally large enough to be registered by the vehicle accelerometers.
13:47:20 (EI+191)		CDR/Seat 1 states to crew, "We're at a hundredth of a G."
13:47:38 (EI+209)		CDR/Seat 1 and PLT/Seat 2 are observed with gloves still ON and MATED (red circles in figure below), visors OPEN. MS2/Seat 4 is observed starting to don gloves (yellow circle in figure) and is also observed with left glove ON but NOT MATED and right glove OFF (figure 1.2-14).

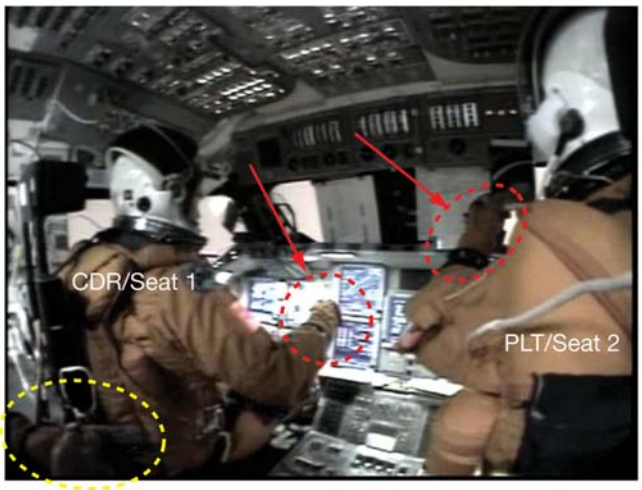


Figure 1.2-14. ***Mission Specialist 2/Seat 4 donning left glove; Commander/Seat 1 and Pilot/Seat 2 with gloves ON and MATED.***

TIME	EVENT
13:47:51 (EI+222)	MS4/Seat 3 with right glove still ON and MATED (figure 1.2-15).

Figure 1.2-15. ***Mission Specialist 4/Seat 3 right glove still ON and MATED.***

13:47:52 (EI+223)	Mach = 24.66 [*Master Timeline, Rev. 15 Baseline*] Qbar = 2.0 psf [*Master Timeline, Rev. 15 Baseline*]
	Elevon and Body Flap Active. When the Qbar increases to 2 psf, the elevons and body flap aerodynamic control surfaces become effective for controlling the vehicle and are added as active effectors to the vehicle control logic.
13:48:45 (EI+276)	**End of Recovered Flight Deck Video.** This is the last frame with a discernable image (figure 1.2-16).

Figure 1.2-16. ***Last discernable image from recovered flight deck video*. [Window 5 and the right side of the crew module visible]**

TIME		EVENT
13:49:16 (EI+307)		Mach = 24.57 [*Master Timeline, Rev. 15 Baseline*] Qbar = ~10.0 psf [*Master Timeline, Rev. 15 Baseline*]
		Roll Jets Deactivated. When the Qbar increases to 10 psf, the roll jets are removed from the control logic. [*Master Timeline, Rev. 15 Baseline*]
13:49:32 (EI+323)		**Start of First Planned Roll to the Right for Energy Management.** [*Master Timeline, Rev. 15 Baseline*] (figure 1.2-17).

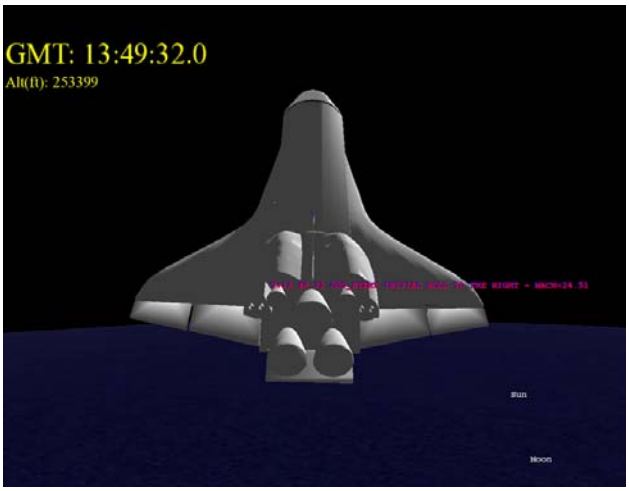


Figure 1.2-17. ***Attitude of*** **Columbia** ***at the start of the first planned roll.***

		Mach = 24.51 [*Master Timeline, Rev. 15 Baseline*] Qbar = ~15 psf [modeling]
13:50:00		**Completion of First Planned Roll to the Right for Energy Management.** [*Master Timeline, Rev. 15 Baseline*] (figure 1.2-18).

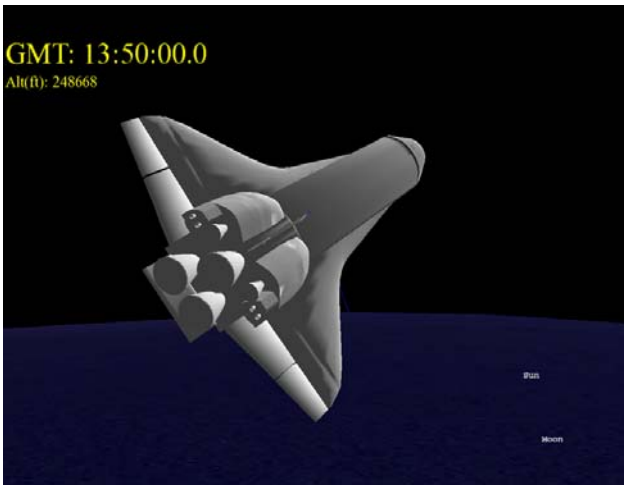


Figure 1.2-18. ***Attitude of*** **Columbia** ***at the end of the first planned roll.***

TIME		EVENT
13:50:30 (EI+381)		First indication of entry heating. A thermal sensor measurement and stimulus indication (MSID) V09T1702A in the aft fuselage center bottom bond line registers a normal rise in temperature due to entry heating. [*Master Timeline, Rev. 15 Baseline*]
13:50:53 (EI+404)		Start of peak heating. [*Master Timeline, Rev. 15 Baseline*]
		Alt = 243,048 feet [*Master Timeline, Rev. 15 Baseline*] Mach = 24.12 [*Master Timeline, Rev. 15 Baseline*] Qbar = ~22 psf [modeling]
13:51:19 -13:52:49		Nominal yaw jet firings were occurring during this time (GMT 13:51:19 (EI+430) to GMT 13:52:49 (EI+520)). When the yaw jet(s) fires, an indicator on the F6 panel will illuminate while the jet(s) is on (see arrows on figure 1.2-19). [*Master Timeline, Rev. 15 Baseline*]

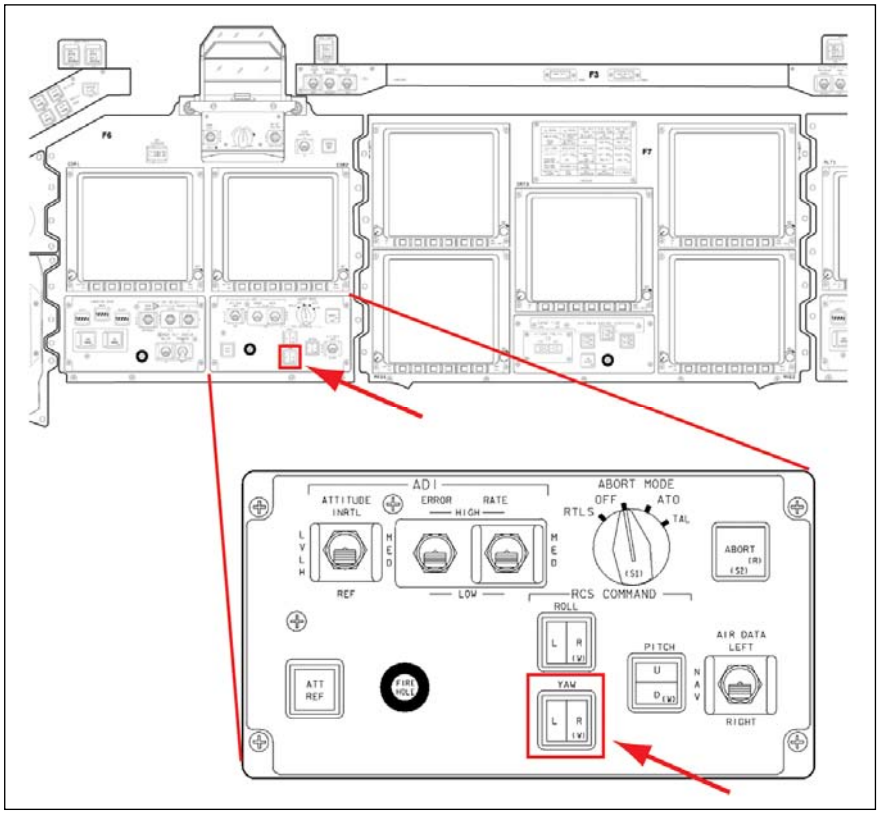


Figure 1.2-19. ***Location of the yaw jet indicator light*****. [Adapted from the Space Shuttle Systems Handbook]**

TIME		EVENT
13:51:46 (EI+457)		Inertial sideslip angle (beta) goes negative (yaw to port/ left) and stays negative until the LOS from *Columbia* at GMT 13:59:32 (EI+923) (figure 1.2-20). [*Master Timeline, Rev. 15 Baseline*]

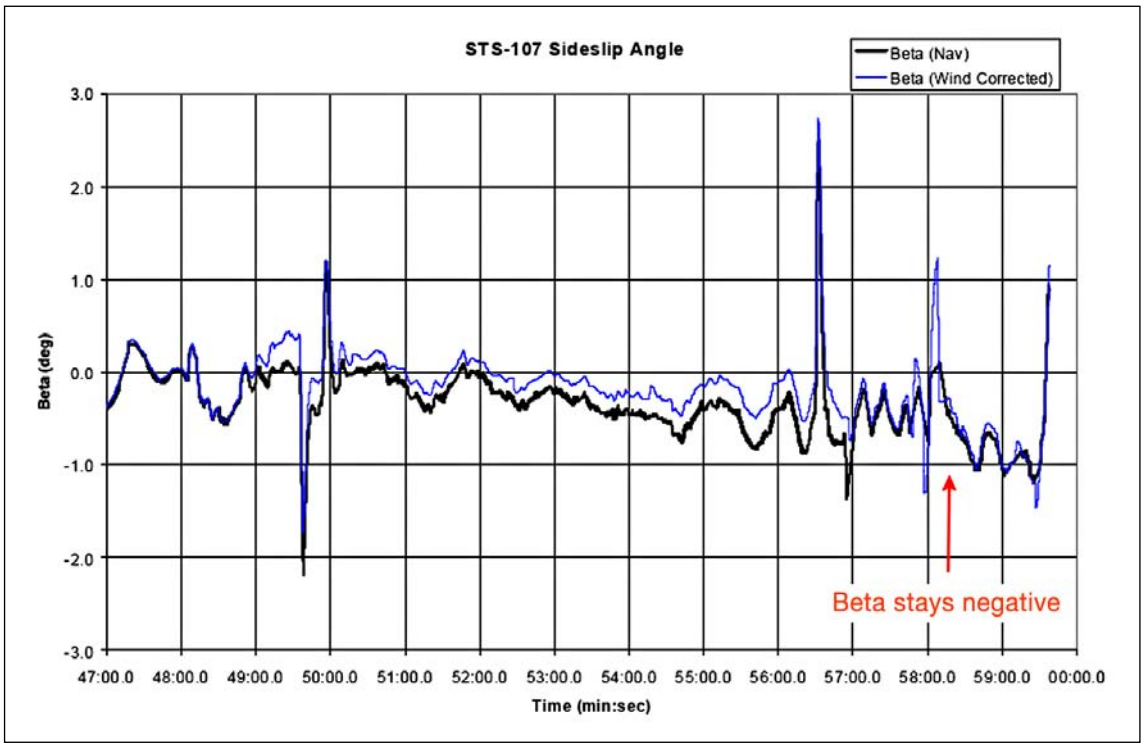


Figure 1.2-20. ***STS-107 sideslip angle vs. time.***[4]

13:52:05 (EI+476)		**Yaw-Moment Changed.** Post-accident analysis determined that this was the first clear indication of off-nominal aerodynamics. This information was not available/visible to the crew or controllers in real time. [*Master Timeline, Rev. 15 Baseline*]
13:52:17 (EI+488)		Left main landing gear brake line temperature sensor (MSID V58T1703A), which was located on the inboard sidewall of the wheel well, starts to indicate an off-nominal temperature rise rate. This is the first indication of off-nominal system readings in the left wing. The information is not visible to the crew. [*Master Timeline, Rev. 15 Baseline*]
		Damage to Left Wing Begins – NLT Time. Post-accident analysis determined that damage inside the left wing began NLT this time.[5]

[4]Integrated Entry Environment Team Final Report, May 30, 2003, Figure I-21, p. I-12.
[5]*Columbia* Accident Investigation Board Report, Volume I, August 2003, pp. 68 and 71.

TIME		EVENT
		Approximately 300 miles west of the California coast (figure 1.2-21) Alt = 236,791 feet [*Master Timeline, Rev. 15 Baseline*] Mach = 23.58 [*Master Timeline, Rev. 15 Baseline*] Qbar = ~27 psf [modeling]

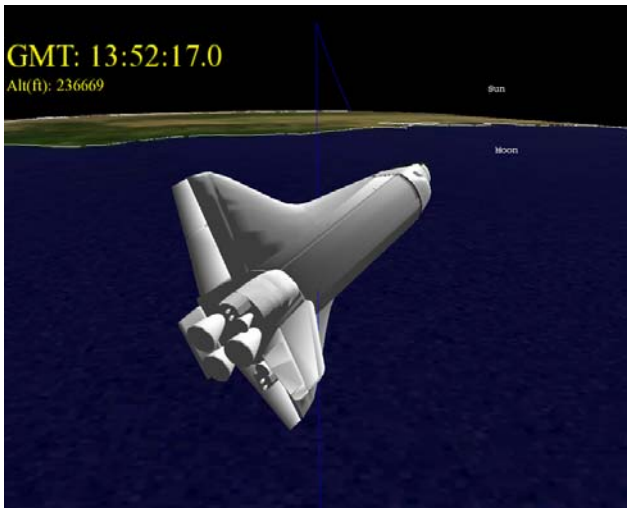


Figure 1.2-21. Columbia *approaching the California coastline*.

13:53:00 (EI+531)		Qbar = ~29 psf
13:53:01 (EI+532)		Off-nominal rolling moment. Post-accident analysis identifies the first clear indication of off-nominal rolling moment. Start of steady growth in roll moment. [*Master Timeline, Rev. 15 Baseline*]
13:53:10 (EI+541)		**First Indication in MCC of Off-nominal Readings.** Four hydraulic return line temperature sensors in the left wing went off-scale low (OSL) between GMT 13:53:10 (EI+541) and GMT 13:53:36 (EI+557). OSL refers to a reading that is below the lower display limit. When several sensors go OSL it usually indicates a suite of sensors has failed. These sensor failures were described in the CAIB Report. This information was not available to the crew.
13:53:15 (EI+546) ±2 sec		At GMT 13:53:15 (EI+546) ±2 sec, ground-based video coverage of *Columbia* is acquired by videographers who are unassociated with NASA (figure 1.2-22).

TIME	EVENT

Figure 1.2-22. ***This is the first frame of ground-based video. Columbia*** **is circled. Time 1 (TM1) shows the Greenwich Mean Time. The Pacific Standard Time displayed on the lower portion of the image is inexact.**

13:53:26 (EI+557) — *Columbia* crosses the California coastline west of Sacramento. [*Master Timeline, Rev. 15 Baseline*] (figure 1.2-23).

Figure 1.2-23. Columbia ***crossing the California coastline.***

Alt = 231,600 feet [*Master Timeline, Rev. 15 Baseline*]
Mach = 23.0 [*Master Timeline, Rev. 15 Baseline*]
Qbar = ~30 psf [modeling]

13:53:38 (EI+569) — Sideslip angle exceeded all previous flight experience.[6]

13:53:46 (EI+577) ±2 sec — **First Observed Incident of Debris Being Shed**. This is most likely a piece of the left wing. Debris 1 is seen just aft of the orbiter envelope 1 second after a trail anomaly that consisted of a noticeably luminescent section of the plasma trail. There were no reported entry observations while *Columbia* was over the Pacific Ocean prior to GMT 13:53:15 (EI+546), so it is

[6]Integrated Entry Environment Team Final Report, May 30, 2003, p. 15.

TIME	EVENT
	not possible to know whether debris was being shed before this time. Image of Debris 1 is 0.6 second after first visual detection of it. Analysis indicates that the mass of the debris was probably less than 8 lbs. Neither the crew nor MCC personnel were aware of the shedding events (blue arrow shows direction of flight) (figure 1.2-24).

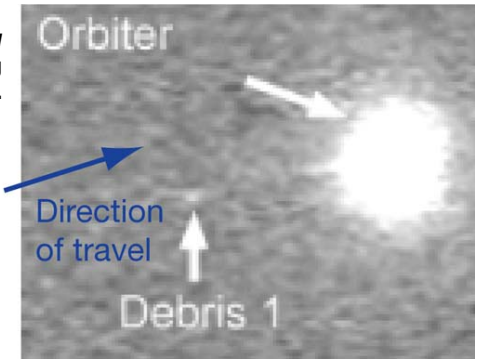


Figure 1.2-24. ***Video capture of the first observed incident of debris being shed.*** **The orbiter is traveling from left to right in this image.**

13:54:20 (EI+611) ±10 sec		The beginning of the slow elevon trim change starts at GMT 13:54:20 (EI+611) ±10 sec. While there is a display that shows the aileron trim movement (figure 1.2-25), the initial change was so small that it would not be detectable by the crew. [*Master Timeline, Rev. 15 Baseline*]

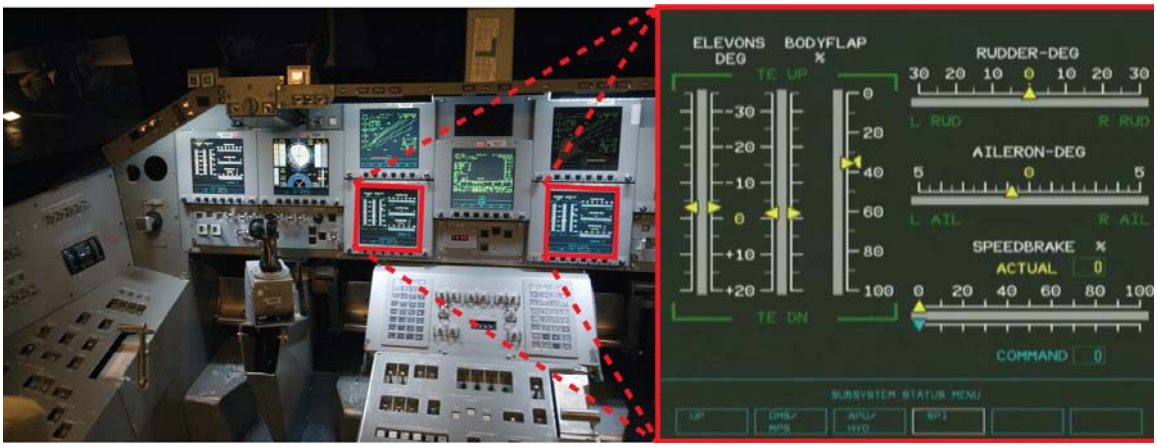


Figure 1.2-25. ***Elevon trim display*****. [Picture of the cockpit and display from the Shuttle Engineering Simulator]**

13:54:24 (EI+615)		**MCC Team Made Aware of Off-nominal Readings.** The MMACS flight controller makes a call over the MCC voice loops that the four hydraulic return line temperature sensors in the left wing went OSL at GMT 13:53:10 (EI+541).
13:54:25 (EI+616)		*Columbia* crosses the California-Nevada border. [*Master Timeline, Rev. 15 Baseline*] (figure 1.2-26).

TIME	EVENT

Figure 1.2-26. Columbia *crossing the California-Nevada border*.

Alt = 227,400 feet [*Master Timeline, Rev. 15 Baseline*]
Mach = 22.5 [*Master Timeline, Rev. 15 Baseline*]
Qbar = ~34 psf [modeling]

13:54:30 (EI+621) — Indication of a second suit pressure check by three to five crew members (figure 1.2-27).

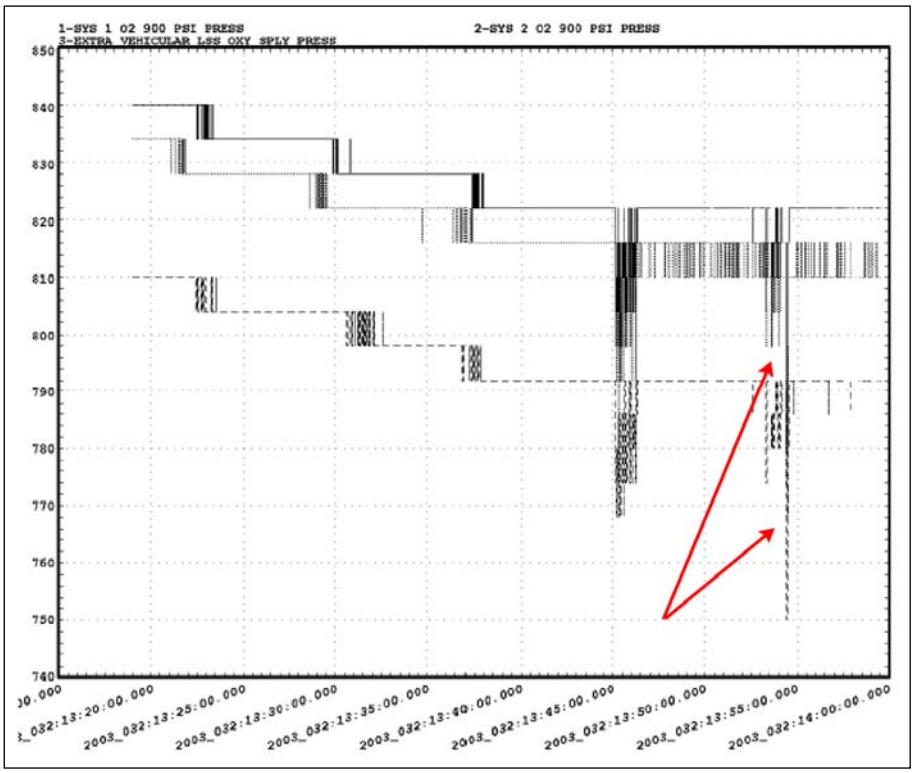
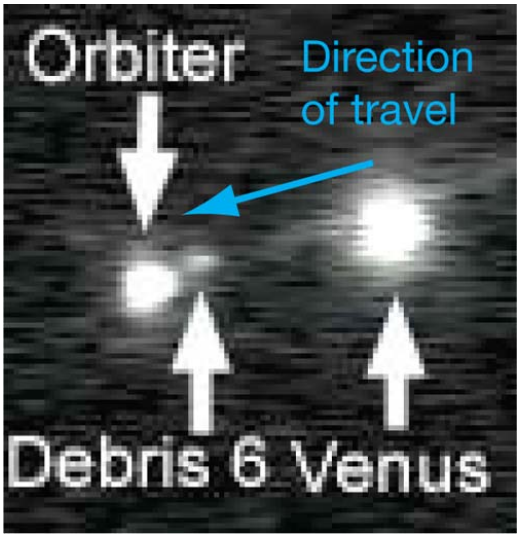


Figure 1.2-27. *Plot of main oxygen supply pressure showing second series of suit pressure checks* (red arrows).

TIME		EVENT
13:54:32 (EI+623)		Debris 6 released. The brightest debris-shedding event occurring in this phase, Debris 6, is first visible on video at GMT 13:54:36 (EI+627) (figure 1.2-28). Ballistic estimates determined that the actual release time was 4 seconds earlier. Luminosity measurements and calculated rates of deceleration were used to determine that the mass was probably a few hundred pounds.[7] There were no data from sensors, instrumental indications, or apparent crew recognition of this debris loss.

Figure 1.2-28. ***The brightest debris event known to have occurred prior to loss of signal.*** **The orbiter is traveling from right to left.**



13:54:33 (EI+624)		At GMT 13:54:33.52 (EI+624.52), RCS yaw jet R3R fired followed by RCS yaw jet R2R at GMT 13:54:33.54 (EI+624.54). It is unknown whether the jet firings were in response to the debris-shedding event. [*Master Timeline, Rev. 15 Baseline*]
13:54:36 (EI+627)		Debris 6 first visible on ground-based video.
13:55:21 (EI+672)		Atmospheric drag on the orbiter was producing the nominal deceleration load on the crew of approximately 0.3 G.
13:56:02 (EI+713)		Aft RCS pitch jets deactivated when Qbar reached 40 psf. At 40 psf, the elevons and body flap have sufficient control authority to control the pitch of the orbiter.
13:56:30 (EI+741)		**First Roll Reversal Initiated.** *Columbia* initiated a roll from right wing low to left wing low (figure 1.2-29).

[7] *Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, October 2003.

TIME	EVENT

Figure 1.2-29. ***Attitude at the start of the roll reversal*****.**

13:56:55 (EI+766)

First Roll Reversal Completed. Left wing low (figure 1.2-30).

Figure 1.2-30. ***Attitude at the completion of the roll reversal*****.**

Alt = 218,817 feet [*Master Timeline, Rev. 15 Baseline*]
Mach = 20.76 [*Master Timeline, Rev. 15 Baseline*]
Qbar = ~43 psf [modeling]

13:57:14 (EI+785) ±1 sec

Starfire Image. An image of the shuttle was taken while it was over the Starfire Optical Range located at the Kirtland Air Force Base, New Mexico. The photograph has been enhanced, and a wireframe representation of the orbiter has been overlaid. The bright areas behind the shuttle could be traced to orbiter sources and, thus, were generally considered to be nominal, as was the asymmetry that was seen in the bright gas around the nose area. The bulges that were seen

TIME	EVENT
	(as marked) at the left wing could not be explained by a nominal condition (figure 1.2-31).

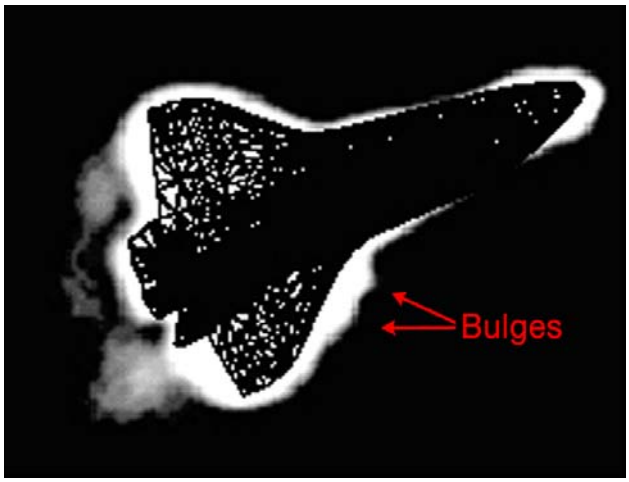


Figure 1.2-31. ***Infrared photograph of the shuttle during entry.***[8]

13:57:54 (EI+825) ±1 sec		Flare 1 is an asymmetrical brightening of the orbiter shape at GMT 13:57:54.7 (EI+825) ±1 sec. This brightening was detected in images taken by a charge coupled device camera on a telescope, so its image is well magnified. Another brightening event was detected 6 seconds later (see GMT 13:58:00 (EI+831)). The two small images are the raw images from the data. The left image is interpreted to be the orbiter in a nominal condition. The image on the right is 0.4 second later. The larger image has been rotated to its correct viewing orientation and enhanced; a wireframe model of the orbiter at approximately the correct scale and orientation has been overlaid (figure 1.2-32).

[8]*Columbia* Accident Investigation Board Report, Volume V, Appendix G.7, Starfire Team Final Report, June 3, 2003, Figure 3, p. 360.

TIME	EVENT

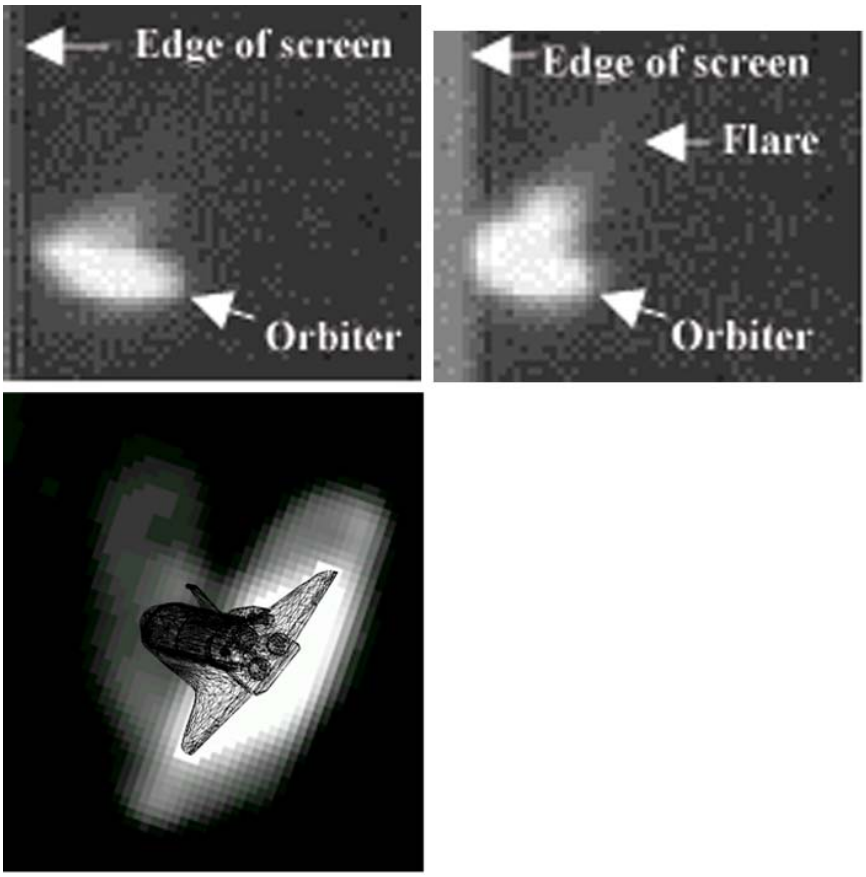


Figure 1.2-32. ***Enhanced video captures showing the first observed flare in the trail.***[9]

13:58:00 (EI+831) ±1 sec — Flare 2 is an asymmetrical brightening of the orbiter shape at GMT 13:58:00.5 (EI+831.5) ±1 sec (figure 1.2-33).

[9]*Columbia* Accident Investigation Board Report, Volume V, Appendix G.7, Starfire Team Final Report, June 3, 2003, Figure 6, p. 361 and Figure 6A, p. 362.

TIME **EVENT**

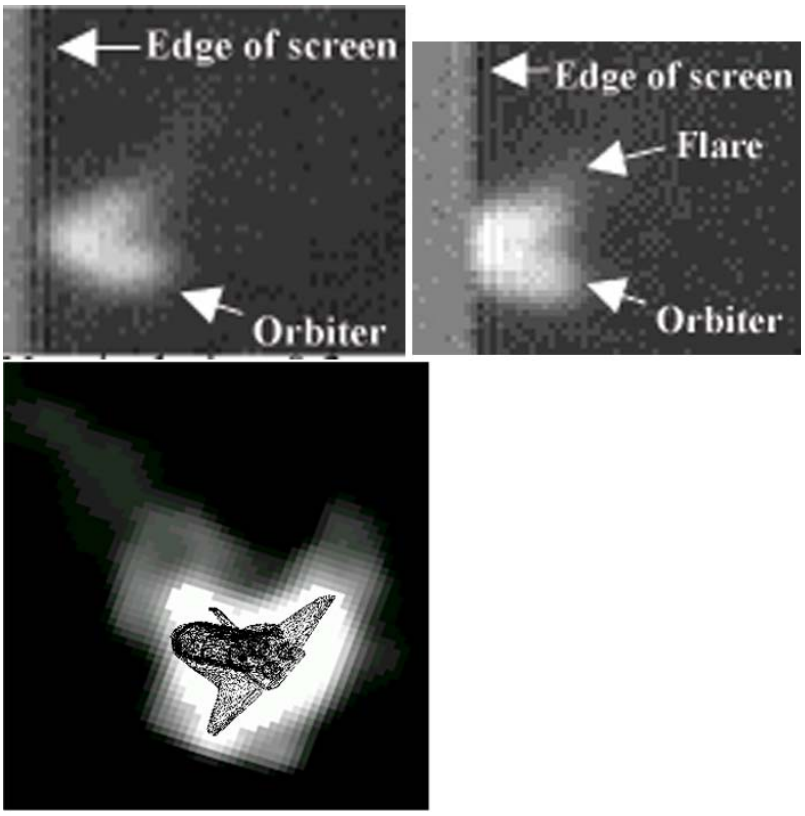


Figure 1.2-33. ***Enhanced video captures showing the second observed flare in the trail.***

Mach = 19.8 [modeling]
Qbar = ~53 psf [modeling]

13:58:03 (EI+834)

Start of "Sharp" Elevon Trim Increase. The FCS is now compensating for increasingly asymmetric aerodynamic loading and is commanding the elevon trim at a much higher than nominal rate. Time is uncertain (±10 sec) (figure 1.2-34). [*Master Timeline Rev. 15*]

TIME **EVENT**

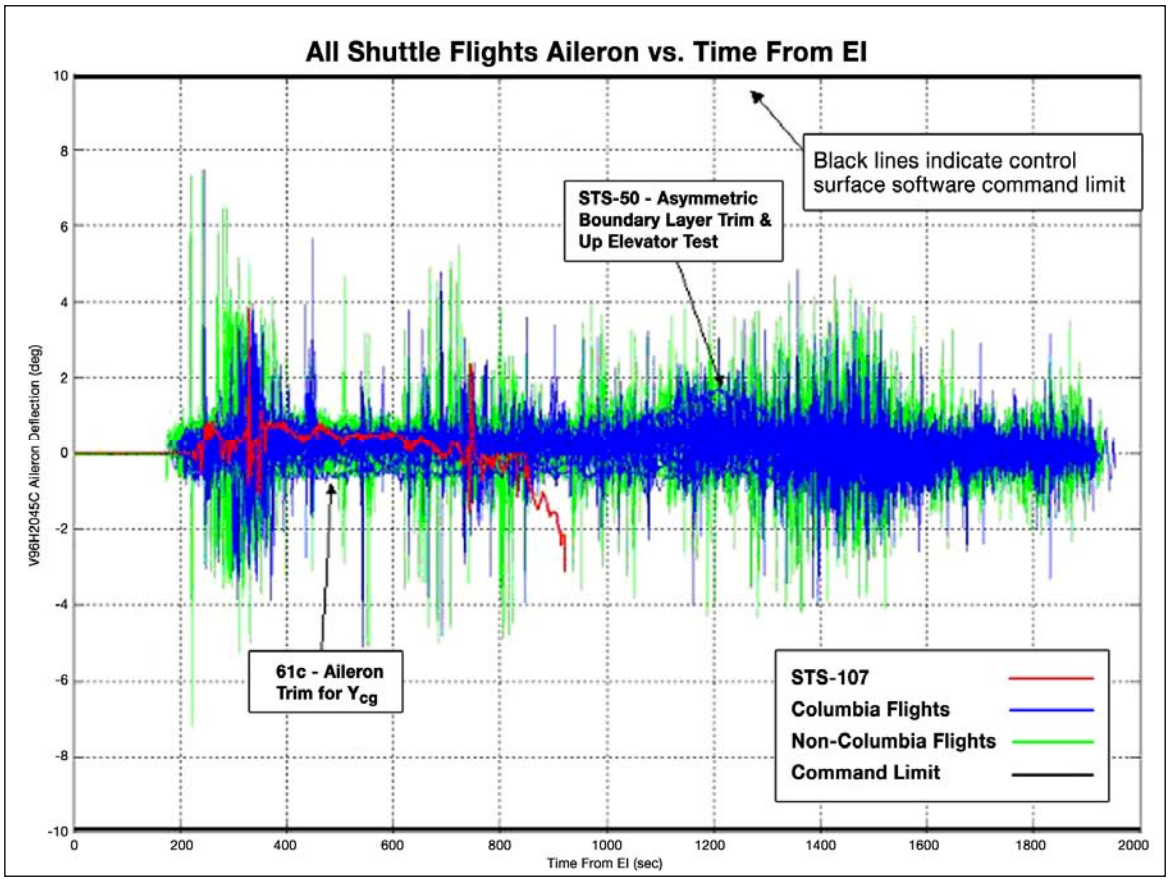


Figure 1.2-34. ***The sharp divergence from previous flight experience starts at approximately GMT 13:58:19 (EI+850).***[10]

Alt = 212,007 feet [*Master Timeline, Rev. 15 Baseline*]
Mach = 19.77 [*Master Timeline, Rev. 15 Baseline*]
Qbar = ~54 psf [modeling]

13:58:12 (EI+843) — End of ground-based video coverage for western portion of entry.

13:58:21(EI+852) ±5 sec — **Littlefield Tile Released**. This piece of tile was recovered in Littlefield, Texas; it is the westernmost piece of recovered debris (figure 1.2-35).

[10]Integrated Entry Environment Team Final Report, May 30, 2003, Figure 6.6-1, p. 30.

TIME	EVENT

Figure 1.2-35. *The Littlefield tile*. [Picture from the *Columbia* Reconstruction Database, debris item no. 14768]

13:58:39 (EI+870) — TIRE PRESS LOB. This message, indicating a left outboard (LOB) tire pressure fault, was recorded by the backup flight software (BFS) in the downlink stack and had a time of GMT 13:58:39.94 (EI+870.94). The fault message was annunciated on the crew displays and with an audio tone. [*TDRS-W data; Master Timeline, Rev. 15 Baseline*]

13:58:40 (EI+871) — Main landing gear left inboard (LIB) tire pressure reading went OSL. Downlink telemetry from the GPC recorded that the BFS was indicating a tire pressure fault message. The fault message was annunciated on the crew displays and with an audio tone. [*TDRS-W data; Master Timeline, Rev. 15 Baseline*]

13:58:41 (EI+872) — TIRE PRESS LIB. This message, indicating an LIB tire pressure fault, was recorded by the BFS in the downlink stack and had a time of GMT 13:58:41.84 (EI+872.84). The fault message was annunciated on the crew displays and with an audio tone. [*TDRS-W data; Master Timeline, Rev. 15 Baseline*]

TIME		EVENT
13:58:48 (EI+879)		A partial voice transmission from *Columbia* is received over the A/G: "And, uh, Hou…" The vehicle and crew were still performing nominally.[11]
		CRT3: FAULT SUMM. The crew called up the fault summary display to look at the messages. [*TDRS-W data*]
13:58:49 (EI+880)		BFS FSM: SM0 TIRE PRESS LIB. This message, indicating an LIB tire pressure fault, was recorded by the BFS in the downlink stack and had a time of GMT 13:58:49.54 (EI+880.54). The fault message was annunciated on the crew displays and with an audio tone. [*Master Timeline, Rev. 15 Baseline*]
13:58:56 (EI+887)		BFS FSM: SM0 TIRE PRESS LOB. This message, indicating an LOB tire pressure fault, was recorded by the BFS in the downlink stack and had a time of GMT 13:58:56.26 (EI+887.26). The fault message was annunciated on the crew displays and with an audio tone. [*Master Timeline, Rev. 15 Baseline*]
13:59:06 (EI+897)		Telemetry records the LEFT MAIN GEAR DOWN-lock sensor transferred to ON. This indicated that the left main landing gear was down and locked in the deployed position. Other sensors indicated that the landing gear door was still closed and the landing gear was locked in the stowed position. The mixed signals would result in the landing gear position indicator for the left gear displaying a "barber pole," which would indicate an indeterminate gear position. Analysis of the data and recovered debris indicates that the landing gear was locked in the stowed position and the landing gear door was closed. The signal, indicating that the gear was down, was a false signal that was likely triggered by damage to the sensor system (sensor, wiring harness, etc.) (figure 1.2-36).

[11] *Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 42.

TIME **EVENT**

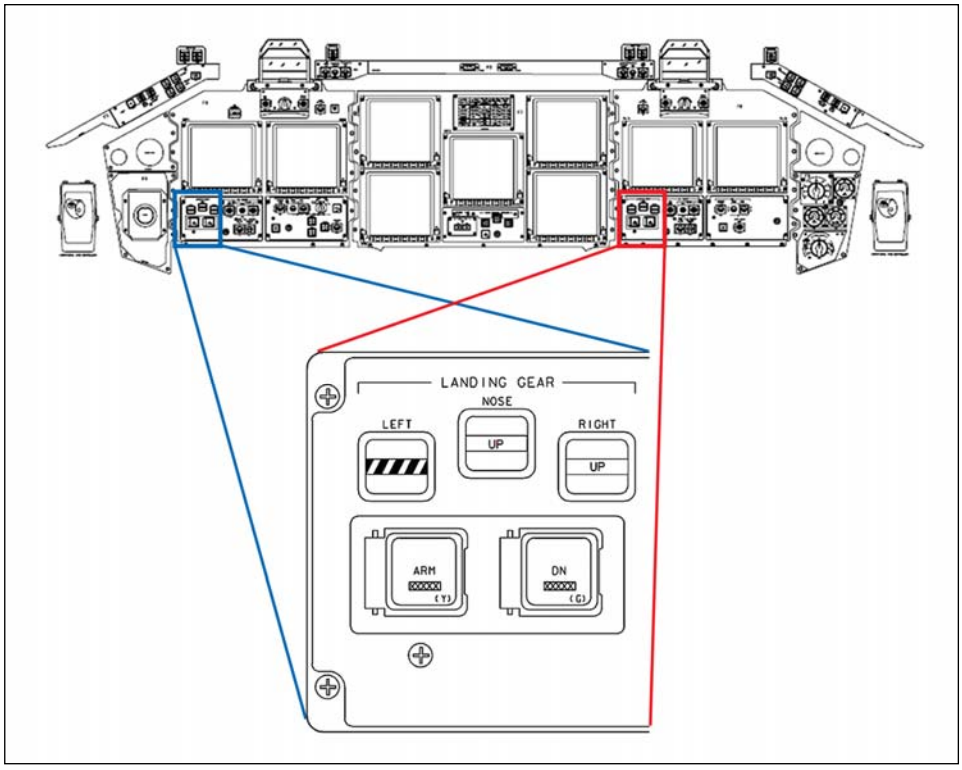


Figure 1.2-36. ***Landing gear indicator panel, identical on both sides of the flight deck forward display panels*****. Left indicator showing "barber pole" (indeterminate position). [Adapted from the Space Shuttle Systems Handbook]**

13:59:24 (EI+915)

The MCC calls *Columbia* regarding the tire pressure fault message, "And, *Columbia*, Houston, we see your tire pressure message and we did not copy your last call."[12] [*STS-107 A/G recording*]

13:59:29 (EI+920)

Aerodynamic control authority is exceeded. Damage to the left wing exceeds the elevon control surface ability to compensate. The aileron trim deflection derived from flight data shows that the aileron trim rate did reach the 4.2 deg/sec rate limit. This indicates that the amount of lateral control (aileron and yaw RCS jets) that was required to trim the vehicle was constantly increasing and by LOS+5 seconds was quickly approaching the limits of the FCS.[13]

Change in vehicle attitude from GMT 13:57:08 (EI+779) to GMT 13:59:29 (EI+920).

13:59:30 (EI+921)

RCS Yaw JETS FIRING. RCS Yaw Jets (R2R and R3R) Begin Firing Continuously. Aft right RCS yaw jets R2R and R3R started firing at GMT 13:59:30.66 (EI+921.66) and GMT 13:59:30.68 (EI+921.68), respectively, to correct an

[12]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 43.
[13]Integrated Entry Environment Team Final Report, May 30, 2003, p. 20.

TIME	EVENT
	increasing yaw to the left, and were still firing when all data were lost at GMT 13:59:37.4 (EI+928.4).
	The only indication the crew would have that the jets were firing is from status lights on the F6 panel underneath the CDR2 display and the fuel quantity display on the overhead panel decrementing (figure 1.2-37).

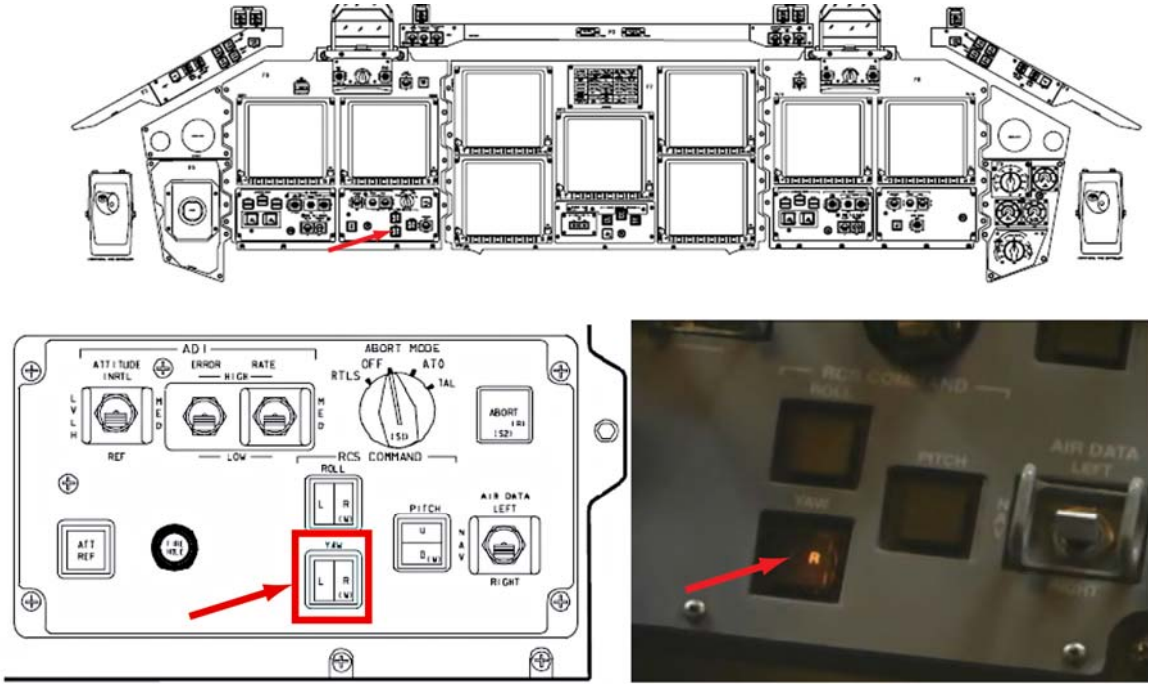


Figure 1.2-37. *Reaction Control System thruster status display*. [Pictures [top and bottom left) adapted from the Space Shuttle Systems Handbook; picture (bottom right) from the Shuttle Engineering Simulator]

13:59:31 (EI+922)		Fault Summary Page (FSP) Message Downlink Stack (last five messages) (figure 1.2-39):

FSP1: SM0	TIRE PRESS	L	OB	32/13:58:56.26
FSP2: SM0	TIRE PRESS	L	IB	32/13:58:49.54
FSP3: SM0	TIRE PRESS	L	IB	32/13:58:41.48
FSP4: SM0	TIRE PRESS	L	OB	32/13:58:39.94
FSP5: MPS[14]	PNEU		REG	32/13:58:04.42

Last observed elevon deflections (figure 1.2-38):[15]

[14]MPS = Main Propulsion System.
[15]Integrated Entry Environment Team Final Report, May 30, 2003, p. 15.

TIME	EVENT
	Left: –8.11 deg (deflected up) Right: –1.15 deg (deflected up)

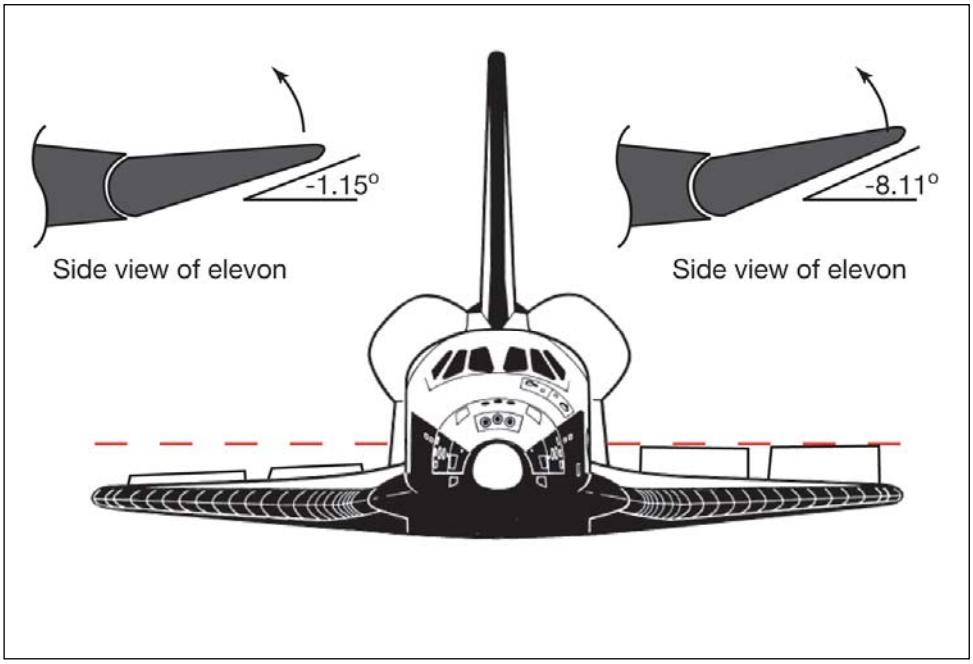


Figure 1.2-38. ***Illustration of the last observed elevon deflections*****.**

- At GMT 13:59:31.4 (EI+922.4) the FCS Channel 4 aerosurface position measurements start trending towards their null values, indicating a failure of the sensor due to a wiring short. This is the first indication of the eventual bypass of FCS Channel 4. [*Master Timeline, Rev. 15 Baseline*]

- At GMT 13:59:31.478 (EI+922.478) all of the FCS Channel 4 bypass valves close (i.e., bypassed condition). This is a leading indicator of an aeroservo actuator failure. [*Master Timeline, Rev. 15 Baseline*]

1.2.3 Phase 3: Loss of signal to Catastrophic Event
[GMT 13:59:32 (EI+923) – GMT 14:00:18 (EI+969)]

13:59:32 (EI+923)

- Near Dallas, Texas
  Alt = ~200,700 feet [*Master Timeline, Rev. 15 Baseline*]
  Mach = ~18.1 [*Master Timeline, Rev. 15 Baseline*]
  Qbar = ~70 psf [modeling]

TIME	EVENT
	Loss of Voice Communications. The last audio transmission from *Columbia*, "Roger, [truncated mid-word]…"[16] was cut off at GMT 13:59:32.136 (EI+923.136). A short-duration loss of voice communications was not unexpected because it coincided with the approximate time at which the on-board communications systems was to switch from the West TDRS to either the East TDRS or the ground station at Kennedy Space Center. [*STS-107 A/G recording*][17]
	LOS. The last valid downlink frame was accepted by the Orbiter Data Reduction Complex at GMT 13:59:32.136 (EI+923.136).[18] The following measurements were recorded in the MCC at the time of LOS: • Cabin pressure = 14.64 pounds per square inch absolute (psia) • Cabin temperature =71.6°F (22°C) • Humidity = 37.9% • ppO_2 levels (three sensors) 3.14 psia (sensor A), 3.14 psia (sensor B), 3.16 psia (sensor C) • Pressure change rate (DP/dt) = 0.004 psi/minute (within sensor bias limit for 0) • Partial pressure of carbon dioxide ($ppCO_2$) = 1.96 mmHg • Cabin temperature setting = full cool • Nitrogen (N_2) supply pressures 1011 psia (System 1), 1067 psia (System 2) • O_2 supply pressures 822 psia (System 1), 809 psia (System 2)
	Start of First Period of RGPC-1 Data. Data reconstructed from GMT 13:59:32.136 (EI+923.136) to GMT 13:59:37.396 (EI+928.396).
	It is possible that the crew noticed the aileron trim increasing. Per procedures, no crew action would be required until the trim reached 3 degrees.
	There is an increase in the roll error as the orbiter uses roll control to correct the yaw error and rate. [*RGPC*]

[16]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 43.
[17]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 43.
[18]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, 03/10/2003.

TIME	EVENT
	The FCS Channel 4 problems that were developing at GMT 13:59:31 (EI+922) had progressed to the point that the FCS Channel 4 fail flags tripped (1 Hz) on all aerosurface actuators. [*Master Timeline, Rev. 15 Baseline; RGPC*]
	During the RGPC-1 period, the vehicle remains in AUTO guidance and control.
	Ground-based video coverage is regained.
13:59:33 (EI+924)	PASS FSM: **FCS CH 4.** The fault message is associated with the removal of FCS Channel 4 from the control loop and would have been annunciated on the crew displays. [*TDRS-E data*]
	Master Alarm.[19] The Master Alarm associated with the FCS Channel 4 fault message would have been annunciated to the crew (figure 1.2-39).

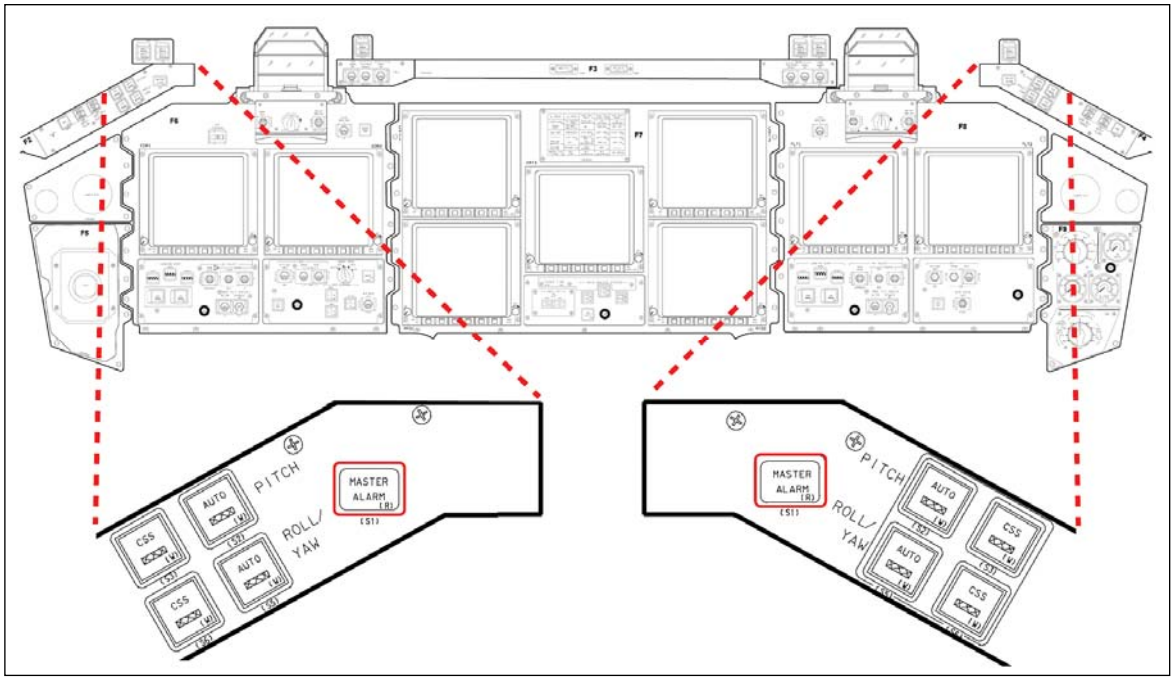


Figure 1.2-39. ***Location of the master alarm light*****. [Adapted from the Space Shuttle Systems Handbook]**

[19]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, p. 4, 03/10/2003.

TIME		EVENT
13:59:34 (EI+925)		FSP Message Downlink Stack (five deep):

FSP1: FCS	CH 4				32/13:59:33.68
FSP2: SM0	TIRE PRESS	L	OB		32/13:58:56.26
FSP3: SM0	TIRE PRESS	L	IB		32/13:58:49.54
FSP4: SM0	TIRE PRESS	L	IB		32/13:58:41.84
FSP5: SM0	TIRE PRESS	L	OB		32/13:58:39.94

13:59:36 (EI+927) **Third RCS Yaw Jet (R4R) Begins Firing Continuously.** Reconstructed data indicate that the DAP commanded a third RCS yaw jet (R4R) to fire at GMT 13:59:36.8 (EI+927.8) and that it fired continuously until end of data at GMT 13:59:37.4 (EI+928.4). [*Master Timeline, Rev. 15 Baseline*]

Aileron trim exceeds 3 degrees.

13:59:37 (EI+928) **LOSS OF CONTROL NO-EARLIER-THAN (NET) TIME**

Beginning of the Orbiter Pitch-up. Based on the time of the ROLL REF message (GMT 13:59:46 (EI+937)) and the divergence from the drag profile required to generate the ROLL REF alarm, this is the probable time that control was lost due to the probable loss of hydraulic pressure to the control surfaces. The loss of hydraulic pressure would have resulted in a Master Alarm being annunciated and the orbiter pitching up. While the drag had yet to diverge out of bounds of the drag profile, the orbiter was no longer in controlled flight. This marks the beginning of the transition from a controlled glide to an uncontrolled ballistic entry. Orbiter heating, lift, and drag would no longer be controlled, the ballistic number would be constantly changing with the changing attitudes, and downrange control would be lost (figure 1.2-40).

TIME **EVENT**

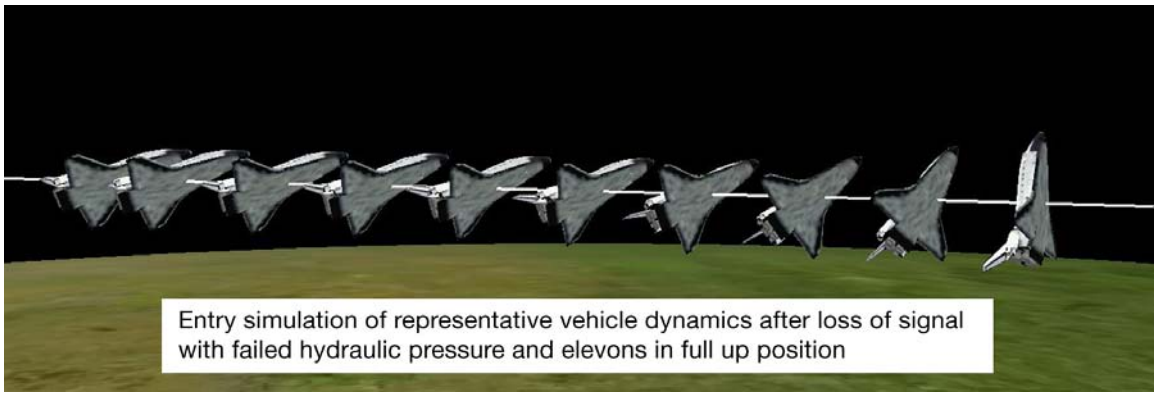


Figure 1.2-40. ***Sequence (1-second intervals) showing a simulation of orbiter loss of control pitch-up from GMT 13:59:37 (EI+928) to GMT 13:59:46 (EI+937).*** **White line indicates vehicle trajectory relative to the ground.**

Video imagery shows a dynamically changing orbiter trail after GMT 13:59:37 (EI+928) with a braided or corkscrew appearance, implying motion of the orbiter. However, the specific attitude of the orbiter cannot be derived from ground-based imagery. Brightening events, objects separating, "puffs," and splitting of the trail are all seen in the video during this timeframe. Ballistic analysis of debris could not positively correlate shedding events seen in the video to a specific orbiter source. However, it is known that the left wing and left OMS pod were being compromised.[20] Figure 1.2-41 shows video frames from GMT 13:59:35.5 (EI+925.5) through GMT 13:59:43.5 (EI+934.5) and displays some of these dynamic changes, although they are much more clearly seen in the video.

[20]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 68.

TIME **EVENT**

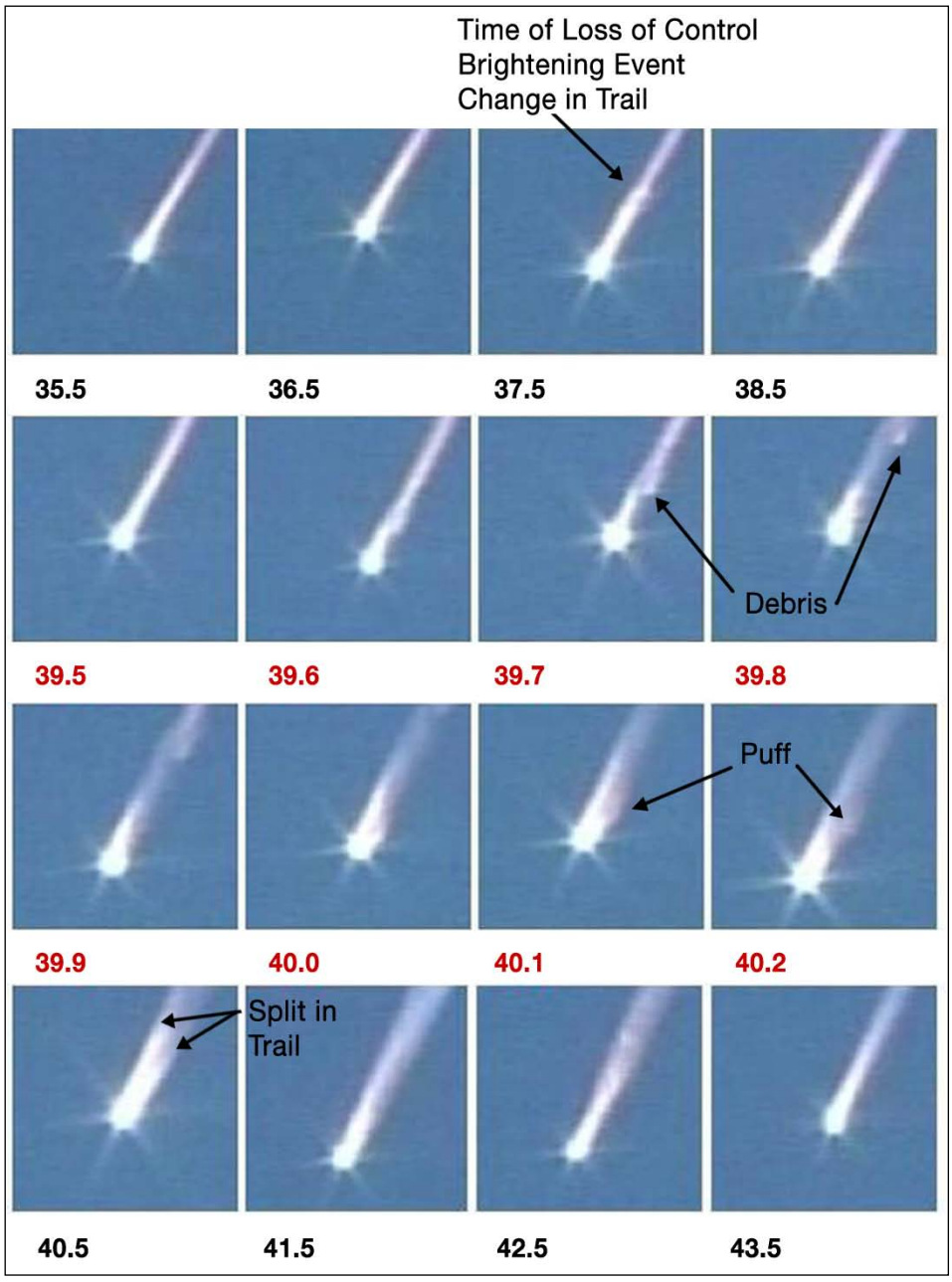


Figure 1.2-41. ***Video frame captures from ground-based video, beginning at GMT 13:59:35.5 (EI+926.5) and ending at GMT 13:59:43.5 (EI+937.5).*** **The numbers below each frame indicate the seconds after GMT 13:59:00. The frames in the first and last rows are 1 second apart. The frames in the second and third rows are 0.1 second apart.**

Beginning of Off-nominal Attitude and Loads. The CDR/Seat 1 and PLT/Seat 2 would have been aware of the off-nominal movement of the orbiter based on information from the flight displays and from the changing view in the forward windows. MS2/Seat 4 and MS4/Seat 3 may also have been aware of this information.

TIME		EVENT
		The rate of change of the elevon trim had reached the maximum allowed by the DAP while in AUTO. [*Master Timeline, Rev. 15 Baseline*]
		RGPC data recorded the left rudder pedal transducer assembly (RPTA) was near null at GMT 13:59:37.118 (EI+928.118),[21] indicating that there were no crew inputs on the rudder pedals.
		Fourth RCS Yaw Jet (R1R) Begins Firing. Reconstructed data indicate that the DAP commanded a fourth RCS yaw jet (R1R) to fire at GMT 13:59:37.3 (EI+928.3). The R1R fired continuously until the end of data at 13:59:37.396 (EI+928.396). With four jets firing continuously, the fuel supply would be rapidly depleted (within 30 seconds).
		End of RGPC-1. This is the end of the ~5-second period of RGPC data at GMT 13:59:37.396 (EI+928.396).[22] There were no confirmed indications of changes in positions of any of the switches on panel R2.[23] No GPC data were recoverable for the next 25 seconds.[24]
13:59:44 (EI+935)		Fuselage upper surface canopy thermocouple BP0439T (above window 2) starts to indicate an off-nominal sharp temperature increase followed by a drop to loss of sensor signal.[25]
13:59:46 (EI+937)		**ROLL REF.** BFS recorded the PASS generating a Roll Alarm fault message. At this time, the drag of the vehicle had exceeded the limits of the entry drag profile. This message occurred less than 10 seconds after the fourth yaw jet (R1R) began firing, suggesting a rapid change in orbiter aerodynamics. [*Master Time Line, Rev. 15 Baseline; BFS, RGPC-2*]
		Debris A (possible Left OMS Pod Cover).[26] Video analysis identifies GMT 13:59:46.67 (EI+937.67) as the earliest time at which Debris A is observed. Debris A was identified as possibly being the cover of the left OMS pod. Debris A fades from view on the video at GMT 14:00:22.97 (EI+973.97).

[21]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, p. 4, 03/10/2003.
[22]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, p. 4, 03/10/2003.
[23]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, p. 2, 03/10/2003.
[24]Data in RGPC-2 had time-stamped data for some events that occurred during this 25-second period between RGPC-1 and RGPC-2.
[25]OEX Data Evaluation of End of Mission Data for STS-107, Vehicle Data Mapping Team, OV-102 Investigation, 08/22/2003, Rev 2.
[26]See Section 2.2, Figure 2.2-9 Debris Tree.

TIME		EVENT
13:59:52 (EI+943)		At GMT 13:59:52.114 (EI+943.114) the BFS recorded the PASS generating an "L RCS LEAK" fault message. [*RGPC-2*] [*Master Timeline, Rev. 15 Baseline*].
		Debris D (possible Left OMS Pod). Video analysis identifies this as the earliest time at which Debris D is observed. Debris D was identified as possibly being the left OMS pod. (figure 1.2-42).

Figure 1.2-42. ***Debris D is depicted in this frame.***

14:00:01 (EI+952)		At GMT 14:00:01.54 (EI+952.54) the BFS recorded an "L RCS LEAK" fault message. [*Master Timeline, Rev. 15 Baseline*]
		RHC Moved. The RHC moved beyond neutral between approximately GMT 14:00:01.7 (EI+952.7) and GMT 14:00:03.6 (EI+954.6). The uncertainty in the time of the event is due to the processing rates of signal detection, signal processing, and message annunciation. This event resulted in the DAP downmode RHC Fault message that was recorded at GMT 14:00:03.637 (EI+954.637). By GMT 14:00:3.678 (EI+954.678) the crew returned the DAP back to AUTO, indicating that this RHC movement was accidental.
		At GMT 14:00:01.90 (EI+952.90) the BFS recorded another "L RCS LEAK" fault message. [*Master Timeline, Rev. 15 Baseline*]
14:00:02 (EI+953)		**Final Recovered GPS State Vector at GMT 14:00:02.12 (EI+953.12).** These data were recovered from the MAGR.[27]

[27] *Columbia* Investigation: GPS Receiver (MAGR) Memory Extraction Summary and Disposition presentation, May 12, 2003.

TIME		EVENT
		PASS FSM: LJET L RCS. This Class 2 alarm message may not be the result of the actual vehicle condition since the message was generated based on data from a sensor path that contained noise. If valid, the message is used to indicate when an RCS jet has failed on or failed off, or is leaking. The failed off condition could be an indication that the fuel was exhausted.
		Beginning of Final Period of RGPC Data at GMT 14:00:02.660 (EI+953.660). Reconstructed GPC data (RGPC-2), lasting from GMT 14:00:02.660 (EI+953.660) to GMT 14:00:04.826[28] (EI+955.826), contained multiple bit errors. Vehicle rates during this period were (Note: there is some uncertainty with the NAV-derived parameters since the high rates may have resulted in corruption of the inertial measurement unit state): • roll rate command from the DAP was at –5.0 deg/sec during the entire RGPC-2 period, indicating that the DAP was in AUTO the entire time (maximum control rates allowed: AUTO is –5.0 deg/sec, control stick steering (CSS) is –6.0 deg/sec) • Roll rate transitioned from +7 deg/sec (right roll) to –23 deg/sec (left roll) • Yaw rate was at the sensor maximum of 20 deg/sec (nose right) • Pitch rate was 5 deg/sec
		Debris B (Portion of Left Wing). Video analysis identifies this as the earliest time at which Debris B is observed.
		Debris C (Portion of Left Wing). Video analysis identifies this as the earliest time at which Debris C is observed.
14:00:03 (EI+954)		At GMT 14:00:03.47 (EI+954.47) BFS recorded an "L OMS TK P" fault message. This message is annunciated when the left OMS oxidizer ullage pressure (V43P4221C) or fuel tank ullage pressure (V43P4321C) is out of limits either high or low.
		DAP DNMODE RHC Fault Message. At GMT 14:00:03.637 (EI+954.637) the DAP Downmode RHC Fault message was recorded in the fault message buffer due to the movement of the RHC beyond neutral between approximately GMT 14:00:01.7 (EI+952.7) and GMT 14:00:03.6 (EI+954.6). This fault message was corroborated by an initialization flag for the aerojet DAP roll stick function. During the final 2-second RGPC period, all available data indicate that the RHC remained

[28]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, p. 1, 03/10/2003.

TIME		EVENT
		in the detent position and that the DAP was in AUTO. This supports the RHC being bumped and the crew immediately returning the DAP to AUTO.
		DAP in AUTO. At GMT 14:00:03.678 (EI+954.678) the RPGC data recorded the first indication from PASS that the DAP was in the AUTO mode.[29] This requires a manual crew input by either the CDR/Seat 1 or the PLT/Seat 2. This manual command was in response to the RHC movement at GMT 14:00:01 (EI+952) moding the DAP out of AUTO. Because moding the DAP from CSS to AUTO requires either the CDR/ Seat 1 or the PLT/Seat 2 to press two buttons located on both panels F6 (CDR/Seat 1) and F8 (PLT/Seat 2), at least one crew member was conscious and able to respond to events that were occurring on board and that the vehicle dynamics were within human performance capabilities for this action.
14:00:04 (EI+955)		**RHC in Detent.** At GMT 14:00:04.179 (EI+955.179) the RGPC data recorded the first indication from PASS or BFS that the RHC was in detent.[30] This indicates that there were no manual inputs at this time on the RHC.
		MSG Reset on CRT 1. BFS recorded an MSG reset on CRT 1 some time after GMT 13:59:37 (EI+928) and before GMT 14:00:05 (EI+956).[31] This event is part of the RGPC-2 data and is considered a valid input. This action would be a nominal crew response to a fault message and requires a crew member to manually acknowledge the message by keyboard entry on the C2 panel.
		End of RGPC-2. End of the final 2-second period of RGPC-2 data at GMT 14:00:04.826 (EI+955.826). At this time, the forward/mid/aft fuselage, PLBDs, right wing, and right OMS pod were still intact based on the following data: [*Master Timeline, Rev. 15 Baseline*] • APUs were running and the panel R2 switches were ON. The APUs are located in the aftbody, indicating that this portion of the vehicle was still intact. • Water spray boiler (WSB) cooling was evident. The WSB is located in the aftbody, indicating that this portion of the vehicle was still intact and the data lines from it to the crew module were still intact.

[29]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, 03/10/2003.
[30]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, 03/10/2003.
[31]STS-107 Investigation Action Response: OVE-204 Crew Inputs After Loss of COMM (Voice); CAIB-MRT-00099, 03/10/2003.

TIME	EVENT

- Panel R2 switches in nominal positions. Since panel R2 was recovered with switches in positions consistent with an attempted APU restart, this indicates that the switch throws occurred after this time.
- Fuel cells were generating power, indicating that electrical power was still being produced.
- Power reactant, storage, and distribution tanks and lines were intact, indicating that this portion of the vehicle was still relatively intact.
- Main Propulsion System still intact indicated that the aft portion of the vehicle was still relatively intact.
- Helium tanks and lines were intact.
- Freon loops and radiators (located in the PLBDs) were intact; quantities, flow measurements, and pressure measurements were nominal. This indicates that the PLBDs were still intact at this time.
- RHCs were in detent during the entire period of RGPC-2, indicating that the crew had not tried to manually control the vehicle during this time period.
- Left RPTA indicated a small left rudder input/offset that remained nearly constant for the duration of RGPC-2, indicating that there may have been some pressure being applied on the pedal by one of the crew members. It should be noted that the RPTA signals are not transmitted to the flight control software at this phase of flight. In addition, the lack of hydraulic pressure would prevent control inputs to the control surfaces.
- Communications and navigation systems in the forebody were performing nominally, indicating that this part of the vehicle was relatively intact.
- Environmental Control and Life Support System (ECLSS) performance was nominal, indicating that the cabin environment was nominal.

The following systems were indicating off-nominal conditions:

- Hydraulic supply pressures were reading 0 psi and the reservoir quantities were at 0% on all three systems; the lack of hydraulic pressure results in the aerosurfaces (elevons, body flap, and rudder) were free to move in the wind stream.
- LIB and LOB elevon actuator temperatures were either OSL or no data existed.
- APU lube oil was possibly being overcooled by the WSB.
- The Flash Evaporator System appears to have shut down.
- Most of the left OMS pod sensors were OSL or off-scale high, or had no data available.

TIME		EVENT
		• The electrical power distribution circuit showed a general upward shift in main bus amps and a downward shift in main bus volts. • Alternating current (AC) Bus 3 (AC3) Phase A inverter was off-line; this would not impair crew or vehicle performance. • Elevated temperature readings were recorded at the bottom bondline centerline skin forward and aft of the wheel wells and at the portside structure over the left wing.
		Another left wing piece separates based on ballistics analysis. This analysis was performed on a piece of recovered structure that was identified as being from the left wing.
14:00:05 (EI+956)		**End of GPC signal.** The last GPC signal, which contained no recoverable data, was recorded at GMT 14:00:05.121 (EI+956.121).
		PLT/Seat 2 takes actions in apparent attempt to restart APUs 2 and 3 . With the loss of hydraulic pressures and the vehicle LOC, the crew likely assumed a generic problem with the APUs. A crew module panel was recovered with switch configurations indicating an attempt by the PLT/Seat 2 to recover the hydraulic systems and hydraulic pressure by performing steps to initiate the restart of APUs 2 and 3. Switches for APU 1 were in the nominal position. Switches for hydraulic circulation pumps 2 and 3 were also in the "On" position. While turning on the hydraulic circulation pump is not on the emergency checklist, it nonetheless can provide some limited hydraulic pressure and shows good systems knowledge by the crew members as they worked to attempt to restore orbiter control. These actions took place after the end of RGPC-2 (after GMT 14:00:05 (EI+956)) and prior to loss of consciousness.
14:00:11 (EI+962)		Video analysis identified a color change in the plume that is likely the result of an OMS tank rupturing.
14:00:13 (EI+964)		The last MADS/OEX recorder data value (left wing spar cap sensor, V12G909A), marked GMT 14:00:13.439 (EI+964.439), is recorded.[32]

[32]OEX Data Evaluation of End of Mission Data for STS-107, Vehicle Data Mapping Team, OV-102 Investigation, 08/22/2003, Rev 2.

TIME **EVENT**

1.2.4 Phase 4: Catastrophic Event to Crew Module Catastrophic Event
[GMT 14:00:18 (EI+969) through GMT 14:00:53 (EI+1004)]

14:00:18 (EI+969)

CATASTROPHIC EVENT. The CE is a period of time during which the orbiter vehicle is undergoing a major structural breakup. At this time, the accelerations on the forebody were estimated to be 3.5 Gs. The breakup sequence progressed over several seconds. Analysis of ground-based video of the event established the first detectable signs at GMT 14:00:18 (EI+969). Based on engineering analysis, the CE is thought to have started with the compromise of the PLBDs, exposing the payload bay longeron sill to entry heating. The skin splice between the midbody and the X_o 582 ring frame bulkhead area, aft of the starboard x-link, failed due to a combination of mechanical and thermal loads. The forebody rotated away from starboard to port, causing the port x-link to fail due to bending loads. As the forebody separated from the midbody, various power, data, and ECLSS lines failed and the crew module was free to move forward and strike the inside of the forward fuselage. At GMT 14:00:25 (EI+976), the separated forebody is marginally visible. The CE is further supported by a collection of evidence that includes ground-based video of a large debris-generating event, the loss of the MADS/OEX recorder function, and the debris footprint (figures 1.2-43 and 1.2-44).

Figure 1.2-43. ***The Catastrophic Event is depicted in these three frames of video that cover 0.1 second.*** **There is no change in the magnification/zoom factor. The third frame represents GMT 14:00:18.3 (EI+969.3).**

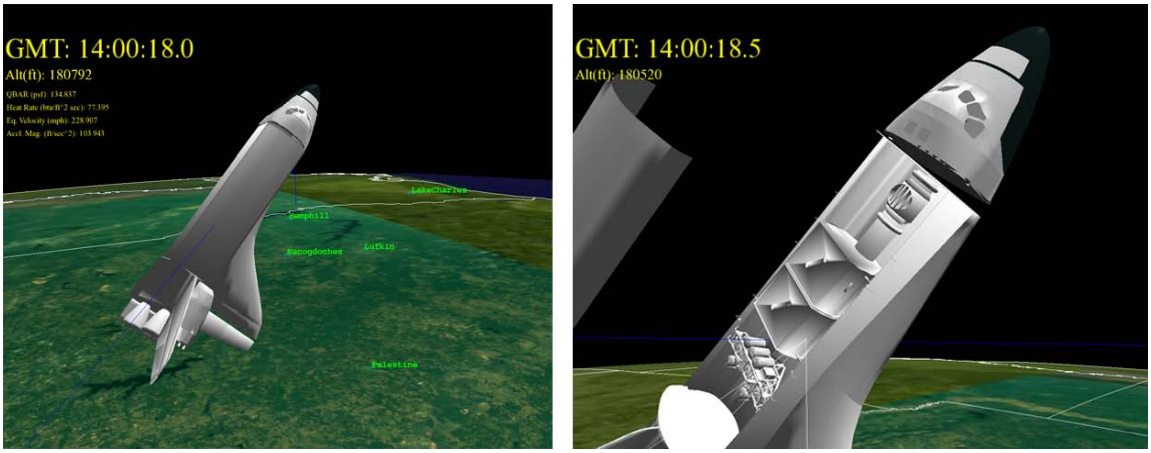


Figure 1.2-44. ***Simulation of the Catastrophic Event.***

TIME	EVENT
	The separation of the forebody from the midbody resulted in: • The crew module and forward fuselage becoming independent from the rest of the vehicle, moving farther downrange and remaining higher in altitude than the rest of the vehicle. Since there was no other debris ahead of the forebody, all material depositions and shock wave damage on the recovered forebody debris was generated by the forebody. • Loss of electrical power – Electrical power for the orbiter systems is supplied from fuel cells located in Bays 1 and 2 of the midbody of the vehicle. Separation of the crew module and forward fuselage from the midbody caused a loss of power to the crew module. Loss of electrical power resulted in the loss of all powered systems, including: o Displays – The loss of displays resulted in a loss of all situational awareness from instrumentation. o Intercom System – The loss of the intercom system meant that crew-to-crew communications could only be performed by shouting. o Lighting – With the cabin lights lost, the only lighting was from the windows and the Cyalume chemical lights that are pre-positioned within the crew module and on the upper arms of the ACES of each of the crew members. o Ventilation – Electrically powered fans for circulating air through the cabin and the CO_2 scrubber shut down, resulting in a loss of cabin ventilation. o O_2 supply –Valves in the orbiter O_2 system are designed to close when power is lost, resulting in a loss of O_2 supply to the cabin and the suit O_2 hoses. Since none of the helmet visors were lowered and locked, it is unlikely that the crew members activated their emergency O_2 supplies since that step is performed after visors are lowered and locked. o MAGR – Last time value was recorded at GMT 14:00:18.6875 (EI+969.6875). No state vector data were recorded at this time. • Sudden change in aerodynamic characteristics – The ballistic number of the forebody by itself is higher than the ballistic number of the rotating intact orbiter. This resulted in a reduction of drag with a corresponding reduction in translational force on the crew, estimated to have dropped from 3.5 Gs to 1 G. • Unknown dynamic changes – In addition to the change in translational forces acting on the crew, the separation of the forebody resulted in a change to rotational loads.

TIME	EVENT
	Prior to the CE, the crew was experiencing rotational loads with the center of rotation located aft of the crew module at the center of mass of the orbiter. As a result of the crew module and forward fuselage separating from the rest of the orbiter, the rotational arm decreased (figure 1.2-45).

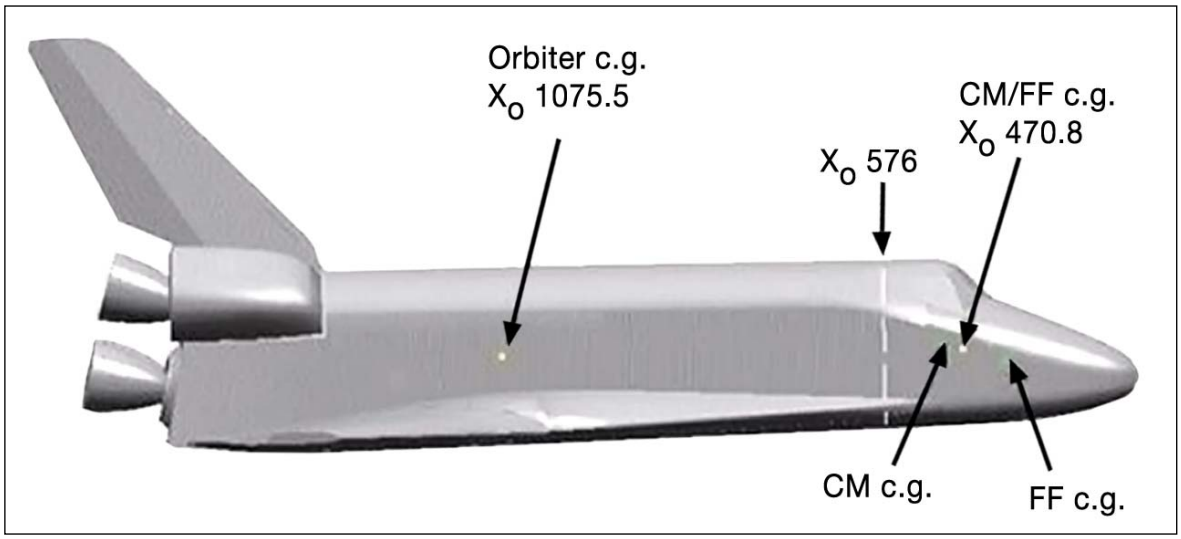


Figure 1.2-45. ***X-axis center-of-gravity locations for the intact orbiter, the crew module, the forward fuselage, and the forebody (crew module plus forward fuselage).*** **The X_o 576 is the aft bulkhead of the crew module.**

		• Based on video analysis, triangulation data, and aerodynamic modeling, the initial rotation rate of the separated forebody was low, the rates built over time, and the forebody never trimmed.
		The estimated vehicle state at GMT 14:00:18 (EI+969): Alt = 181,000 feet [modeling] Mach = 15 [modeling] KEAS = 228 [modeling] Qbar = 83 psf [modeling]
		DEPRESSURIZATION BEGINS – NET Time. This is the first event of lethal potential.
		Loss of consciousness and cessation of respiration.
14:00:19 (EI+970)		The MADS/OEX recorder stopped recording at GMT 14:00:19.44 (EI+970.44).[34]

[33] This symbol is used to indicate a vehicle-related event that occurred after the separation of the crew module and forward fuselage from the midbody.
[34] *Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 73.

TIME	EVENT
14:00:25 (EI+976)	First visual indication that the orbiter had broken into multiple pieces (figure 1.2-46).

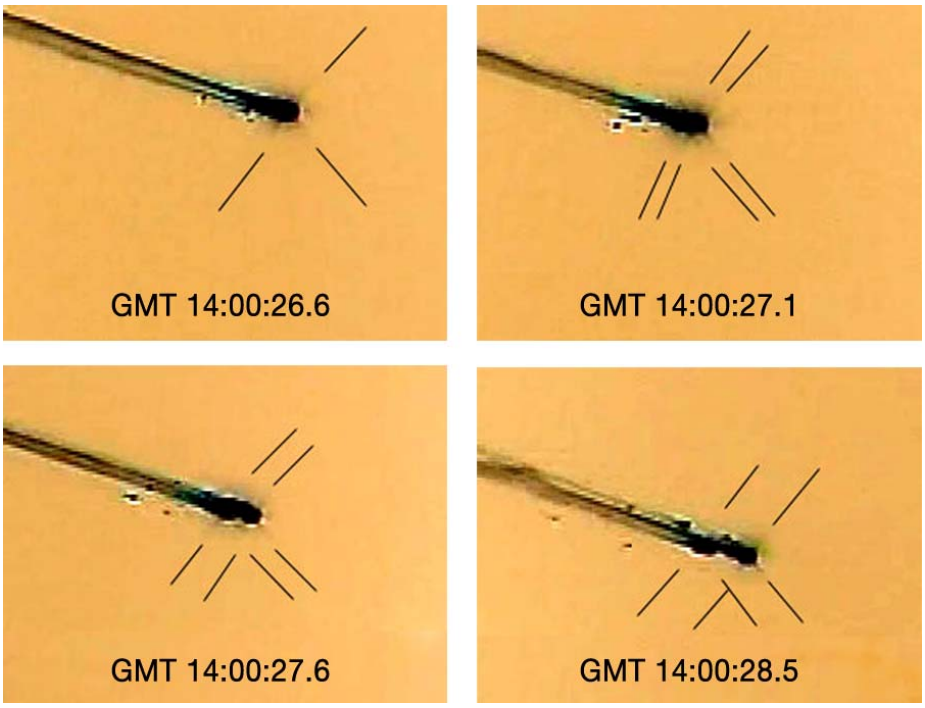


Figure 1.2-46. ***Color-inverted video images of the start of the double star event, GMT 14:00:26.6 (EI+977.6).*** **Black lines have been added to more clearly identify the two separate objects.**

14:00:35 (EI+986) ±5 sec	**DEPRESSURIZATION BEGINS – NLT Time.** This time is based on the release times for items from a storage compartment within the crew module.

1.2.5 Phase 5: Crew Module Catastrophic Event to Total Dispersal
[GMT 14:00:53 (EI+1004) through GMT 14:01:10 (EI+1021)]

14:00:53 (EI+1004)	**CREW MODULE CATASTROPHIC EVENT Begins.** The CMCE is the initiation of a period of time during which the forebody was undergoing a major structural breakup. The breakup sequence progressed over several seconds and ends at approximately GMT 14:01:10 (EI+1021). Analysis of ground-based video of the event established the first detectable signs at GMT 14:00:53 (EI+1004). Based on engineering analysis, CMCE is thought to have started with the failure of the forward fuselage. Once the forward fuselage began to break away, the exposed crew module rapidly failed due to the combined effects of the high G-loads, aerodynamic forces, and thermal loads. The flight deck maintained structural integrity longer than the middeck (figure 1.2-47).

TIME		EVENT

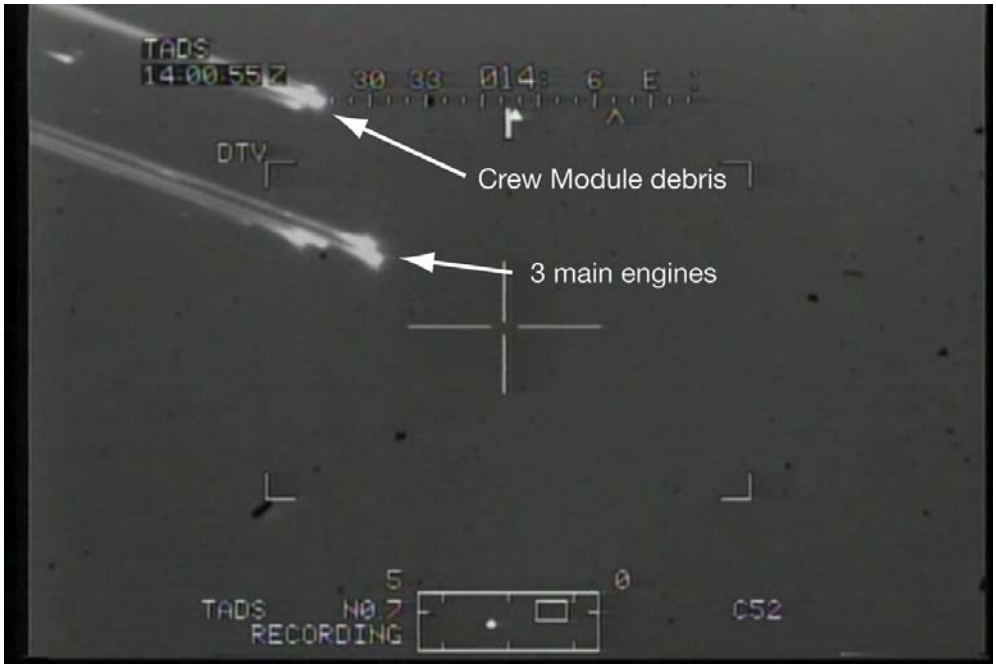


Figure 1.2-47. ***Image from Apache[35] video showing crew module debris*** **(top cluster)** ***and the three main engines*** **(cluster to left of cross hairs).**

		Thermal intrusion into the crew module – NLT time.
		Inertial reel straps fail – Metallic deposits on the inertial reel straps indicate that globules of molten metal were present in the crew module prior to the straps failing and retracting into the inertial reel housing.
14:00:57 (EI+1008)±5 sec		Based on ballistic analysis, a recovered piece of the forward RCS separated at this time.
14:00:59 (EI+1010)		**DEPRESSURIZATION COMPLETE – NLT Time.** Based on video evidence, the crew module no longer had sufficient structural integrity to maintain cabin pressure. However, the cabin depressurization was probably complete well before this time.
14:01:01 (EI+1012)±5 sec		**Middeck CEE Released.** Based on ballistic analysis, the earliest piece of recovered CEE from the middeck crew members separated at this time.
14:01:07 (EI+1018)±5 sec		**Middeck Accommodation Rack (MAR) Separates.** Based on ballistic analysis, the MAR, which was located on the port side of the middeck, separated at this time.

[35]The "Apache" video was filmed from an Apache helicopter.

TIME		EVENT
14:01:08 (EI+1019)±5 sec		**Flight Deck CEE Released.** Based on ballistic analysis, the earliest piece of recovered CEE from the flight deck crew members separated at this time.

1.2.6 Phase 6: Total Dispersal to ground impact
[GMT 14:01:10 (EI+1021) through approximately GMT 14:35:00 (EI+3051)]

14:01:10 (EI+1021) — **TD** – Forward fuselage and crew module have fragmented into pieces too small to be detected on any of the ground-based videos (figure 1.2-48).

Figure 1.2-48. *Image is from the Apache video*. Dotted circle indicates area where crew module debris is last visible. The three points in the lower right are the three main engines.

14:35:00 (EI+3051) — Approximate time at which the crew remains and the majority of the crew module debris completed the free fall to the ground.

Chapter 2 – Vehicle Failure Assessment

2.1 *Motion and Thermal Analyses*

2.2 *Orbiter Breakup Sequence*

2.3 *Crew Cabin Pressure Environment Analysis*

2.4 *Forebody Breakup Sequence*

2.1 Motion and Thermal Analyses

This section discusses the trajectory, dynamic attitude, and thermal analyses of the *Columbia* accident performed in support of the Spacecraft Crew Survival Integration Investigation Team (SCSIIT). The motion analyses provided information about the loads experienced by the crew and the vehicle through the evolving conditions. Because the thermal environment for atmospheric entry is highly dependent on the aerodynamic properties of an object as well as the aerodynamic conditions such as altitude and velocity, the aerodynamic analyses also provided reference data for thermal analyses. The purpose of the thermal analyses was to aid in understanding the probable failure sequence by comparing the physical condition of certain key items of debris with predicted entry heating.

"Trajectory" refers to the translational motion of an object relative to the Earth, and provides information about the deceleration of an object due to atmospheric drag. The trajectory of the intact orbiter after communications were lost was estimated from available aerodynamic data. However, this type of data did not exist for the forebody as a separate object following orbiter breakup at the Catastrophic Event (CE). An average ballistic number was approximated[1] for the forebody. A 3 degree-of-freedom trajectory simulation used this approximate ballistic number to estimate the trajectory of the forebody. The "reference trajectory" was a continuous trajectory used for ballistic and thermal analyses. The reference trajectory was a sequential combination of the trajectory of the intact orbiter followed by the trajectory of the forebody after separation from the orbiter.

"Dynamic attitude motion" refers to the rotational motion of an object relative to the trajectory. This information was necessary since aerodynamic drag was the predominant load on the vehicle. Because drag always acts in a direction opposite to a vehicle's forward motion, a rotating vehicle will experience the drag force in varying directions relative to the body axes. Simulations were developed to characterize the dynamic attitude motion of the intact orbiter and the forebody. The first simulation modeled estimated dynamic attitude for the intact orbiter from loss of signal (LOS) to the CE. This simulation incorporated telemetry and the Modular Auxiliary Data System/orbiter experiment (MADS/OEX) recorder data into an existing orbiter aerodynamic model. The second simulation modeled the estimated dynamic attitude of the free-flying forebody from the CE to the Crew Module Catastrophic Event (CMCE). This simulation had no direct data available. The aerodynamic characteristics of the forebody were estimated and incorporated into an aerodynamic model. The rotational loads and attitudes were then combined with the drag loads identified through the trajectory analysis to estimate loads in all axes for both the intact orbiter and the forebody.

The data from these model-based analyses should be considered only as *representative* of the type of motion the vehicle and crew experienced. The results of these analyses were compared to a relative motion analysis obtained from ground-based video. The analyses were found to be in general agreement.

To better quantify the limits of crew survival for future flights, the survivability of an exposed crew module (CM) following orbiter breakup for ascent and entry conditions were examined from a thermal perspective. The results were compared to the estimates currently used in crew procedures and to the conditions experienced for both the *Challenger* and the *Columbia* accidents. Thermal analyses were also completed on individual recovered items to predict the amount of heating that they would have experienced given a specific

[1]Ballistic number is an indicator of the performance of an un-powered object flying through the atmosphere. It is characterized by the ratio of the object's weight over its aerodynamic drag.

release time. These results were compared to the thermal damage observed on debris items to confirm ballistic release times and to help sequence events.

The following is a summary of findings, conclusions and recommendations in this section:

Conclusion L3-1. Complete loss of hydraulic pressure to the aerosurfaces resulting from the breach in the left wing was the probable proximal cause for the vehicle loss of control.

Recommendation L3-2. Future vehicles should be designed with a separation of critical functions to the maximum extent possible and robust protection for individual functional components when separation is not practical.

Finding. The forebody and the CM of the orbiter are not aerodynamically stable in attitude with any initial rates or lateral center of gravity (c.g.) other than zero.

Finding. Triangulation results suggest that the free-flying forebody rotated at approximately 0.1 rev/sec in a multi-axis motion.

Conclusion L2-1. Between orbiter breakup and the forebody breakup, the free-flying forebody was rotating about all three axes at approximately 0.1 rev/sec and did not trim into a specific attitude.

Finding. The estimate for maximum thermal survivability on ascent of 280,000 feet is a reasonable estimate.

Finding. The maximum thermal survivable breakup altitude for the CM on entry is approximately 150,000 feet.

Conclusion L3-3. The actual maximum survivable altitude for a breakup of the space shuttle is not known.

Recommendation L2-2. Prior to operational deployment of future crewed spacecraft, determine the vehicle dynamics, entry thermal and aerodynamic loads, and crew survival envelopes during a vehicle loss of control so that they may be adequately integrated into training programs.

Recommendation L3-3. Future spacecraft design should incorporate crashworthy, locatable data recorders for accident/incident flight reconstruction.

Finding. Thermal analyses predicted that entry aeroheating alone was insufficient by an order of magnitude to produce the observed thermal damage on the x-links. Therefore, the x-links must have experienced other heating mechanism(s) in addition to normal entry heating.

Conclusion A13-1. Titanium may oxidize and combust in entry heating conditions dependent on enthalpy, pressure, and geometry.

Conclusion A13-2. The heating from a Type IV shock-shock impingement and titanium combustion (in some combination) likely resulted in the damage seen by the forward payload bay door rollers and the x-links.

Recommendation A13. Studies should be performed to further characterize the material behavior of titanium in entry environments to better understand optimal space applications of this material.

2.1.1 Intact orbiter trajectory

Figures 2.1-1 through 2.1-3 define the body axis coordinate system and the attitude angles that will be discussed in this section. Figure 2.1-1 defines the body axis coordinate system (X_{BY}, Y_{BY}, Z_{BY}) and the pitch, yaw, and roll angles. Figure 2.1-2 defines the angle of attack (α), and figure 2.1-3 defines sideslip (β). These angles are defined relative to the X_{BY} axis and the velocity vector (V). Proj(V) is the projection of the velocity vector (line of sight along the trajectory).

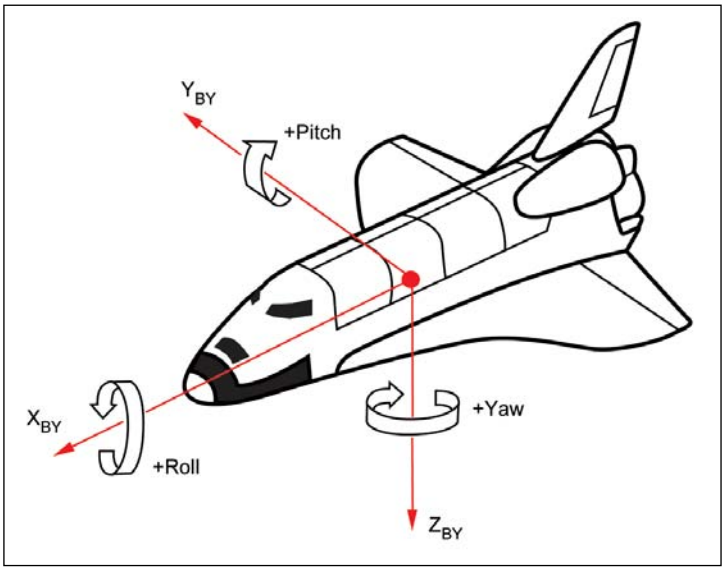


Figure 2.1-1. *Depiction of the body axis coordinate system and pitch, yaw, and roll definitions*.

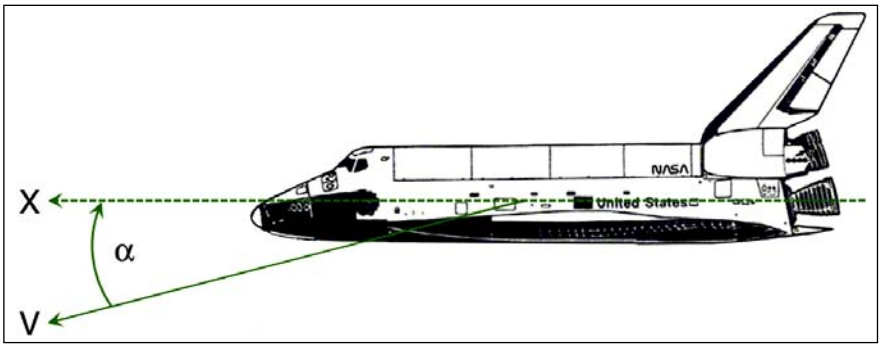


Figure 2.1-2. *Depiction of angle of attack*.

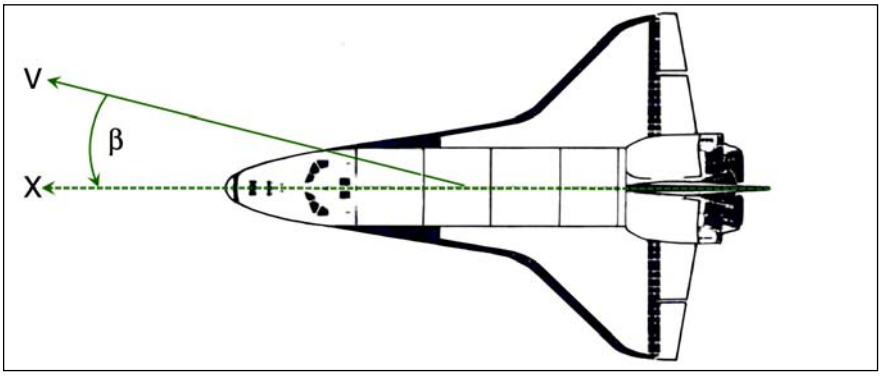


Figure 2.1-3. *Depiction of sideslip*.

Nominal dynamic, 3-axis coupled orbiter aerodynamic stability has been discussed in many documents and will not be discussed here. There are many documents available that discuss the orbiter's dynamic stability.[2,3,4] The aerodynamic data used in this aerodynamic analysis were obtained from the Orbiter Aerodynamic Databook.[5] The data in the Databook are limited to the operational range of the orbiter. After loss of control (LOC), the orbiter was probably out of that range in some parameters. However, no other data exists, and the Databook was the best source available. Engineering judgment was used to determine which data applied best to the conditions.

Some assumptions were necessary to provide a continuous reference trajectory for ballistic and thermal analyses. Four different sets of condition assumptions were used sequentially to determine the reference trajectory. To help clarify the differences for each of these phases, some discussion follows of these conditions and how they affected the trajectory.

An uncontrolled object entering the Earth's atmosphere is on a ballistic trajectory (see Ballistic Tutorial, Appendix). However, a vehicle may use aerodynamic controls and vehicle properties to generate lift, which modulates drag, to alter the ballistic number and, hence, the trajectory. The orbiter can generate lift when in the appropriate attitude. A simplified, static, 3-axis uncoupled analysis of aerodynamic stability shows that when in the high-altitude hypersonic flight regime, the orbiter is not statically stable in attitude. Active control is necessary to maintain orbiter stability. This active control is provided by the general purpose computers (GPCs) through the digital autopilot (DAP). The DAP uses the Flight Control System (FCS), a blend of Reaction Control System (RCS) jets and the elevons, body flap, and (at lower Mach numbers) rudder/speed brake aerosurfaces to control the orbiter attitude. As the orbiter descends into the atmosphere, the aerosurfaces have more control authority and the RCS jets play a lesser role in control.

The orbiter uses a drag-velocity profile to reduce its velocity, maintain heating and aerodynamic loads within orbiter limits, and reach the landing site. Figure 2.1-4 shows the entry drag-velocity profile and boundaries. The lower lift boundary is called the equilibrium glide boundary. This is typically defined as having 20% of vehicle lift remaining to adjust the trajectory, which is accomplished by modulating the angle of attack and bank angle. Bank angle also changes the trajectory laterally, so periodic roll reversals are accomplished to keep the orbiter from drifting too far crossrange.

[2]William T. Suit, "Summary of Longitudinal Stability and Control Parameters as Determined from Space Shuttle Columbia Flight Test Data," NASA Technical Memorandum 87768, August 1986.
[3]Robert Blanchard, Kevin Larman, Christina Moats, "Rarefied-Flow Shuttle Aerodynamics Flight Model," NASA Technical Memorandum 107698, February 1993.
[4]Robert Day, "Coupling Dynamics in Aircraft: A Historical Perspective," NASA Special Publication 532, 1997.
[5]Orbiter Aerodynamic Databook, STS85-0118, Volumes 1, 2, and 5, August 2001, Volume 3, February 1996, Volume 4, January 1994, Volume 6, December 2000.

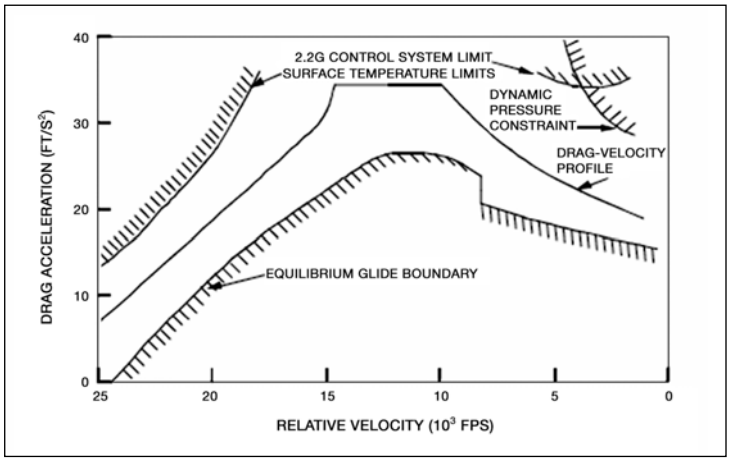


Figure 2.1-4. ***Entry guidance drag-velocity profile and limits.***[6]

Existing orbiter aerodynamic data were used to estimate the ballistic number for the first phase. This phase of the trajectory was the "Nominal Orbiter Phase" with an intact, controlled orbiter ballistic number of 108 pounds per square foot (psf). This reflected conditions when signal was lost at Greenwich Mean Time (GMT) 13:59:32.

At some point after LOS, the orbiter lost attitude control. The degradation of the left wing appeared to be the most obvious cause. The Integrated Entry Environment Team pursued a more detailed investigation.

As mentioned above, the drag-velocity profile is controlled by the orbiter's angle of attack and bank angle. Increasing the bank angle increases the drag slowly, while increasing the angle of attack increases the drag quickly. When the reference drag profile computed by the on-board guidance is below the equilibrium glide boundary, a "ROLL REF" alarm is annunciated to the crew. During STS-107, this alarm was annunciated at GMT 13:59:46, only 9 seconds after the LOS.

To determine which parameter (bank angle or angle of attack) caused the ROLL REF alarm, the Integrated Entry Environment Team ran simulations varying each parameter.[7] The thermal degradation was causing increased drag on the left wing, which would induce increasing yaw and/or roll. However, the parameters driving the ROLL REF message proved to be much more sensitive to angle-of-attack deviations than bank angle deviations. Since LOS was at GMT 13:59:37, only 9 seconds passed before the message was annunciated. Based on these simulations, the Integrated Entry Environment Team concluded that it was most likely that a large angle-of-attack change (rather than bank angle) triggered the ROLL REF alarm, although the actual rate of change of the angle of attack could not be determined.

Reconstructed general purpose computer (RGPC) data from GMT 14:00:02.6 (referred to as RGPC-2) show all three hydraulic system pressures at zero, and the hinge moments of the aerosurfaces in the "up" position. Loss of hydraulic power would cause the pitch, roll, and yaw aerosurfaces to float. Pitch and roll RCS jets are no longer used for control at this Mach number and altitude, while yaw jets are still incorporated into control logic and could continue to provide some control in this axis.

Significantly, ground-based video shows a marked change in the appearance of the orbiter's trail at GMT 13:59:37 (immediately after the end of the first RGPC data set, RGPC-1) (figure 2.1-5). The width of the trail increases at this time, which likely indicates a change in the orbiter's flight condition. In addition to the change in the width of the trail, the trail appears to pulse or "corkscrew" over a period of less than 1 second. It is possible that a large debris event (such as loss of a major portion of the wing) may have caused

[6]Space Shuttle Orbiter Operational Level C Functional Subsystem Software Requirements; Guidance, Navigation and Control; Part A; Entry Through Landing Guidance.
[7]EG-DIV-08-32 – Integrated Entry Team Report Appendix G - Post-LOS Analysis.

the LOC. However, the debris event closest to the LOS that was apparent in the video occurred 2 seconds, or more than 60 frames (figure 2.1-6), after the change in the trail's appearance. This suggests that it was subsequent to the LOC and was not the cause. Video time errors are up to 1 second, but the relative sequence of the events (brightening, followed by a debris event) is not changed.

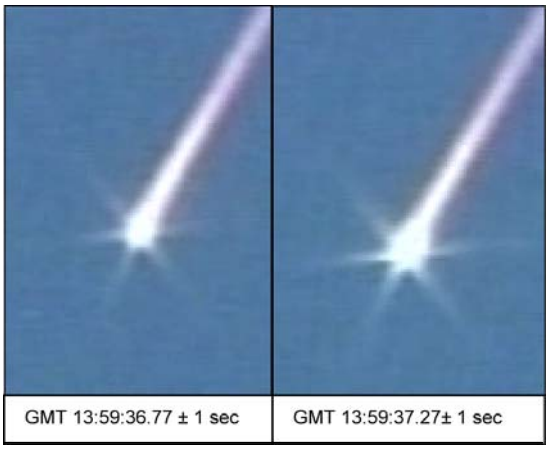


Figure 2.1-5. ***Noticeable change in brightness at GMT 13:59:37±1 second.***

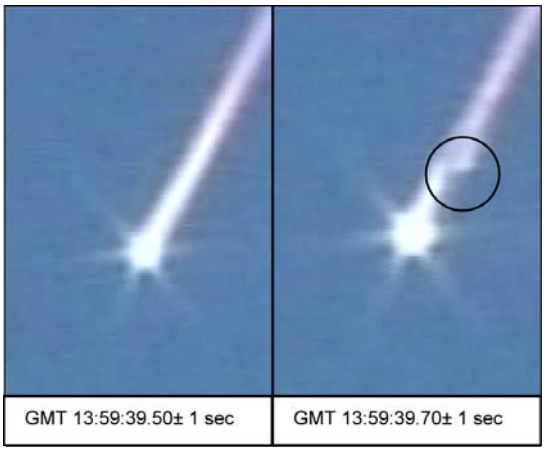


Figure 2.1-6. ***Noticeable debris shedding event*** **(circled at right)** ***at GMT 13:59:39±1 second.***

It was concluded that the most credible scenario for LOC was the loss of all hydraulic systems, causing the aerosurfaces to float and resulting in an uncontrolled pitch-up. Since the left wing contains locations where all three hydraulic systems have lines that are in close proximity (including the left gear well), it is probable that the loss of hydraulics was due to thermal damage as a result of the breach in the wing. Consequently, the SCSIIT defined the vehicle LOC to begin at GMT 13:59:37, immediately after the LOS, when it was determined by inference that hydraulic power to the aerosurfaces was lost. Prior to this time, *Columbia*'s DAP was still in control of the orbiter. The FCS was commanding the elevons and the RCS yaw jets to counteract the increasing drag on the damaged left wing. The LOC marks the time when the *ability* to control the vehicle was lost, not when *Columbia* departed known attitudes (which occurred shortly thereafter).

Separation of redundant features is an important element in survivability design.[8] It should be noted, however, that with the RCS jets continually firing, propellant would have been rapidly exhausted, causing an inevitable LOC regardless of the condition of the hydraulic systems.

Conclusion L3-1. Complete loss of hydraulic pressure to the aerosurfaces resulting from the breach in the left wing was the probable proximal cause for the vehicle loss of control.

Recommendation L3-2. Future vehicles should be designed with a separation of critical functions to the maximum extent possible and robust protection for individual functional components when separation is not practical.

Figure 2.1-7 shows the increasing angle-of-attack simulation. The figure shows the orbiter attitude at 1-second intervals, beginning at GMT 13:59:37 and ending at GMT 13:59:46. The white line in the figure is the reference trajectory.

[8]Ball, Robert E., *The Fundamentals of Aircraft Combat Survivability Analysis and Design*, Second Edition, AIAA, Reston, VA, 2003 (The AIAA Textbook); http://www.aircraft-survivability.com/pages/books_frame.html.

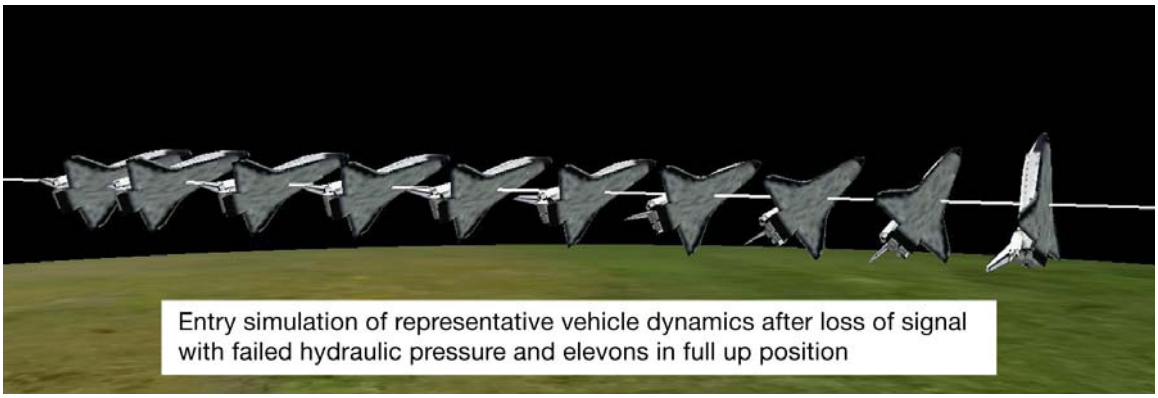


Figure 2.1-7. ***Sequence (1-second intervals) showing a simulation of orbiter loss of control pitch-up from GMT 13:59:37 to GMT 13:59:46.*** **White line indicates vehicle trajectory relative to the ground.**

Following LOC, the ability of the orbiter to generate lift was greatly reduced due to the uncontrolled changes in attitude. These changes in attitude also affected ballistic number, which is dependent on the cross-section of an object presented to the drag vector. Consequently, the ballistic number for the orbiter after LOC had to be estimated. This estimate was obtained by averaging ballistic numbers from a variety of attitudes.

The periodic loss of structure as the orbiter shed debris also affected the mass properties and the cross-section of the orbiter. Debris shedding events were not understood enough to model, so for the purposes of this assessment mass properties were held constant.

This second phase of the trajectory was termed the "High Drag Orbiter Phase." The average ballistic number for the conditions between LOC and the last available GPC data (start of RGPC- 2, GMT 14:00:02.6) was estimated to be 41.7 psf. The motion of the orbiter was assumed to be benign enough that some lift was still generated, and this lift was incorporated into the trajectory. The drop in ballistic number as the orbiter changed attitude from the nominal orbiter ballistic number of 108 psf to a high drag condition of 41.7 psf would increase the deceleration of the orbiter from the nominal deceleration profile.

At the beginning of RGPC-2 and before the CE (orbiter breakup), the average ballistic number for this phase was still estimated to be 41.7 psf. However, the orbiter was assumed to be generating no lift, and only drag was incorporated into the trajectory. For this "No-lift Orbiter Phase," it was predicted from simulations that the orbiter's angle of attack was varying from 30 to 120 degrees and the damaged left wing was not producing lift. The angle of attack assumed for this phase of the simulation was 72 degrees because it was a mid-value angle of attack and closely corresponded with the last angle of attack data value that was recovered.

2.1.2 Forebody trajectory

The forebody trajectory analysis was initially performed by the Crew Survival Working Group (CSWG) and later updated by the SCSIIT. Once again, the assumption was made that mass property changes were negligible and the ballistic number was averaged across a range of attitudes.

The "Forebody Phase" begins at GMT 14:00:23, which was an early assessment time of orbiter breakup during the initial investigation. The average ballistic number of 150 psf for the detached forebody was estimated by Johnson Space Center (JSC) Engineering immediately after the *Columbia* accident. This estimate was used in a ballistic analysis to generate debris search areas. This analysis was provided to the *Columbia* Accident Investigation Board (CAIB) and published in the CAIB Report.

Using more detailed information from ground-based video, the Image Science and Analysis Group (ISAG) determined that the orbiter breakup occurred at GMT 14:00:18.3.[9] Later, higher-fidelity mass properties of the CM based on flight data were combined with a refined estimate of the geometry of the forebody to provide a more precise average ballistic number of 122 psf. Because ballistic analysis on many objects had already been conducted using the 150-psf reference trajectory, it was preferable not to change to the updated CE time and 122-psf reference trajectory for the additional ballistic analyses. A sensitivity study was conducted to determine how changing the reference trajectory average ballistic number from 150 psf to 122 psf would affect the release times. It was found that the release times only changed by 1 second. Since the overall error in ballistic release times was ±5 seconds, this is not significant. Therefore, the original 150-psf forebody reference trajectory was used for the additional ballistic analyses. Figure 2.1-8 shows the two forebody reference trajectories. The dotted lines are the 122-psf trajectory parameters and the solid lines are the 150-psf trajectory parameters for an idealized forebody trajectory. The plot shows the differences in the CE times, loads, heat rates, altitudes, and equivalent velocity between the two trajectories. The trajectories were propagated out even past the CMCE, the time at which the forebody broke up, to enhance the evaluation of the differences.

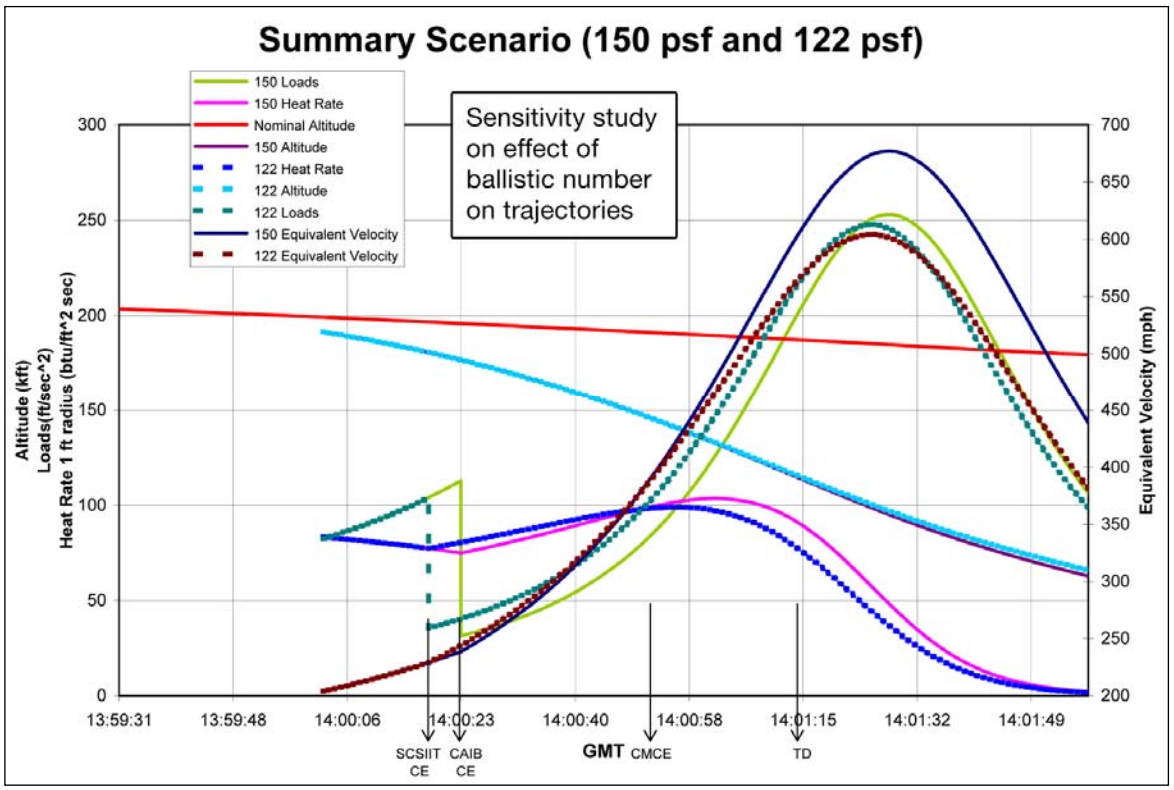


Figure 2.1-8. ***Comparison of 150-pounds-per-square-foot and 122-pounds-per-square-foot forebody projected trajectories*****.**

The significance of this sudden change in ballistic number can also be seen in figure 2.1-8. As the ballistic number went from 41.7 psf to 150 psf, the deceleration dropped from approximately 3.3 G to approximately 1 G as experienced by the forebody. Figure 2.1-9 shows a more expanded view of this change in deceleration.

[9]*Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, October 2003.

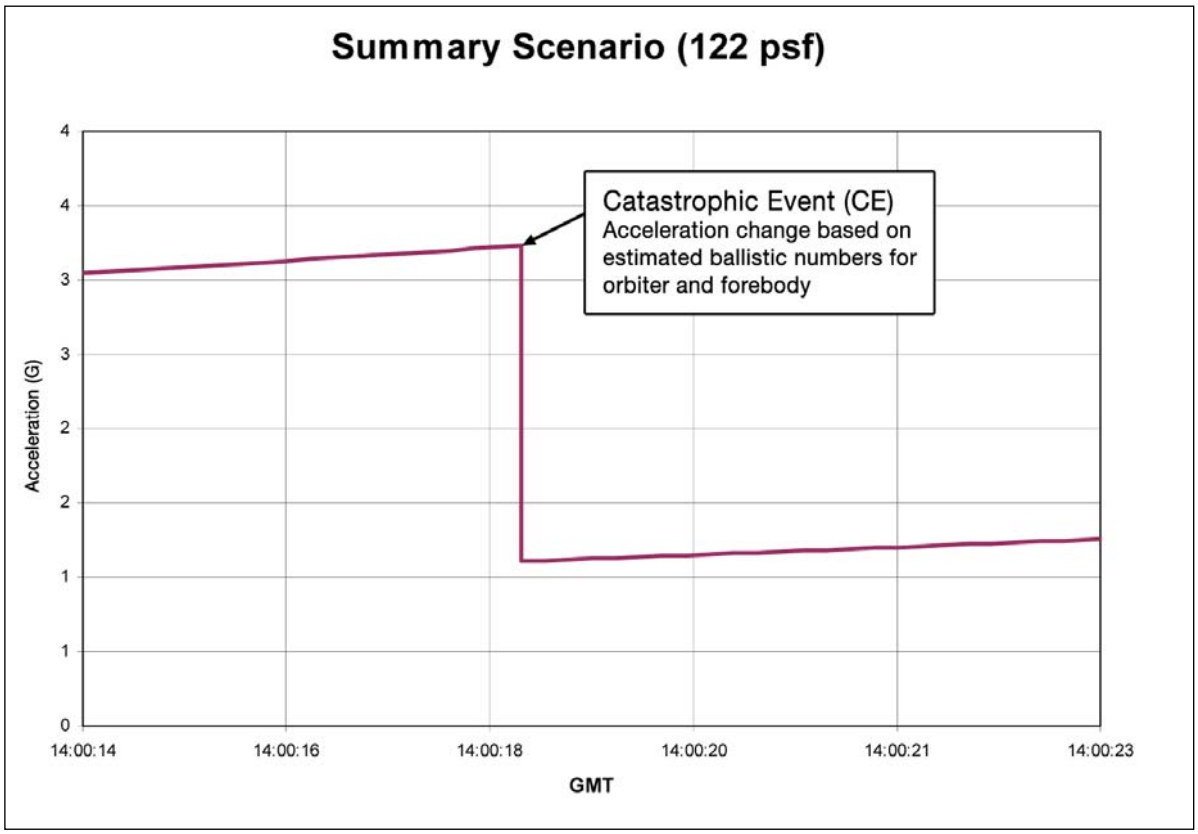


Figure 2.1-9. ***Estimated change in total G experienced by the forebody due to a change in ballistic number from the orbiter breakup.***

In summary, the reference trajectory is divided into the following four phases:

1. "Nominal Orbiter Phase," with an average ballistic number of 108 psf and aerodynamic data recovered from the vehicle. This phase starts 10 seconds before entry interface at 400,000 feet and lasts until LOC (GMT 13:43:59 to GMT 13:59:37).

2. "High Drag Orbiter Phase," with an average ballistic number of 41.7 psf and incorporating some lift generation. This phase starts at LOC and lasts until the beginning of RGPC-2 (GMT 13:59:37 to GMT 14:00:02).

3. "No-lift Orbiter Phase," with an average ballistic number of 41.7 psf and no lift generation (72-degree angle of attack assumed). This phase begins at the start of RGPC-2 and ends shortly after the beginning of the orbiter breakup at the CE (GMT 14:00:02 to GMT 14:00:23).

4. "Forebody Phase," with an average ballistic number of 150 psf. This phase begins shortly after the CE and ends at the forebody breakup (GMT 14:00:23 to GMT 14:00:53).

2.1.3 Intact orbiter attitude dynamics through the Catastrophic Event

An entry simulation was developed to estimate the attitude dynamics of the orbiter and the resulting accelerations on the vehicle structure and crew from LOS to the CE. When the orbiter was designed, there were no requirements to assess catastrophic scenarios to understand how the orbiter might behave in an out-of-control scenario. Test data were available to verify aerodynamic coefficients for the orbiter in nominal

conditions, but for this investigation existing models had to be extrapolated beyond flight experience to understand the probable motion of the orbiter.

Recommendation L2-2. Prior to operational deployment of future crewed spacecraft, determine the vehicle dynamics, entry thermal and aerodynamic loads, and crew survival envelopes during a vehicle loss of control so that they may be adequately integrated into training programs.

The simulation used the vehicle's preflight predicted mass properties. Changes in the c.g., moments of inertia, and mass due to the thermal damage were unknown and, therefore, not included. The aerodynamic characteristics determined for attitudes that were outside of the vehicle database were also based on the "undamaged" geometry. The initial conditions of the simulation were based on the downlinked GPC data at LOS including position, velocity, attitude, any alarm/warning-related data; and the recovered MADS/OEX recorder sensor data. The first set of RGPC data (RGPC-1) and the second set of RGPC data (RGPC-2) were used in an attempt to synchronize the simulation with the actual flight conditions. These RGPC data indicated that all hydraulic systems had failed by this point, which would result in the elevons free-floating, probably in the full-up position. The yaw rates during the RGPC-2 period exceeded the maximum value of the data scale ("pegged out"). All of the rates were high enough that they may have affected the inertial measurement units' ability to maintain a reference, so the quality of acceleration, rate, or attitude data is unclear. However, the period of excessive rates was relatively short and it was the only data available. The MADS/OEX recorder pressure data from the lower right wing surface pressure transducers indicate large oscillations in the angle of attack 5 seconds after LOS. These MADS/OEX recorder pressure data were matched to the simulation and showed good agreement in terms of rates and attitude excursions. Even though this entry simulation used all the available data, damage to the wing and aerodynamic properties outside the database of experience could not be accounted for in this analysis. However, the results are representative of the attitudes, rates, and characteristic motion the vehicle probably experienced.

Figures 2.1-10 through 2.1-12 show the plots of the estimated orbiter angle of attack, roll angle, and sideslip angle from the start of the LOC period until the CE based on the simulation. The representative motion seen in these plots shows that the orbiter was oscillating around the velocity vector in all three axes.

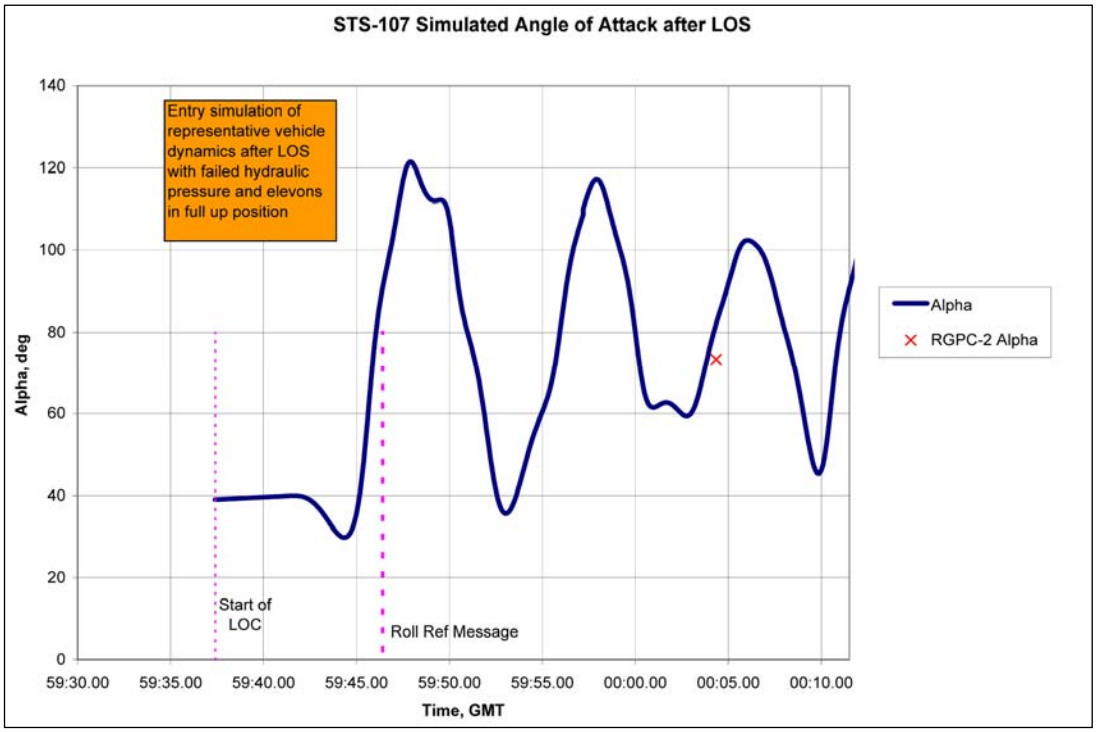


Figure 2.1-10. *STS-107 simulated angle of attack after loss of signal*. Red data indicate alpha condition recorded in the reconstructed general purpose computer-2 data.

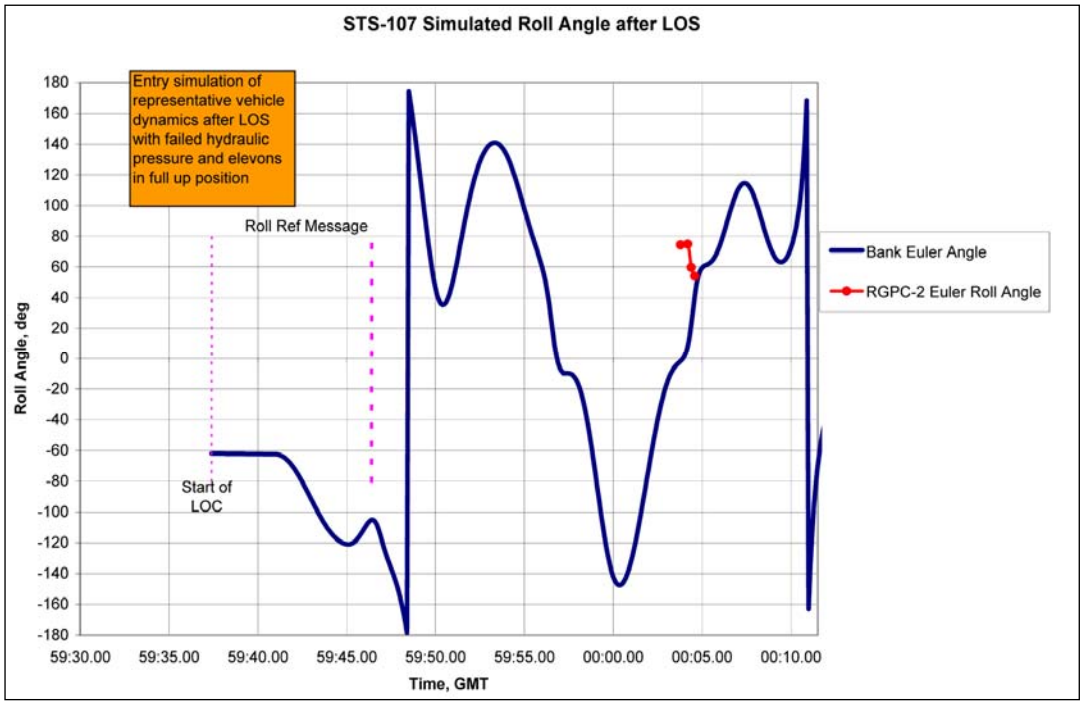


Figure 2.1-11. *STS-107 simulated roll angle after loss of signal*. Red data indicate roll conditions recorded in the reconstructed general purpose computer-2 data.

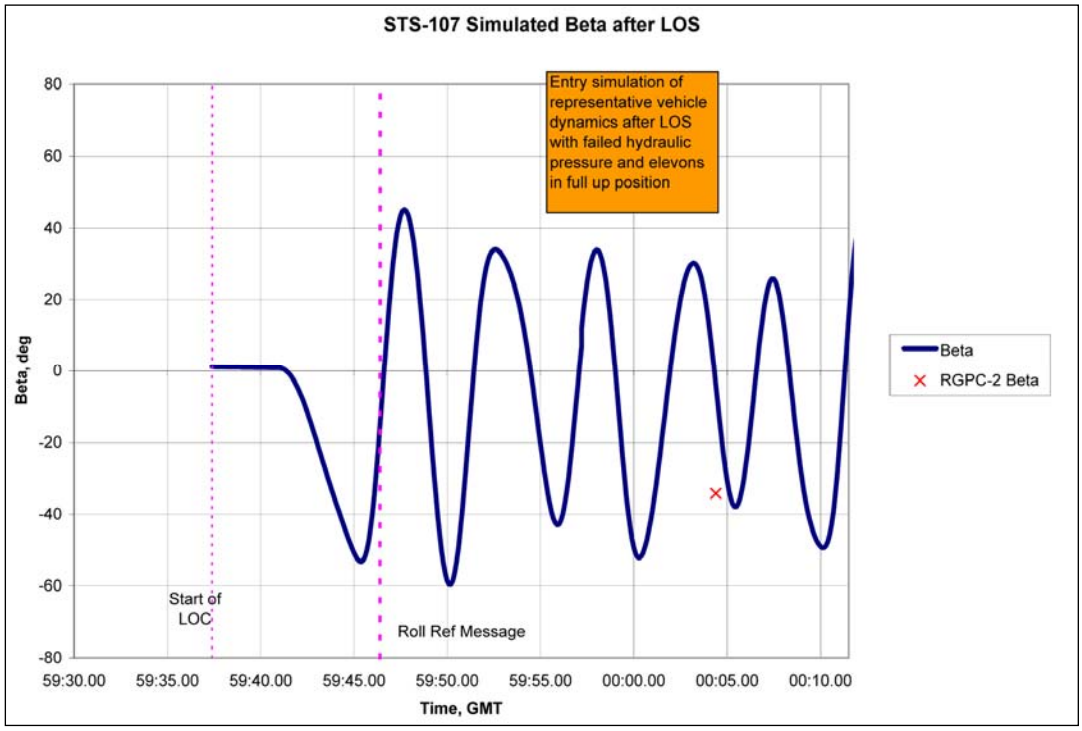


Figure 2.1-12. *STS-107 simulated sideslip angle after loss of signal*. Red data indicate beta condition recorded in the reconstructed general purpose computer-2 data.

Figure 2.1-13 is a second-by-second graphical depiction of the simulated motion of the orbiter during this period. The sequence begins at the start of the LOC period (GMT 13:59:37) and ends at the CE (GMT 14:00:18). The view is from a point in front of the orbiter's direction of travel, looking backward along the velocity vector. The snapshot for GMT 14:00:04 shows the left wing departing intact. In reality, the left wing did not come off all at one time but was shedding debris over a period of time.

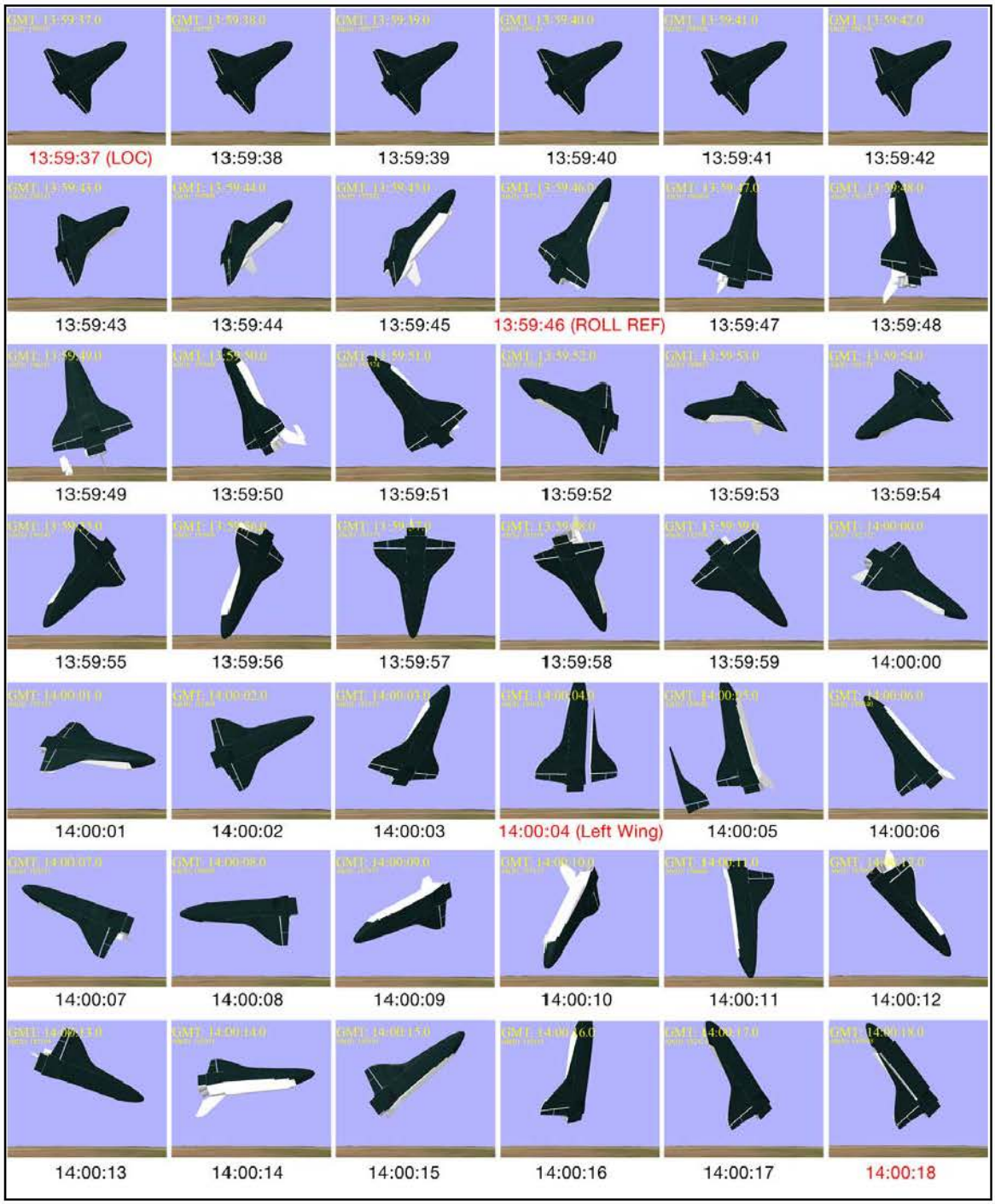
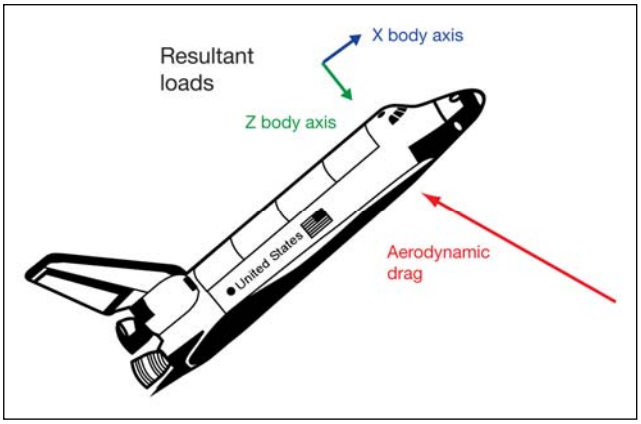


Figure 2.1-13. ***Entry simulation snapshot sequence*****. Times are in Greenwich Mean Time.**

With the sequence viewed from a vantage point looking back up the velocity vector, it is apparent that the predominant orientation of the orbiter remained "belly into the wind" with large excursions in pitch, roll, and yaw. This motion can be characterized[10] as a slow (30 to 40 degrees per second), highly oscillatory spin. As the simulation progresses in time, the rates and attitude excursions increase.

The entry simulation combined with trajectory data yielded the accelerations at the orbiter center of mass, the CM center of mass, and the x-links.[11] The total acceleration consists of the translational acceleration due to drag plus the rotational acceleration due to the rotation of the orbiter. The magnitude of the rotational acceleration is dependent on the distance from the center of rotation, and the direction of the load is always away from the center of rotation. The translational acceleration is always along the velocity vector as the orbiter decelerates due to aerodynamic drag. However, as the orbiter rotated, the direction of this deceleration load rotated with respect to the orbiter body axes. Sometimes the translational and rotational accelerations add together, other times they subtract from one another. The addition or subtraction of the translational and rotational accelerations depends upon the orientation of the orbiter at that particular time. Figures 2.1-14 and 2.1-15 show this concept.

Figure 2.1.-14. ***Representation of resultant body axis loads imparted from aerodynamic drag in nominal entry attitude.***



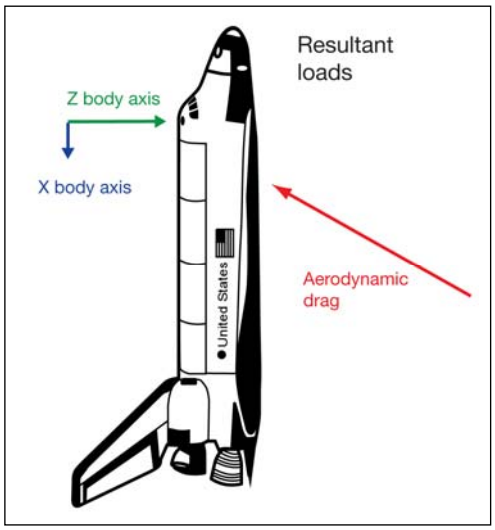


Figure 2.1-15. ***Representation of change in resultant body axis loads imparted from aerodynamic drag in off-nominal attitude.***

[10]Flight Test Demonstration Requirements for Departure Resistance and Post-Departure Characteristics of Piloted Airplanes, Air Force MIL-F-83691B, Change 1, 5/31/96.
[11]The x-links connect the crew module to the forward fuselage at the X_o 582 ring frame bulkhead.

The accelerations are shown in figures 2.1-16 through 2.1-18. These figures show the accelerations due to rotational motion of the orbiter plus the translational accelerations due to atmospheric drag. There is one figure for each orbiter axis, and each figure shows the accelerations at the three different vehicle locations – at the orbiter center of mass, at the CM center of mass, and at the x-link location. Included in each figure is a pictorial that helps describe the effective motion of crew members that would result from these accelerations.

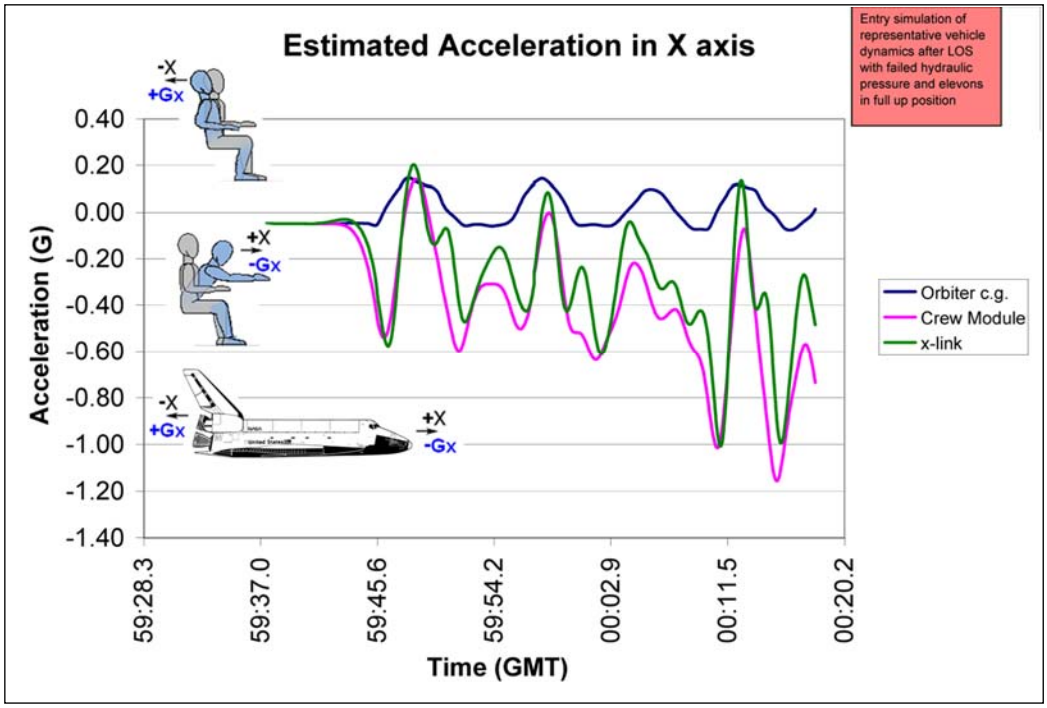


Figure 2.1-16. ***Estimated accelerations at the crew module in the X axis from the loss of signal to the Catastrophic Event*.**

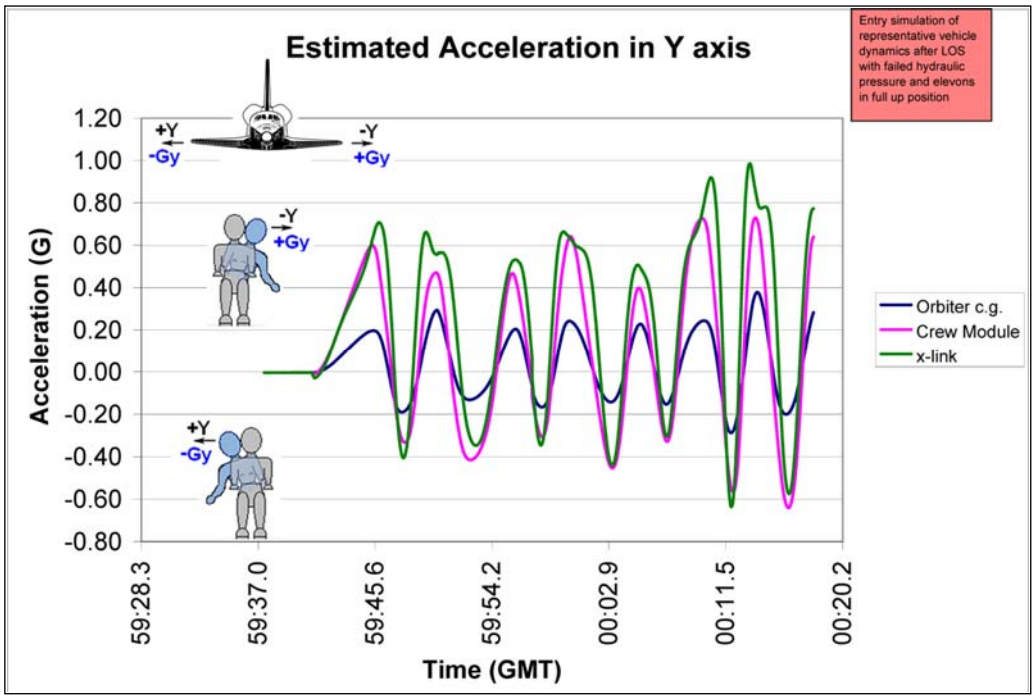


Figure 2.1-17. ***Estimated accelerations at the crew module in the Y axis from the loss of signal to the Catastrophic Event*.**

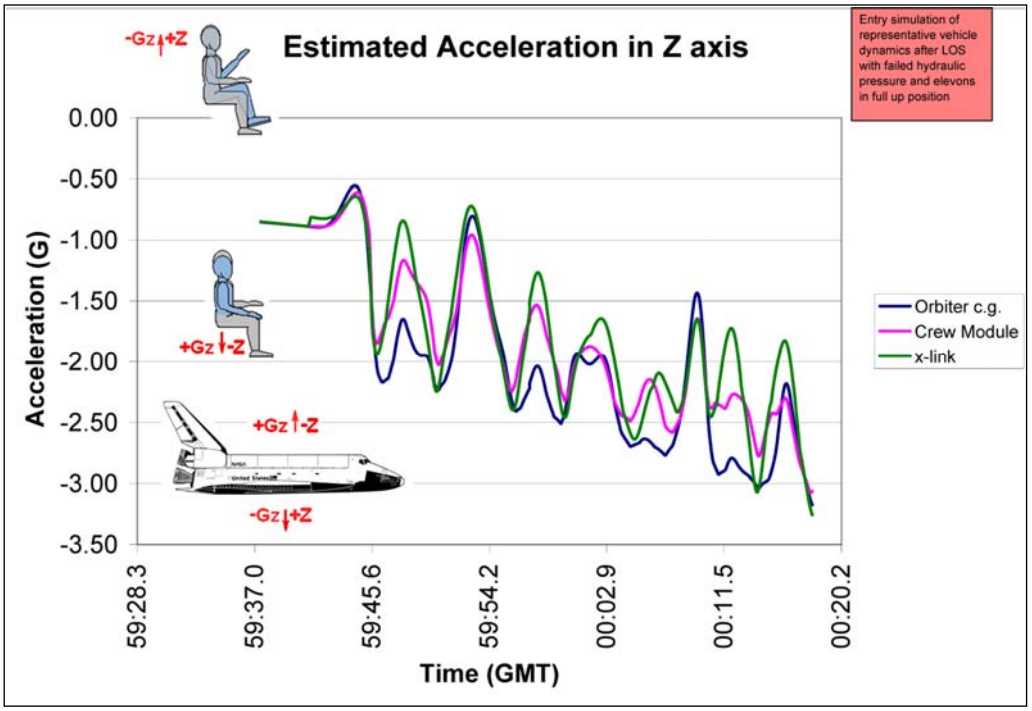


Figure 2.1-18. ***Estimated accelerations at the crew module in the orbiter Z axis from the loss of signal to the Catastrophic Event.*** **[Note: Crew and orbiter Z axes have opposite convention]**

As shown in the figures' pictorials, the motion experienced by the crew members inside the vehicle during the intact orbiter LOC time period is best represented by a swaying motion side-to-side, a pull forward in the seat, and a push down into the seat.

2.1.4 Post-Catastrophic Event attitude dynamics of the forebody

The SCSIIT received anecdotal information that the *Challenger* CM was believed to have reached a stable attitude ("trimmed") in a nose-into-the-wind condition. This belief was based on a video that shows a brief clear view of the CM in this orientation (figure 2.1-19), and the fact that the CM appeared to have impacted the water in a slightly nose-down condition.[12] As a result, the assumption upon which crew procedures are based (see Section 3.3) was that in an LOC/breakup scenario, the CM would trim nose-down after reaching terminal velocity. Although many types of data regarding the *Challenger* accident are available, they are almost entirely limited to information about the cause of the accident and are not related to crew survival. Relevant information was generally located by contacting personnel who worked at NASA at the time. No data could be found to support this anecdotal conclusion other than the video. Ground-based video of the forebody of Columbia implied that the forebody was rotating. Consequently, the SCSIIT set out to assess the assumption that the *Columbia* forebody came to an aerodynamically stable attitude.

[12]Report of the Presidential Commission on the *Challenger* Accident, Volume III, Appendix O, NASA Search, Recovery and Reconstruction Task Force Team Report.

Figure 2.1-19. ***Video-capture showing*** **Challenger** ***crew module*** **(circled in red)** ***pointed nose into the direction of travel*****.**

No telemetry, test data, or previous analysis existed to characterize the behavior of the CM or the forebody after separation from the rest of the orbiter. Therefore, the aerodynamic properties of the forebody had to be estimated and incorporated into an aerodynamic simulation to determine the forebody dynamics. The objective of this study was to examine the stability and possible trim attitudes of the free-flying CM and forebody [CM plus forward fuselage (FF) (figure 2.1-20)] from the CE to the CMCE. These data, when combined with the forebody trajectory aerodynamic drag data, provided the loads applied to the forebody during this phase. For simplicity, the initial conditions for altitude, velocity, and mass properties were held constant through out this 35 second period. The initial condition altitude was 176,790 feet with an initial velocity of Mach 14.4. These initial conditions were extrapolated from the last known vehicle state and projected through the simulation in the preceding section to the CE. The forebody mass properties were estimated using the evaluation of the orbiter breakup (see Section 2.2) and engineering judgment. This motion analysis was based solely on approximations and modeling. Therefore, the results should be considered indicators of general trends, and representative of the approximate accelerations and aerodynamic behavior that likely occurred.

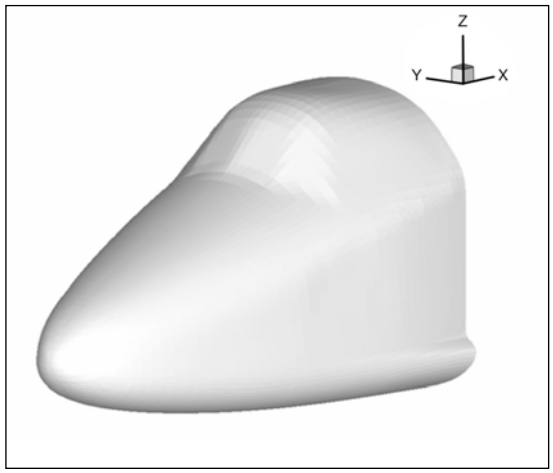


Figure 2.1-20. ***Forward fuselage/crew module (forebody) configuration*****.**

Before any simulations were run, the pitch stability of the CM and the forebody was examined. This was a static, 3-axis uncoupled analysis. Plots of the pitching moment coefficient, $C_{m,}$ vs. angle of attack showed two possible statically stable attitudes for the forebody at –100º and 125º angles of attack (CM stable attitudes were slightly different). Figure 2.1-21 shows the attitudes with the arrow indicating the direction of the wind.

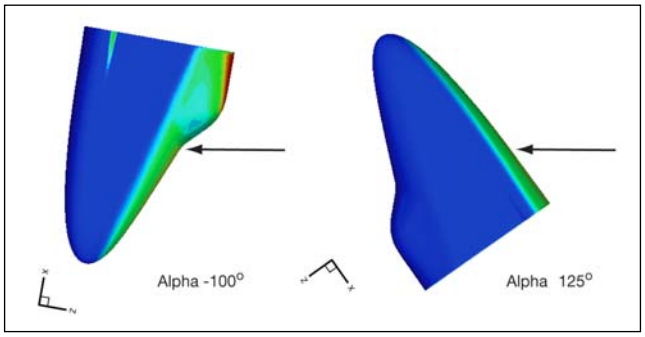


Figure 2.1-21. ***Pitch stable attitudes for the forebody*** **(arrow shows direction of wind).**

However, evaluation of the rolling and yawing moment coefficients, C_l and C_n showed that the statically stable attitudes in pitching moments (C_m) are not stable in either yaw or roll. Figure 2.1-22 shows the contour plot of all three moments across an alpha and a beta sweep. Locations where all three moments coincide would be stable in all three axes. Although at alpha approximately +50 degrees the moments appear very close to coinciding, close inspection showed that they do not.

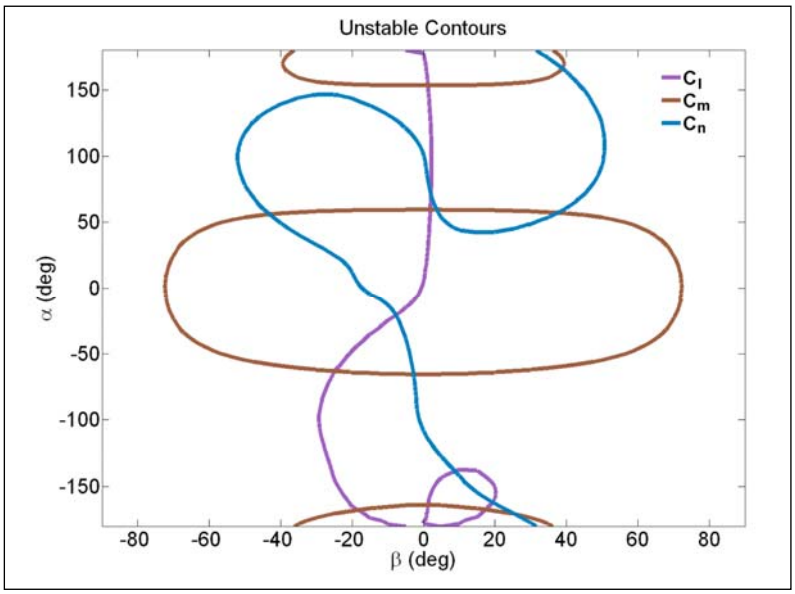


Figure 2.1-22. ***Contour plot of pitch, roll, and yaw moments for a crew module showing no stability due to slight offset in lateral center of gravity*.**

To verify this conclusion that the –100º and 125º angle-of-attack attitudes were not stable in the other axes, a forebody simulation 6-degree-of-freedom simulation was run. The initial attitude rates were zero, the initial angles of attack were set to the pitch trim points (–100º and 125º), and the sideslip angle was set to zero. In each case, after a few seconds the forebody began to rotate about the X axis with an increasing rate, and a "wobble" developed about the Y and Z axes. After evaluating the simulations, it was determined that the reason for the instability was a lateral offset of the c.g. The lateral c.g. limit for normal orbiter operations[13] ranges from ±1.0 in. to ±1.5 in. maximum depending on the X_{cg}. The lateral c.g. for *Columbia*'s CM was 0.7 in. from the centerline. This is well within operating limits for the orbiter, but greater than zero.

The simulation was re-run with the lateral c.g. set to zero. Both the –100º and 125º angle-of-attack cases were stable when starting with no initial rates. However, when the simulation was initiated with any initial rate in any axis, the forebody became unstable. The forebody never reached a stable attitude when initialized in any other attitude than one of the statically stable angles of attack. Since the orbiter was rotating at the time of breakup, it is expected that the free-flying forebody would also be rotating upon separation from the orbiter. It is concluded that the *Columbia* forebody did not attain a stable attitude.

The team also assessed whether a free-flying CM (with no forward RCS compartment or gear well attached) would come to a stable attitude. This is the *Challenger* case. The analysis was performed using the same tools. The assessment showed that the CM alone would not stabilize either.

Finally, it was speculated that the trailing wires and cables from the CM or forebody may have provided some aerodynamic stability (similar to the tail on a kite). The cables were modeled as 10 stiff poles trailing behind the CM with a diameter of 1.5 in. (based on maximum wire bundle size), a length of 10 feet (figure 2.1-23),

[13]Flight and Ground System Specification, NSTS-07700, Volume X, Book I, Section 3.3.1.2.1.5 Orbiter – CG limits, November 10, 1998, p. 3-142.

and a mass of 2.43 lbs. per foot based on a bundle of 715 strands of 22-gauge wire. The resultant simulation showed no significant effect on stability for either the CM or the forebody. This is most likely because of the low mass of the cables compared to the forebody.

Modeling is subject to some inherent uncertainty regarding the actual motion dynamics. However, all analyses consistently showed a lack of stability.

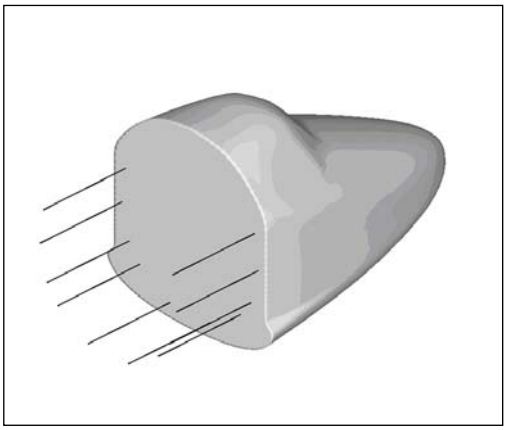


Figure 2.1-23. ***Modeled trailing wire configuration for forebody.***

Finding. The forebody and the CM of the orbiter are not aerodynamically stable in attitude with any initial rates or lateral center of gravity (c.g.) other than zero.

To determine the angular velocities and the accelerations at various locations in the forebody for the time period from the CE to the CMCE, two dynamic simulations were performed. Both cases used actual STS-107 c.g. conditions and an initial forebody attitude of $\alpha = 90$ degrees and $\beta = 10$ degrees. This attitude was extrapolated from the final attitude at the end of the LOC-to-CE simulation discussed earlier. Although the attitude of the forebody could be projected from the LOC simulation, the rates imparted to the forebody at the moment of separation cannot be determined. One case was run without any initial rotation rates, and one case was run with initial attitude rates. The initial attitude rates were extrapolated from the final rates from the LOC simulation and are –70 deg/sec in roll, –30 deg/sec in pitch, and 25 deg/sec in yaw. These rates assume that the forebody separated and maintained the same rates experienced by the intact orbiter at the moment of the CE. This is not likely, but the rates were assumed to be representative. Additionally, the shape of the forebody was held constant through the simulation, although it is likely that the forebody's shape and mass properties were changing due to debris shedding. The simulation is believed to address the dominant variables affecting the motion of the forebody. However, because of the many assumptions used, the simulations can only be considered representative of the type of motion that most probably occurred and not an exact determination of a specific attitude or rate at a specific time.

Figures 2.1-24 through 2.1-26 shows the forebody's angular velocity for each case.

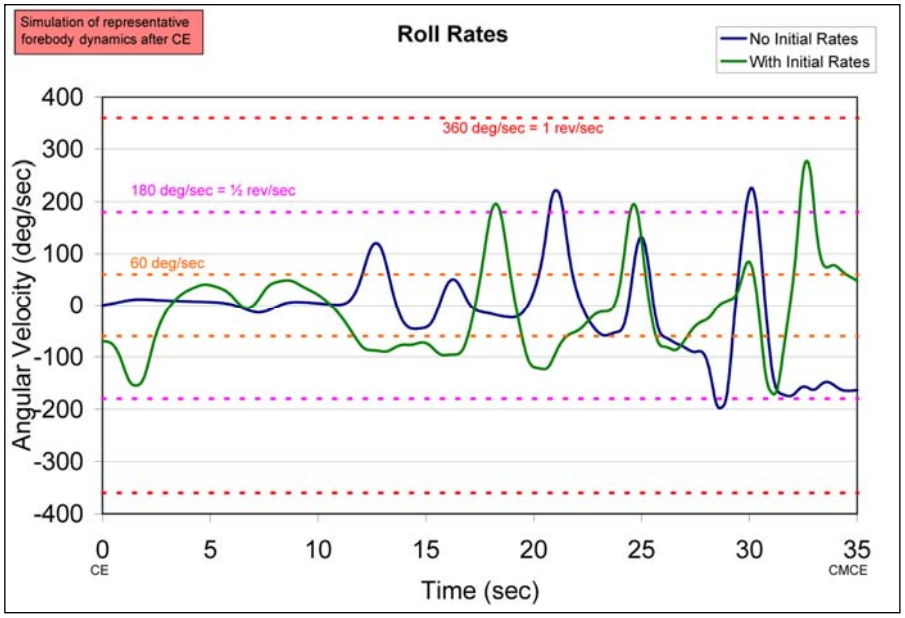


Figure 2.1-24. ***Free-flying forebody roll rates, with and without initial rates.***

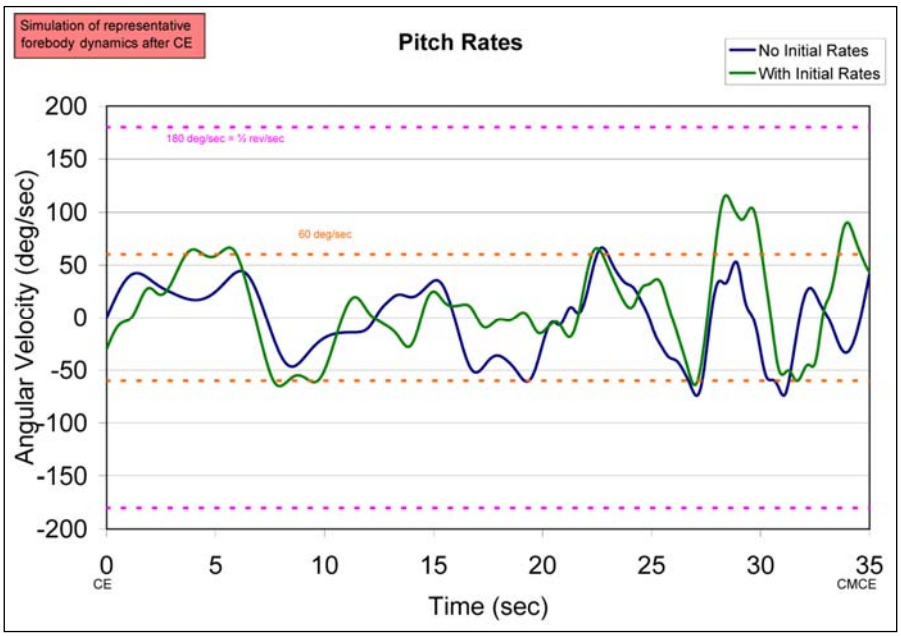


Figure 2.1-25. ***Free-flying forebody pitch rates, with and without initial rates*.**

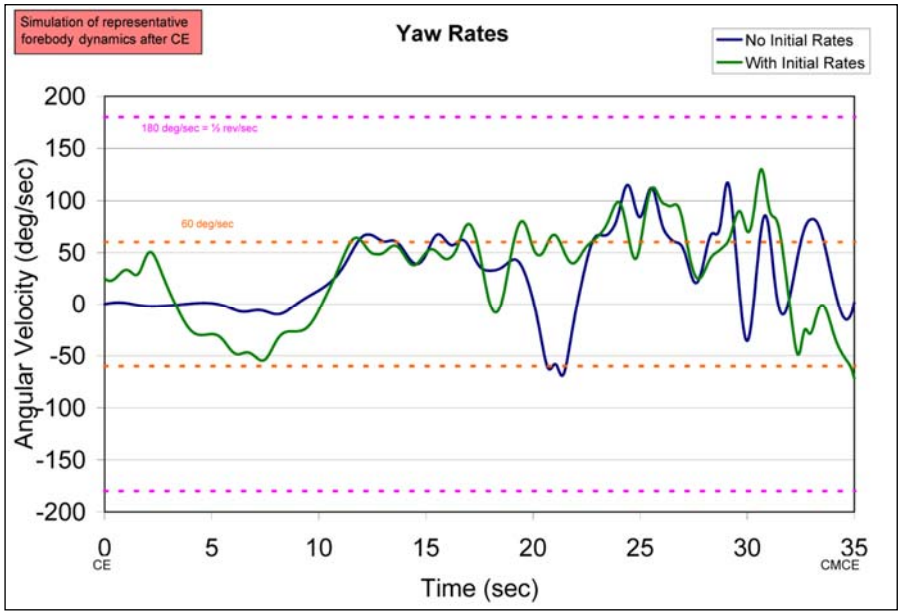


Figure 2.1-26. ***Free-flying forebody yaw rates, with and without initial rates*.**

To summarize the data shown above, the analysis shows that for the first 12 to 15 seconds of free-flying motion with no initial rates, the rotation rates remained extremely low. For the case with initial rates, they remained fairly low (<60 deg/sec, or ~0.17 rev/sec) as well. Following that period, with or without initial rates, the rotation rates climbed to as high as 0.5 rev/sec by the CMCE.

This dynamic attitude motion was combined with trajectory data in the same way as previously described for the out-of-control orbiter to determine accelerations at crew seat positions. Each seat was located at a different location from the forebody c.g., so there are slight differences for each location. Figure 2.1-27 shows a chart of the simulation accelerations from the CE to the CMCE for a representative seat position. The accelerations are given in units of G and represent *total* acceleration.

The increasing atmospheric drag experienced by the forebody is seen (figure 2.1-27) in the trend of increasing accelerations. The oscillations are due to the forebody rotation.

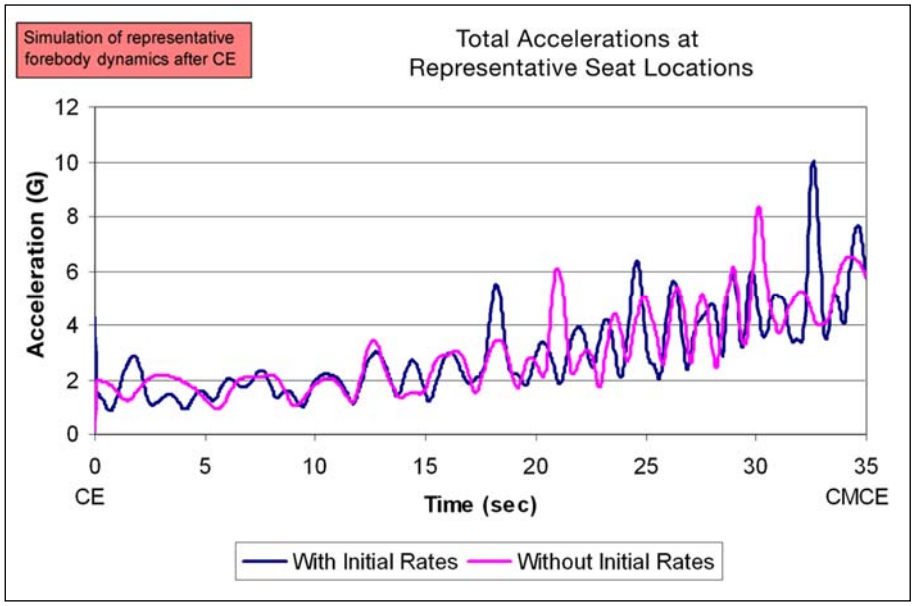


Figure 2.1-27. ***Accelerations at a representative seat location*.**

As the simulation progressed past 15 seconds, the increased rates and increasing aerodynamic drag resulted in higher accelerations at each crew position.

The conclusion of the forebody motion analysis was that the forebody was not in a stable attitude from the CE to the CMCE, but rotating about all three axes at a range from 0.1 to 0.5 rev/sec, with increasing rates and accelerations.

2.1.5 Relative motion comparison

In ground-based video of the mishap, it was noted that when the free-flying forebody could be identified, the image is periodically fluctuating in brightness. This suggested that the forebody was tumbling, which is consistent with the earlier motion analysis. It was not possible to determine the rate of rotation based on the brightening events because the forebody was irregularly shaped and rotating on more than one axis. Therefore, relative motion was selected as an alternate way of evaluating the motion of the forebody in the available video.

Relative motion analysis compares the rate of change of the movement of objects in the field of view (FOV) of a video. Rate of change can provide an estimation of relative motion experienced by the objects within a single frame of reference. Two relative motion analyses were performed. One applied to deceleration of the forebody during breakup (see Section 2.4). The other was a triangulation of relative motion of the forebody as compared to the main engines from two different ground-based videos between GMT 14:00:27.12 (9 seconds after the CE) to GMT 14:00:52.12 (just prior to the CMCE). This provided insight into the free-flying forebody motion for comparison to the aerodynamic analyses described above.

The basic assumption of the triangulation task was that the selected reference orbiter main engine had a known trajectory that was stable compared to the unknown relative position of the forebody. This is a reasonably good assumption because the engine ballistic number was quite high (> 200 psf) compared to the ballistic number of the forebody (~122 psf), and the engines were assumed to be relatively compact and

non-aerodynamic compared to the forebody. Any relative motion between the two trajectories in the video was assumed to be a result of motion of the forebody.

The orbiter engine trajectory was very well understood for several reasons. First, the impact was registered seismically, providing an exact impact time which in turn provided an excellent understanding of its ballistic trajectory. Second, the engines were visible throughout the video through separation and the end of the video, so there was positive identification of the engine. A reference trajectory for the engine was calculated with positions computed at 20-Hz intervals to provide the "known" point in the relative motion.

Two videos with known ground coordinates relative to the trajectory were selected. These videos were known as Hewitt and Mesquite (WFAA4, see Section 2.2, Table 2.2-1) based on the nearest town in Texas where they were recorded (see Section 2.2 and Chapter 4). The FOV was calculated and then trimmed and stabilized from analysis of the imagery, and details about the camera and location were provided by the original photographers. There were a few frames without the engines or the forebody in view; these were not used in the analysis.

An interactive tool was developed to solve for unknown 3-dimensional locations by using 2-dimensional image pairs. The output of this tool led to a reference-trajectory-relative, time-stamped motion path for the forebody.

Figure 2.1-28 provides a snapshot of the tool with known and unknown points selected on still images from the Hewitt and Mesquite videos. The cyan-magenta-yellow axes represent the known engine location specified in the reference trajectory. The red-green-blue axes, with red dot, depict the location of the triangulated CM.

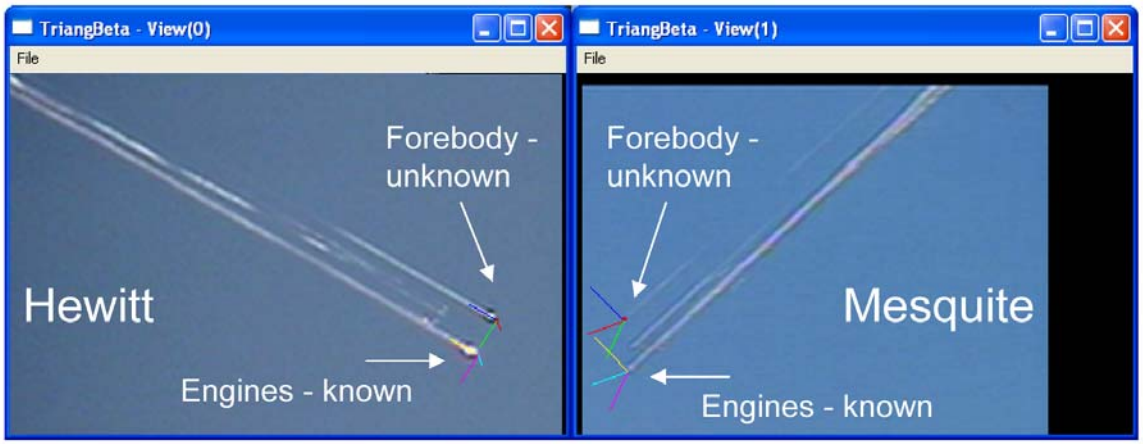


Figure 2.1-28. ***Depiction of the triangulation tool*****.**

With these data, the relative motion in each axis were plotted (figure 2.1-29).

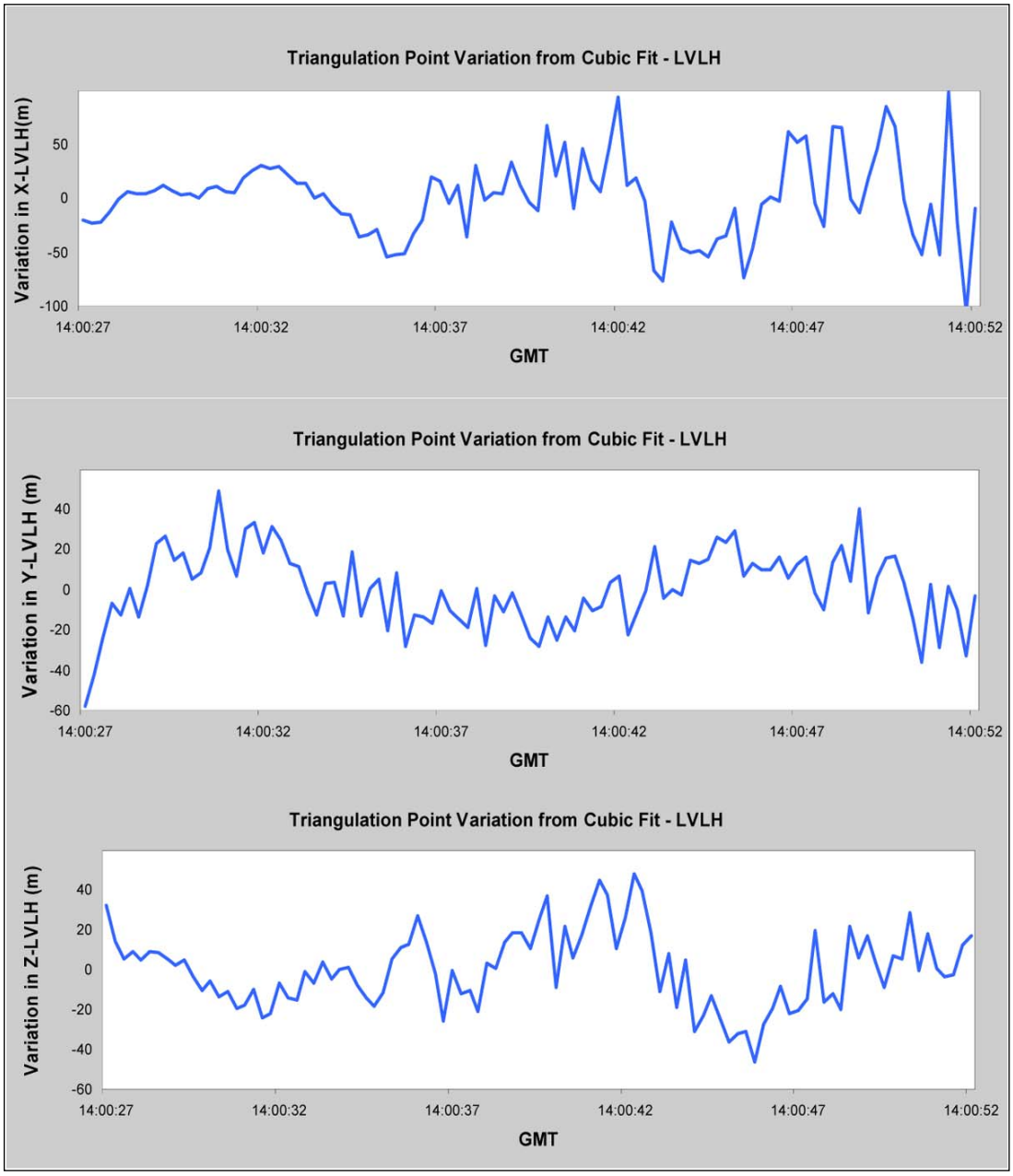


Figure 2.1-29. ***Relative motion between the forebody and the main engines.***

These data essentially represent a wobble motion between the two objects. Since the main engines had a much higher ballistic number and a more compact shape, the wobble is assumed to be predominantly due to rotational motion of the forebody changing lift and trajectory properties slightly as the orientation to the drag vector changed.

A Fast-Fourier Transform analysis was performed on the X-, Y-, and Z-axis data to look for frequency of motion. The frequency of the wobble motion was assumed to correlate to the rotation rate in that axis (figure 2.1-30).

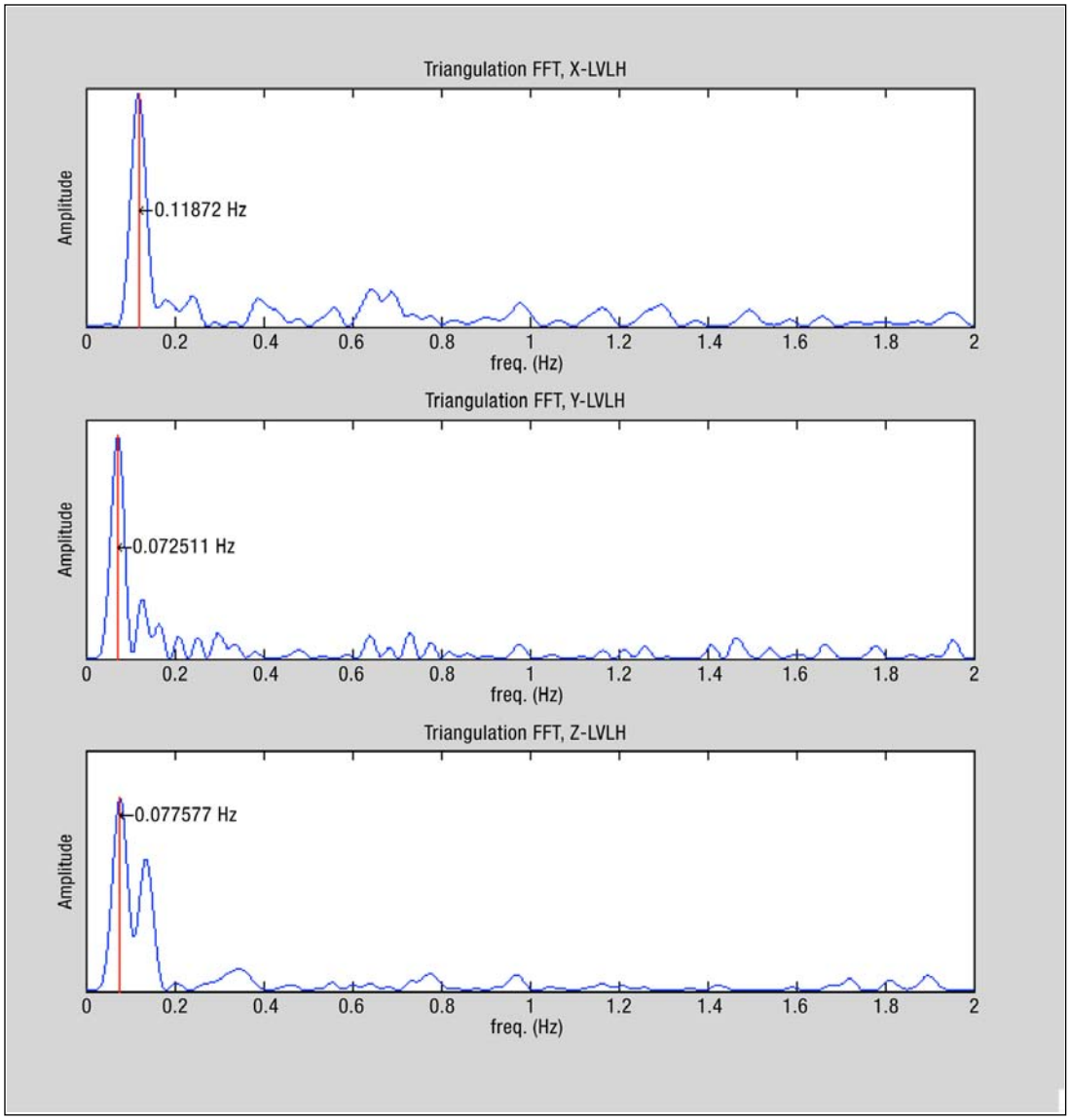


Figure 2.1-30. ***Fast-Fourier transform of relative motion in X, Y, and Z axes in hertz.***

The results show a remarkable level of consistency in frequency of wobble motion in all three axes from approximately 0.07 to 0.12 Hz (cycles per second). This wobble cycle is assumed to reflect the rate of rotation in each axis in revolutions per second. The aerodynamic analysis predicted a rotation rate of approximately 0.1 to 0.5 rev/sec in multiple axes. The triangulation analysis thus provided support for the aerodynamic analysis in that the rotation rates are the same order of magnitude as found in the aerodynamic simulation.

Finding. Triangulation results suggest that the free-flying forebody rotated at approximately 0.1 rev/sec in a multi-axis motion.

Conclusion L2-1. Between orbiter breakup and the forebody breakup, the free-flying forebody was rotating about all three axes at approximately 0.1 rev/sec and did not trim into a specific attitude.

2.1.6 Thermal analyses introduction

Entry guidance and control is principally designed to dissipate the high energy that is associated with the orbital velocity of the vehicle, and arrive at the appropriate altitude and velocity conditions for landing, while managing vehicle heating during entry. The atmospheric entry from orbital velocities produces an extreme thermal and chemical environment. At the most fundamental level, this is due to transfer of the kinetic and potential energy of the entry vehicle into thermal and chemical energy in the atmosphere.

The Earth's atmosphere is composed primarily of molecular nitrogen (N_2) and oxygen (O_2), which comprise approximately 78% and 22% of air, respectively. At hypersonic entry speeds, a strong shock wave forms ahead of the vehicle. This shock wave compresses the ambient air to extreme pressures and temperatures. Hypersonic separation dynamics of debris and shock wave interactions at this altitude are not well understood but probably amplified the effects of entry heating.[14] The temperature and pressure in the shock environment are sufficiently high that the air begins to react chemically. At orbital entry velocities, the primary chemical reactions that occur are dissociation of atoms into charged particles (plasma), dissociation of molecular species, and recombination of the resulting atoms. Molecular dissociation occurs when molecules separate into their constituent atomic species ($O_2 \rightarrow O+O$, $N_2 \rightarrow N+N$). Further, monatomic O and N may then combine to form nitric oxide ($N + O \rightarrow NO$). The amount of dissociation is a function of the vehicle's kinetic energy and the strength of the molecular bonds. Dissociation of atoms in charged particles (plasma) also occurs, but is reduced with altitude and Mach number. For the conditions of the *Columbia* mishap, plasma effects were not considered significant. For the case of the space shuttle orbiter at peak heating conditions, essentially all of the molecular O_2 and approximately 50% of the molecular N_2 are dissociated into atomic O and N in the region immediately behind the shock wave.

While the ambient atmosphere is composed of molecular N_2 and O_2 that may be quite cold, the flowfield around the entry vehicle is composed primarily of reacting molecular N_2 and O_2, monatomic N and O, and NO, all at extreme temperatures. At the vehicle surface, the extreme temperature causes conductive heat transfer, a process in which thermal energy is conducted into the vehicle surface from the adjacent hot gas. Further, the presence of dissociated N and O atoms allows for two other heat transfer mechanisms – catalytic heating and oxidation heating. Both of these processes occur when exothermic reactions (atomic recombination and oxidization, respectively) occur at the vehicle surface. The thermal protection materials used on the orbiter are designed specifically to minimize the influence of these two phenomena and the conduction of the high temperatures into the vehicle's structure.

The efficiency by which energy is transferred from the hot, chemically-reactive shock layer gas into the vehicle is called the heat transfer coefficient. For the trajectory conditions that the orbiter experienced just before breakup, theoretical models predict less than 5% of the energy in the flow is actually transferred into the vehicle. The remainder of the energy is left in chemical and thermal modes in the wake of the vehicle. However, at off-design conditions (e.g., vehicle damage or off-nominal flight attitudes), substantially higher heat transfer coefficient values are possible, resulting in much higher energy (or heat) transfer into the vehicle.

In a nominal orbiter entry trajectory, the heating rates vary with time and location on the vehicle. The highest peak heating rate of about 60 British thermal unit (Btu)/ft^2-sec occurs at the wing leading edge. Shock waves forming on different parts of the orbiter can intersect creating shock-shock interactions. These shock-shock interactions influence a vehicle's aerodynamics and increase the heat transfer rate and pressure where the interaction impinges on the vehicle's surface.

A thermal analysis was performed on the CM to understand the maximum survivable altitude for an orbiter breakup. Additionally, thermal analyses were performed to compare predicted entry heating to actual debris condition for several items. The purpose of these assessments was to understand the sequence of events and help verify ballistic release times. Analyses were performed on several specific items of debris, including a

[14]NASA TM X-1669, "Flight Experience with Shock Impingement and Interference Heating on the X-15-2 Research Airplane," October 1968, p. 7.

helmet, boot soles, a key piece of middeck floor, a laptop computer, the MADS/OEX recorder, the attach fittings of the CM to the FF, and a payload bay door (PLBD) roller.

Object Reentry Survival Analysis Tool (ORSAT) was used for these thermal analyses. ORSAT incorporates algorithms for trajectory simulation, atmospheres, aerodynamics, aeroheating, and thermal modeling, but its strength is its capability to combine all those algorithms into a time-efficient analysis. For one analysis, the ORSAT outputs were compared to two other models, Boundary Layer Integral Matrix Procedure-Kinetic (BLIMP-K) and Systems Improved Numerical Differencing Analyzer (SINDA).

Simulation and Optimization of Rocket Trajectories (SORT) was used to model the times of release from the reference trajectory for thermal and ballistic analyses. Snewt is a computer program that uses the modified Newtonian method to compute a surface pressure distribution and various aerodynamic coefficients. This was used to predict whether a debris item would stabilize in a given attitude during entry.

For more information on ORSAT, SORT, Snewt, BLIMP-K and SINDA, see Chapter 4.

2.1.6.1 *Thermal analysis – crew module*

The *Columbia* accident prompted a careful look at the question: What is the highest altitude at which an orbiter breakup can occur so that a bare-metal CM can reach the ground without being compromised thermally? The LOC/Breakup Cue Card procedure for crews is based on the estimate that the CM would survive aeroheating if separation took place below 280,000 feet during ascent. This estimate was obtained by a simple thermal analysis, for ascent only, which was performed after the *Challenger* accident.[15] This analysis was based on the assumption that the CM would trim nose into the velocity vector. The thermal element clearly is the most important for entry, but can also play a role on ascent depending on the altitude and speed at the time of vehicle breakup. The team wanted to verify the results of the earlier analysis and to perform an analysis for entry aeroheating.

Obviously, the maximum crew survival altitude is more than just the thermal survival of the CM. Other considerations are the structural capability of the CM to withstand dynamic loads, the crew's ability to escape, and the capability of the existing crew escape equipment (CEE). See the sidebar "Maximum Survival Altitude of the Crew Module" on the following page for a discussion of these issues.

To accomplish the thermal analysis, proper heating rates and sufficient modeling of the CM were required. The heating rates were established by modeling the trajectory of the CM after it had separated from the rest of the vehicle. The initial state vector conditions were dictated by the trajectory profile used. Initially, three trajectory profiles were defined: a typical ascent trajectory; the STS-107 predicted entry from a 39-degree inclination orbit; and a typical International Space Station (ISS) mission entry from a 51.6-deg inclination orbit. Two different inclinations were chosen for entry because entries from higher inclination orbits have higher relative velocities and hotter temperatures. However, analysis showed that there was no detectable difference in the maximum survivable thermal altitudes between the two entry trajectories. Figure 2.1-31 displays the altitude vs. velocity for each of the three trajectory profiles. By design, the ascent trajectory has much lower velocities for a given altitude than the entry trajectories.

[15]ES34-87-47M, Crew Escape Thermal Response Study, May 1987.

Maximum Survivable Altitude of the Crew Module

A more detailed discussion of the maximum survivable altitude of the CM must address the structural and equipment capability of the CM as well as the thermal capability. The CM is relatively strong compared to other orbiter structures as it is designed to sustain a significant delta-pressure load against the vacuum conditions in space. Segments of the CM are also designed to withstand crash-landing loads in some axes of up to 20 G. However, the design intent is to have structure absorb enough of the loads so that crew members in their seats are able to egress, not necessarily to have the CM stay structurally intact. The bottom line is that it is not possible to know for certain what load conditions the CM is truly capable of withstanding. According to an estimation of forces on the Challenger *CM immediately after the explosion, the* Challenger *CM likely experienced 16 G to 21 G*[16] *at orbiter breakup and yet apparently maintained integrity all the way to water impact. Deceleration load spikes in a ballistic trajectory are highly dependent on the initial condition; the lower the velocity at the breakup, the flatter the load spike will be. At higher Mach numbers (generally higher altitudes in the profile) the load spike will be quite high as the change in ballistic number is exacerbated. Since each breakup scenario and resulting trajectory loads case would be different, it can only be said that the higher the altitude, the higher the loads will be, and that based on actual circumstances it may or may not be a factor in loss of CM integrity.*

Rotational forces are yet another constraint to survival. First, they can increase the overall loads that are experienced by the structure and lead to structural breakup. Additionally, they can constrain the crew from moving to the hatch for a bailout. Analysis of loads for the Columbia *mishap showed that rotational loads may be quite high for an unstable, rotating CM (possibly up to 3 G at the seats farthest from the CM c.g.). While it is possible for a restrained crew member to brace and maintain consciousness, this condition would hinder and likely prevent seat egress and bailout. Rotation rate will be dependent both on the initial conditions at the breakup and on the aerodynamic conditions at breakup. In general, at higher altitudes the vehicle will behave more like a spacecraft and less like an airplane. At higher altitudes, momentum and mass properties dominate rotation rate, and rates will take longer to increase. At lower altitudes, the aerodynamic coefficients affecting stability will begin to dominate, and rotation rates can ramp up quickly. In general, the crew should not assume that rotation rates will dampen out and should make every attempt to move to the hatch as quickly as possible for egress.*

The final consideration is the capacity of the suit to support egress (see Section 3.2). The advanced crew escape suit (ACES) is rated to 100,000 feet. This rating is not a performance limit, it is a certification limit. However, no ultimate determination has been made on the maximum temperature and loads at which the suit can survive. Since each bailout situation would result in a different combination of heat and total airloads, it is not possible to pinpoint a specific altitude as a limit.

[16]JSC 22175, STS-51L, JSC Visual Data Analysis Sub-Team Report, Appendix D9, June 1986.

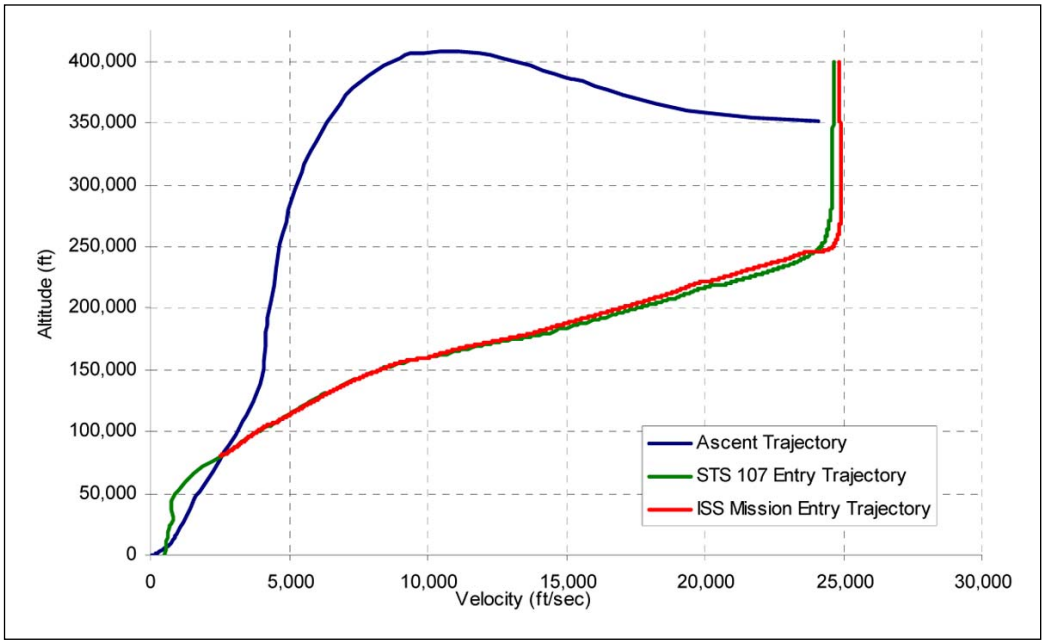


Figure 2.1-31. ***Altitude vs. velocity for selected trajectory profiles*****.**

The analysis was conducted assuming that the CM was intact without any protection from the FF and accompanying Thermal Protection System (TPS). This was a conservative assessment. If the FF and accompanying TPS stays with the CM, it would provide some thermal protection for the CM. This appears to have happened for *Columbia*. However, predicting the exact level of protection provided by the TPS in various scenarios is impossible due to the infinitely variable configurations possible after a chaotic event such as a vehicle breakup. Additionally, the aft bulkhead of the CM is not protected by TPS. Overall, the assumption of no FF/TPS protection results in a conservative (worst-case) estimate of the thermal survival capability of the CM.

The orientation of the CM has a significant impact on the heating model. A stable orientation will result in a more rapid burn-through while a rotating object will distribute heat throughout the object and take longer to burn-through. The preceding motion simulations suggest a slow initial rotation with a gradual increase in rate. However, the *Challenger*-era analysis assumed a stable attitude. *For the purposes of verification and comparison*, both stable and rotating conditions were evaluated. The stable attitudes that were determined in the aerodynamic trim analysis were too complex to model, so the team selected two critical burn-through areas: the aft bulkhead and the forward bulkhead of the CM. These attitudes were defined as the selected bulkhead facing into the velocity vector, the worst-case thermal scenario. Because the post-*Challenger* analysis assumed a stable, nose-down attitude, the analysis assumed the forward bulkhead critical case.

The CM bulkheads have thin sections of skin supported by thicker webs. This complex geometry could not be modeled using the available thermal analysis tool (ORSAT) without substantial modification of the code. Simplifying assumptions were used to bound the problem and provide insight into relative behaviors.

This geometry has the potential for quicker burn-through due to the regions of thin skin, but it also has more mass (due to the thicker webs) that can absorb and distribute the heat away from the thin areas. It is assumed that the actual survivable altitudes would be bounded by analysis on optimistic and conservative cases. Due to these assumptions and constraints, the bulkheads were modeled as flat plates.

For the stable attitude cases, optimistic and conservative bulkhead thicknesses were used to bracket the results. For the optimistic (best-case) modeling, the thickness was established by averaging the entire mass of the bulkhead over the entire area. This optimistic approach produces a large thickness that will distribute heat better and results in higher survivable altitudes because the thicker skin takes longer to burn-through.

It should be noted that aluminum is a good conductor, and it is possible that the bulkhead area can shunt heat to the other areas of the CM, potentially making this "optimistic" estimate conservative. However, heat transport through complex shapes is poorly understood and extremely difficult to model, so it was not included in this analysis.

The conservative (worst-case) thickness for the forward bulkhead critical attitude was approximated by the thinnest portion of the bulkhead. The conservative thickness for the aft bulkhead critical attitude was more complex; it was an average of the thin skin sections of the bulkhead (not using the thickness of the webs). This approach results in lower survivable altitudes because the thinner skin will burn-through sooner. This conservative approach represents failure as a result of hole formation in the thinnest areas. This is conservative because structural failure in a thin individual location is not likely to cause total CM structural failure.

For the case of a spinning, non-stable CM, the CM was modeled as a cylinder. For this case, the skin thickness was averaged based on the mass and dimensions of the entire CM.

The next question was, at what temperature is failure expected to occur? Failure occurs as a result of both thermal and structural loads. There are three types of structural loads: translational loads from aerodynamic drag; rotational loads from vehicle rotation; and loads due to internal cabin pressure. The strength properties of aluminum (the principal structural constituent of the CM) are severely diminished as the temperature increases. Aluminum melts at about 900°F (482°C) to 1,100°F (593°C) depending on the alloy. However, at about 400°F (204°C), the strength of the aluminum begins to seriously degrade and the metal loses its ability to maintain structural integrity. Therefore, when the thermal analysis showed that the selected area on the CM reached 400°F (204°C), the CM was assumed to fail. Figure 2.1-32 shows the estimated strength vs. temperature for aluminum.

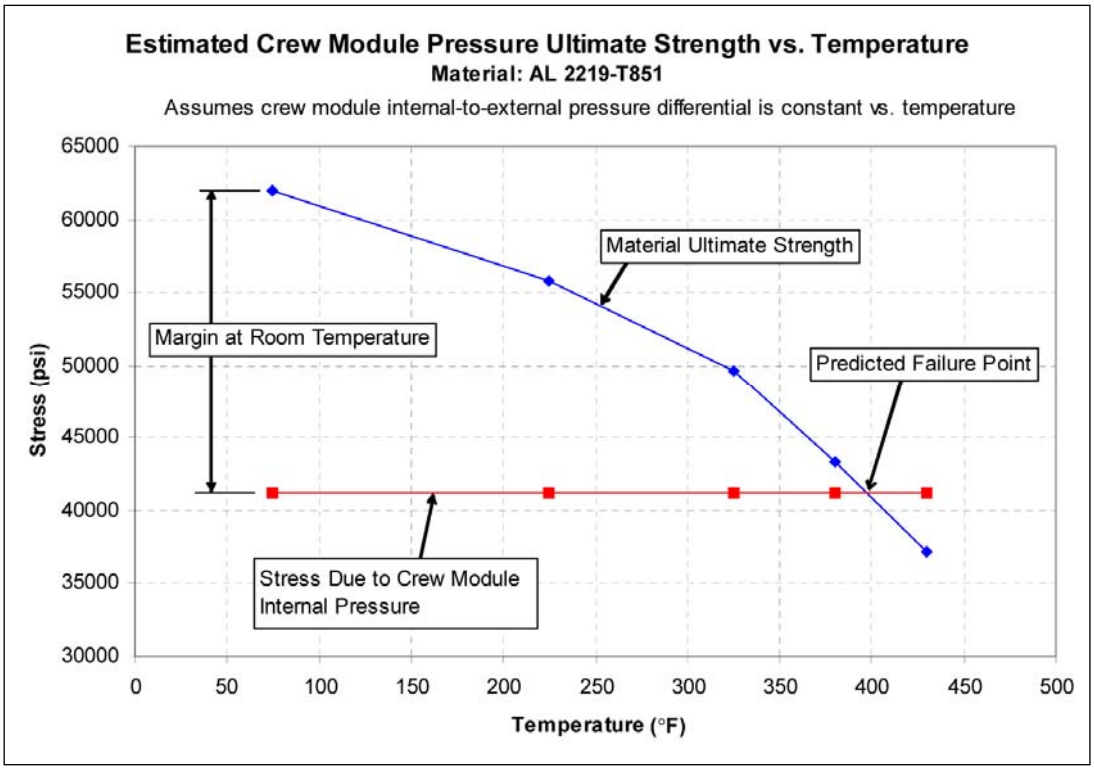


Figure 2.1-32. ***Crew module stress vs. temperature.***

Table 2.1-1 shows the results of the analysis. The table is divided into the two different trajectory profiles: ascent and ISS mission entry. Under each trajectory is a listing of all the cases and the values for their respective ballistic number, skin thickness, and resulting maximum survivable CM altitude.

Table 2.1-1. ***Maximum Survivable Altitude of Crew Module***

	Ballistic number (psf)	Thickness (in.)	Maximum Altitude (feet)
Ascent			
Aft bulkhead (optimistic)	88	0.34	>300,000
Aft bulkhead (conservative)	88	0.07	250,000
Fwd bulkhead (optimistic)	100	0.25	>300,000*
Fwd bulkhead (conservative)	100	0.03	120,000
Spinning	90	0.15	>300,000
ISS Mission Entry (51.6-deg inc)			
Aft bulkhead (optimistic)	88	0.34	160,000
Aft bulkhead (conservative)	88	0.07	130,000
Fwd bulkhead (optimistic)	100	0.25	140,000
Fwd bulkhead (conservative)	100	0.03	105,000
Spinning	90	0.15	150,000

*Correlates to the post-*Challenger* analysis case.

The analysis showed that the worst case is the forward bulkhead critical stable attitude (nose into the wind, and hole formation resulting in total structural failure). The stable attitude case was done only to compare to the earlier *Challenger* post-accident analysis, which concluded that 280,000 feet was the maximum thermal survivable altitude of the CM on ascent. However, the most relevant and realistic data come from the spinning case since the CM is not believed to have a stable aerodynamic trim condition. This analysis yielded a maximum thermal survivable breakup altitude that was greater than 300,000 feet on ascent and 150,000 feet on entry.

Finding. The estimate for maximum thermal survivability on ascent of 280,000 feet is a reasonable estimate.

Finding. The maximum thermal survivable breakup altitude for the CM on entry is approximately 150,000 feet.

As previously mentioned, this only provides the thermal element for the maximum survivable altitude for a orbiter breakup. See the sidebar, "Maximum Survival Altitude of the Crew Module," for a more detailed discussion.

In summary, the shuttle CM could probably thermally survive a breakup up to 300,000 feet on ascent and below 150,000 feet during entry. Aerodynamic loads may cause rapid structural failure of the CM at lower altitudes. Rotational loads may prevent the crew from translating to the crew hatch. High loads may also result during crew separation if the Mach number is still high (particularly on entry). The ultimate survival limits (altitude, air speed, thermal load, etc.) of the ACES remain unknown.

Conclusion L3-3. The actual maximum survivable altitude for a breakup of the space shuttle is not known.

Recommendation L2-2. Prior to operational deployment of future crewed spacecraft, determine the vehicle dynamics, entry thermal and aerodynamic loads, and crew survival envelopes during a vehicle loss of control so that they may be adequately integrated into training programs.

2.1.6.2 *Thermal analysis – helmet*

A thermal analysis was performed on a helmet to compare with ballistic estimates of release time. ORSAT predicted that helmets released at GMT 14:01:03 would match the debris results when the thermal decomposition temperatures were used to assume how much of each material would remain (see Section 3.2). This coincides remarkably with the ballistic estimates from SORT, which showed helmet release times within ±10 seconds of this time. This also coincides with the time range determined for the breakup of the forebody.

2.1.6.3 *Thermal testing – boot soles*

The geometry of the boot soles was too complex to model for ORSAT, especially because of the many alternatives for initial conditions. Boot soles of flight-like boots were thermally tested in an attempt to match the observed thermal damage (see Section 3.2). Test results could not be correlated directly to the debris observations, possibly because the test conditions did not sufficiently approximate the entry environment conditions.

2.1.6.4 *Thermal analysis – lithium hydroxide stowage volume door*

The lithium hydroxide (LiOH) stowage volume door was recovered with portions of the two seats still attached (figure 2.1-33). The LiOH door is located on the middeck floor with seats 6 and 7 attached to the door (figure 2.1-34). The condition of the door was markedly different from other segments of the middeck floor in that significant thermal erosion of the thinner areas had occurred (see Section 2.4). The hypothesis was formed that additional mass attached to this piece of debris resulted in higher heating. A thermal analysis was performed to determine whether the additional mass provided by the two seats and/or two crew members would result in the observed thermal damage.

Figure 2.1-33. ***Recovered lithium hydroxide door with some seat legs still attached*.**

Figure 2.1-34. ***Example of an intact lithium hydroxide door with seats 6 and 7 attached*.** **[Crew Compartment Trainer]**

Composed of aluminum 7075, the LiOH door is about 41 in. long and 28 in. wide, and has a mass of 41.6 lbs. Figures 2.1-35 and 2.1-38 show the upper surface and lower surface of the LiOH door. The thickness of the door varies and, in some cases, is as low as 0.05 in. The average thickness of the LiOH door is 0.39 in.

Photographs of the recovered LiOH door are shown in figures 2.1-36 and 2.1-38. The thermal failure corresponds to the thin sections of the door. Sections that are thicker show little or no thermal erosion. From observations of the material deposition patterns and the burn-throughs on the recovered door, the major thermal effects were directional, with flow impinging primarily on the bottom side (the side with the stiffeners). This is supported by the highly directional deposition on seats 6 and 7 (see Section 3.1).

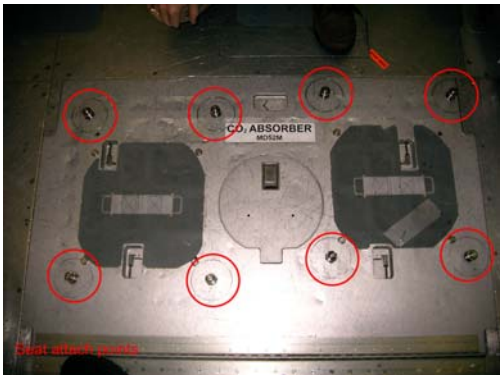


Figure 2.1-35. ***Upper surface of the lithium hydroxide door, facing the middeck*** **(mockup).**

Figure 2.1-36. ***Top surface of the recovered lithium hydroxide door, facing the middeck*****.**

Figure 2.1-37. ***Lower surface of the lithium hydroxide door, facing the lower equipment bay*** **(mockup).**

Figure 2.1-38. ***Lower surface of the recovered lithium hydroxide door, facing the lower equipment bay*****.**

SORT estimated that the release time occurred at GMT 14:00:59. Four configurations of the LiOH door were analyzed to capture the possible release geometries. Configuration 1 was the LiOH door re-entering by itself. Configuration 2 was the LiOH door with two empty seats attached. Configuration 3 was the LiOH door, the two seats, and a suited occupant seated in one of the seats. Configuration 4 was two suited occupants in the seats attached to the LiOH door. The possible pitch trim attitudes for these configurations were predicted using Snewt, and the results can be seen in figures 2.1-39 through 2.1-43. Untrimmed (tumbling) cases were also evaluated for each configuration to determine the sensitivity of thermal effects to attitude. All these configurations assume that the LiOH door (and the attached seats and crew members) was a single free-flying unit with no other CM structure attached.

The main difference among these configurations is the ballistic number. The ballistic number determines the trajectory, thus the velocity profile. The velocity, in turn, affects the heating rate. For objects with large ballistic numbers, the heating rate is greater than for objects with small ballistic numbers.

Since the minimum thickness of the door is 0.05 in., the door was initially modeled with this thickness to determine whether the aluminum would ablate. Ablation is defined as the amount of energy required for an object to reach its melting temperature *and* overcome its heat of fusion. An object demises when the heat of ablation has been reached; in other words, the objects burns through. This is a lower threshold case than is realistic because the thicker areas of the door would act as a heat sink. If the 0.05-in. plate ablated, the thickness was increased to see at what plate thickness it would not ablate.

The door release time was also adjusted 5 and 10 seconds earlier than the estimated time predicted by SORT to see whether the release time had an impact on the results. The initial temperature in the analysis was 80°F (27°C) for all cases except one, where the initial temperature was increased to 200°F (93°C). This was done to determine the sensitivity to the initial conditions; for example, if the CM interior environment was heated significantly prior to breakup and release of the door. However, there is no evidence to suggest that this was the case.

Configuration 1 (Door alone)

The first configuration was a simple flat plate (representing the door) entering by itself. In this configuration, the door was modeled both for the tumbling case and entering normal to the flow fixed angle of attack of 90 deg (figure 2.1-39). This configuration has the smallest ballistic number, so it will receive less heat than the others. The ballistic number was 2.9 psf for the normal to flow case and 6.1 psf for the tumbling case.

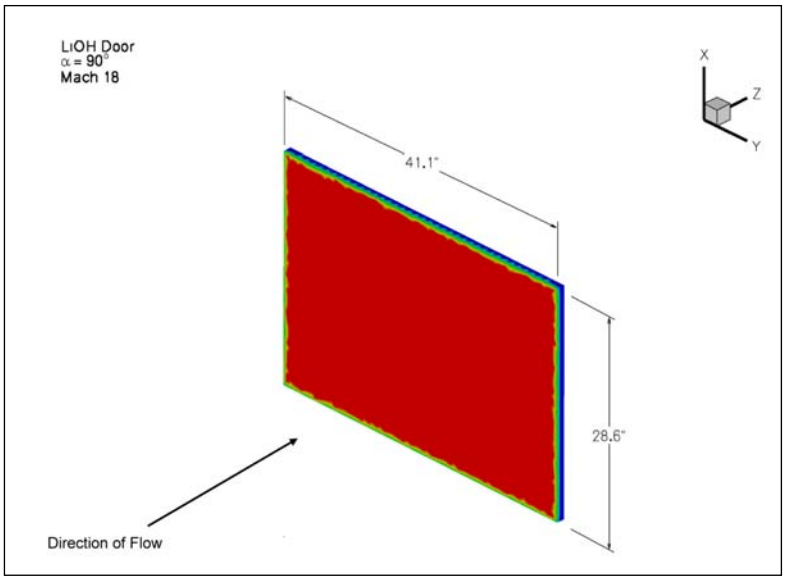


Figure 2.1-39. ***Aerodynamic model of the lithium hydroxide door exposed by itself at an angle of attack of 90 degrees*****. [Configuration 1]**

For this first configuration, neither the tumbling nor the normal-to-flow case predicted that the aluminum melting temperature would be reached.

Configuration 2 (Door with two seats)

The second configuration was the door with empty seats. Since each of the seats has a mass of about 50 lbs., the total aerodynamic weight increased to 141 lbs. This includes the two seats (100 lbs.) plus the door (41 lbs.). In this configuration, the door and the seats were modeled for the tumbling case and also entering at fixed angles of attack, –92 deg or 96 deg (figures 2.1-40 and 2.1-41). This configuration has the second smallest ballistic number. The worst-case (highest) ballistic numbers for the non-tumbling and tumbling cases were 11.5 psf and 20.0 psf, respectively.

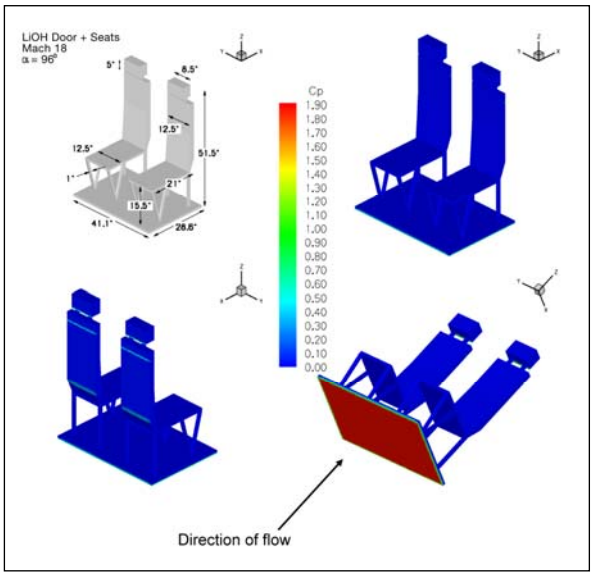


Figure 2.1-40. ***Pressure distribution of the lithium hydroxide door and seats at an angle of attack of 96 degrees*****. [Configuration 2]**

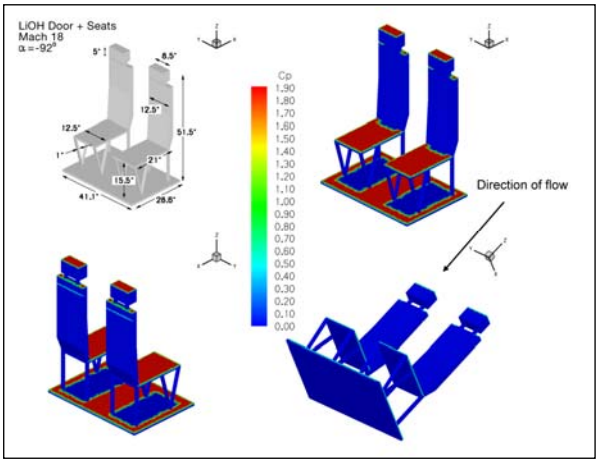


Figure 2.1-41. ***Pressure distribution of the lithium hydroxide door and seats at an angle of attack of –92 degrees*****. [Configuration 2]**

The worst-case tumbling results showed that the melting temperature of the aluminum would be reached, but the estimated heating profile would not overcome the heat of fusion, so no burn-through was predicted.

Configuration 3 (Door with one seated crew member)
In the third configuration, the aero mass was increased to 401 lbs. (41-lb. door with two 50-lb. seats and one suited occupant in seat 7, with a weight of 170 lbs. plus 90 lbs. of launch entry suit). The analysis performed with Snewt did not reveal a stable trim attitude. The ballistic number was 34.0 psf. Figure 2.1-42 shows the geometry.

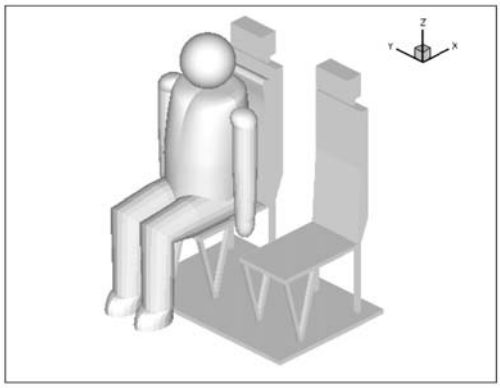


Figure 2.1-42. ***Geometry of a single suited occupant on the lithium hydroxide door*****. [Configuration 3]**

For the given initial conditions, a 0.05-in.-thick plate was predicted to demise. With the thickness increased to 0.07 in., the plate is predicted to survive.

Configuration 4 (Door with two seated crew members)
The fourth configuration was the door with two suited crew members in the seats. Figure 2.1-43 shows this configuration for the trim attitude of 122 deg. In this configuration, the total aerodynamic mass is 690 lbs. This configuration was evaluated for both the tumbling case and a fixed angle of attack in an effort to understand the sensitivity to the attitude condition. This configuration has the largest ballistic number with the worst-case (highest) ballistic number for the non-tumbling case of 42.4 psf and 48.6 psf for the tumbling case.

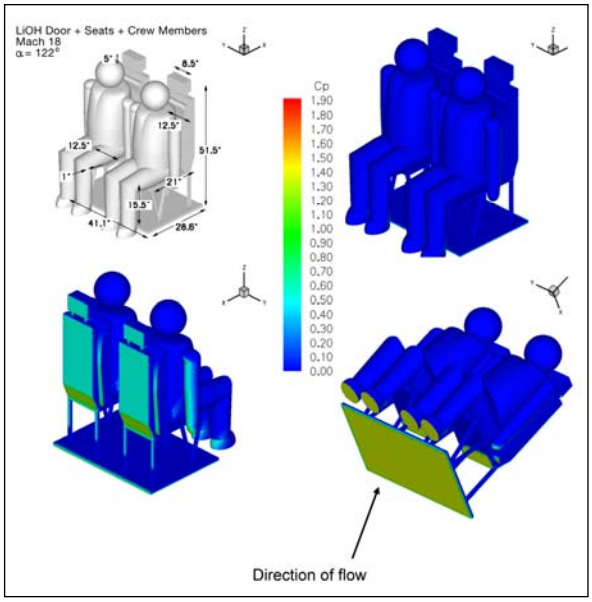


Figure 2.1-43. ***Pressure distribution of the lithium hydroxide door and seats with suited crew members at an angle of attack of 122 degrees*****. [Configuration 4]**

Since the 0.05-in.-thick plate was predicted to demise in the third configuration, it also is predicted to demise in the fourth configuration, which has a larger ballistic number for both the tumbling and the non-tumbling conditions. The 0.05-in.-thick plate is predicted to demise 6 seconds after being exposed. A 0.08-in.-thick plate is predicted to demise 17 seconds after being exposed. However, a 0.09-in.-thick plate

is predicted to survive. Increasing the initial temperature to 200°F (93°C) for the 0.05-in.-thick plate case did not change the time predicted for the plate to demise.

The heat rates generated by ORSAT were compared to another model, BLIMP-K. The heat rates predicted by ORSAT were about 12% less than those predicted by BLIMP-K, but were generally comparable. A detailed thermal math model using the SINDA and ORSAT heating rates was constructed for a section of the door and confirmed the results predicted by the ORSAT. In this model, only the fourth configuration was predicted to demise.

For the GMT 14:00:59 estimated release time, the analysis implies the third and fourth configurations (door with one and two crew members, respectively) can produce the thermal damage observed on the recovered LiOH door. Since visual inspection of the recovered door suggests that the object was not tumbling, the stable fourth configuration is the most thermally viable solution. However, both the third and the fourth configurations are improbable because the thermal effects would have melted the seat straps and released the crew members. The surrounding debris was evaluated to determine whether the LiOH inside the compartment caught on fire. Significant portions of LiOH canisters were recovered as well as other items stored in the compartment. Both the canisters and the structures of the compartment were not seriously thermally damaged, which strongly suggests that a fire did not occur.

It is likely that the simplified nature of the assessment could not accurately model this complex object. As a result, the ORSAT analysis on the LiOH door was inconclusive.

2.1.6.5 *Thermal analysis – payload and general support laptop computers*

Some recovered debris items were identified as pieces from the crew's payload and general support laptop computers, none of which were recovered intact. An analysis was performed to determine whether aerothermal heating could cause the destruction of a laptop after it was released from the CM or if it had to be pre-heated inside the CM. If the laptop had to be pre-heated, this analysis could give an indication of what the thermal environment was inside the CM before breakup.

The estimated maximum temperature from aeroheating alone was well above the temperature at which the battery will explode, likely fragmenting the laptop casing. No conclusion could be made about the thermal environment inside the CM before breakup from this analysis.

2.1.6.6 *Thermal analysis – Modular Auxiliary Data System/orbiter experiment recorder*

The OEX recorder, which was part of *Columbia*'s MADS, was found near Hemphill, Texas in near-perfect condition. Figure 2.1-44 shows the recovered OEX recorder. The data on the tape in this recorder were critical to the accident investigation, making the recorder one of the most important recovered items from *Columbia*.

Figure 2.1-44. *Recovered orbiter experiment recorder from STS-107*. [*Columbia* Reconstruction Database, debris item no. 54057]

Recommendation L3-3. Future spacecraft design should incorporate crashworthy, locatable data recorders for accident/incident flight reconstruction.

The rectangular shape of the box made it a good candidate for ballistic analysis providing a release time with high confidence. The box showed no signs of thermal erosion or exposure to high temperature environment. Text labels that have been imprinted on the box still remain. The data that were recovered from the recorder were recorded on magnetic tape that delaminates at 125°F (52°C). An analysis was performed to estimate the thermal damage that a free-flying OEX recorder would receive due to aerothermal heating after release from inside the CM.

The OEX recorder was located in the lower equipment bay (underneath the middeck floor) of the CM. It is loosely covered by a fiberglass shroud that channels cooler air over the recorder. The shroud is bolted over the OEX recorder but is not directly attached.

The OEX recorder case is composed of aluminum 6063. The thermal properties of aluminum 6061 were used in the analysis because the values were already in the ORSAT material database and the differences in thermal properties to aluminum 6063 were negligible. The shell thickness is 0.25 in. and the overall dimensions of the box are 19 in. long, 15.5 in. wide, and 5 in. high. The total weight of the box is 53.8 lbs.

Ballistic analysis produced an estimated release time of GMT 14:01:02. A state vector from SORT was used as a starting point in ORSAT to simulate the trajectory and heating rates of an entry from that time – altitude 131,780 feet, velocity 12,717 feet per second (fps), and flight path angle –6.3 deg. The initial temperature of the OEX recorder used was 80°F (27°C), however this may be a conservative assumption because the internal electronics of the OEX recorder cause the device to run at relatively high temperatures.

Figure 2.1-45 shows the surface temperature profile for the OEX trajectory. The surface temperature was predicted to reach 470°F (243°C). This is well below the melt temperature of the aluminum, 1,100°F (593°C). Aluminum structurally weakens at 400°F (204°C), and some deformation would be expected if the recorder achieved this temperature. However, there is no deformation of the aluminum casing.

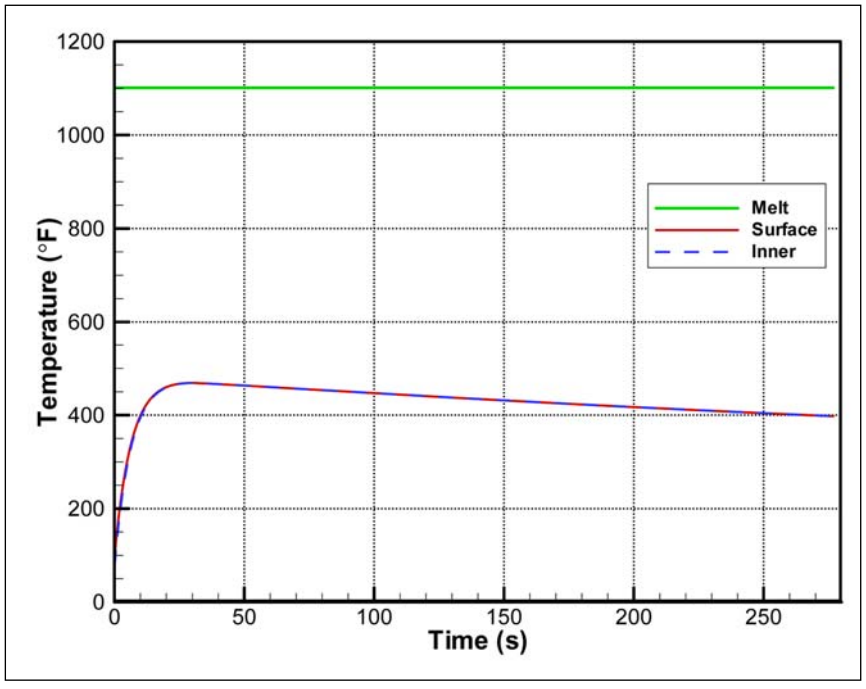


Figure 2.1-45. ***Temperature vs. time of the orbiter experiment recorder shell if released at GMT 14:01:02.***

The lack of external thermal damage is quite noteworthy. Anodizing is still mostly present, as are the exterior labels. Exposed wires still have insulation. Finally, since aluminum is a good conductor it is expected that the internal wall temperature would have been close to the surface temperature. Since the recorder's magnetic tape delaminates at 125°F (52°C), this also suggests that the aluminum may not have reached the predicted high temperature of 470°F (243°C).

The conclusion is that the OEX recorder was not released from the CM independently. If the temperatures reached as high as predicted, the magnetic tape would have delaminated and the casing would have likely been deformed. This analysis combined with the relatively pristine condition of the recorder seems to imply that something must have protected the recorder during entry before it was completely exposed to the aero-thermal heating environment. Ultimately, no conclusion can be made as to what ancillary structure or other mechanism protected the OEX, only that it was protected.

2.1.6.7 *Thermal analysis – x-links*

Examination of the recovered CM attachment fittings, known as the x-links, identified intriguing thermal damage. The two x-links attach the CM to the FF and the midbody and carry load in the orbiter X body axis (see Section 2.4 for diagrams of structure). While both x-links showed melting in the same relative locations, the starboard x-link displayed more thermal damage than the port x-link. Because the x-links are composed of Titanium 6Al-4V, which has a melt temperature of approximately 3,000°F (1,649°C), very high heating is required to create the damage that was observed on the recovered x-links. Analyses were performed to determine the thermal mechanism that could have caused this damage. Figure 2.1-46 shows a model of a pristine port and starboard x-link, while figure 2.1-47 shows the two recovered *Columbia* x-links placed side by side.

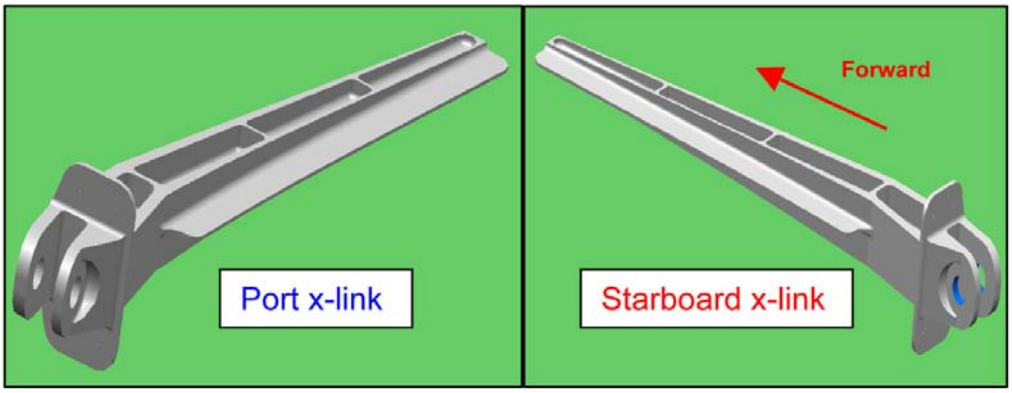


Figure 2.1-46. ***Drawings of pristine port and starboard x-links*****.**

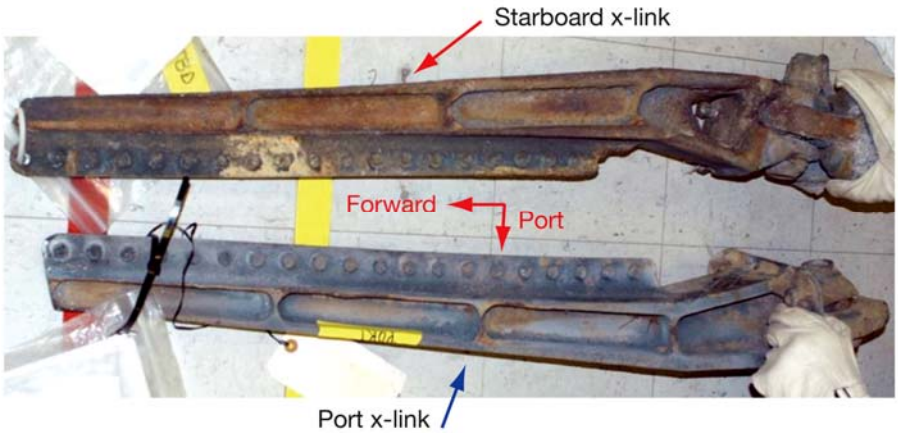


Figure 2.1-47. ***Comparison of port and starboard x-links recovered from*** **Columbia. [*Columbia* Reconstruction Database, debris item no. 1678, and *Columbia* Reconstruction Database, debris item no. 1765]**

The heating experienced by the starboard x-link was severe enough to melt away the top flange and to burn a hole through the 0.25-in.-thick web. Figure 2.1-48 shows another comparison of the two x-links with a better viewing angle to see the flange damage. Figure 2.1-49 shows a close-up view of the starboard x-link where a hole has been melted through the web.

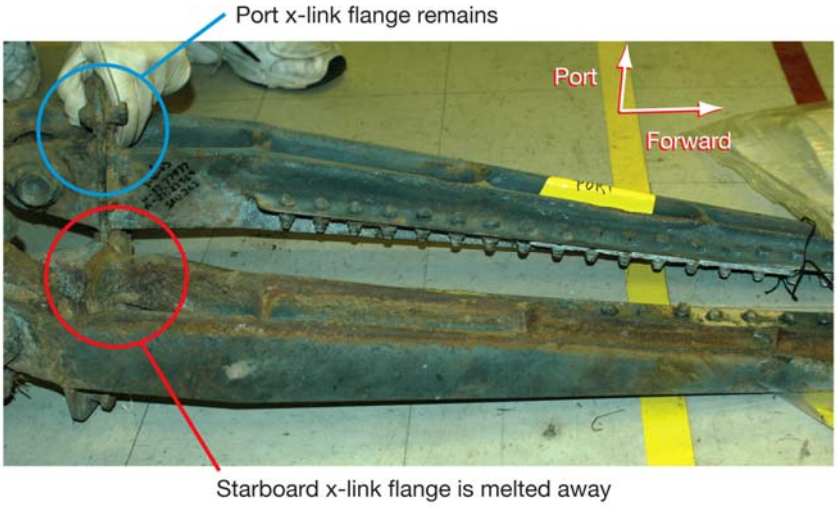


Figure 2.1-48. *Comparison of port and starboard x-link flanges*. [*Columbia* Reconstruction Database, debris item no. 1678 (top) and *Columbia* Reconstruction Database debris item no. 1765 (bottom)]

Figure 2.1-49. *Close-up of the hole in the starboard x-link*. [Columbia *Reconstruction Database, debris item no. 1765*]

At first it was assumed that the x-links received the thermal damage from entry heating as independent free-flying objects. For the ORSAT analysis, an x-link was modeled as a simple box (38 × 3 × 1.2 in.) at a stable attitude with a mass of 22 lbs. It was assumed the x-link broke off the orbiter at 200,000 feet with a relative velocity of 17,145 fps and a flight path angle of -0.68 deg. Based on debris field analysis, the actual separation was likely at a lower altitude and Mach number with lower resultant heat rates, making this assessment conservative.

The ORSAT analysis predicted that entry heating would not be sufficient to cause the damage seen. Also, a free-flying x-link was not predicted to stabilize in one attitude; a tumbling x-link would diffuse the heat better and also not result in the directional heating seen. The similar melting patterns on the two x-links suggests that they were in the same relative orientation at the time the thermal event occurred, presumably still in place attaching the CM to the FF. This led to a more detailed analysis to understand what heat rate would be required to show the damage seen.

The amount of heat to completely melt an object is known as the heat of ablation. The heat of ablation per mass has two components, which are additive: latent heat and sensible heat. Latent heat is the amount of energy in the form of heat released or absorbed by a substance during a phase transition (such as from solid to liquid). The latent heat per mass for titanium is 187 Btu per pound of mass (Btu/lbm). Sensible heat is potential energy in the form of thermal energy or heat for an object. The amount of sensible heat per mass required to raise the temperature of titanium to its melting temperature can be determined by integrating the plot from the initial temperature to the final temperature (figure 2.1-50). Data did not extend beyond 1,600°F (871°C).

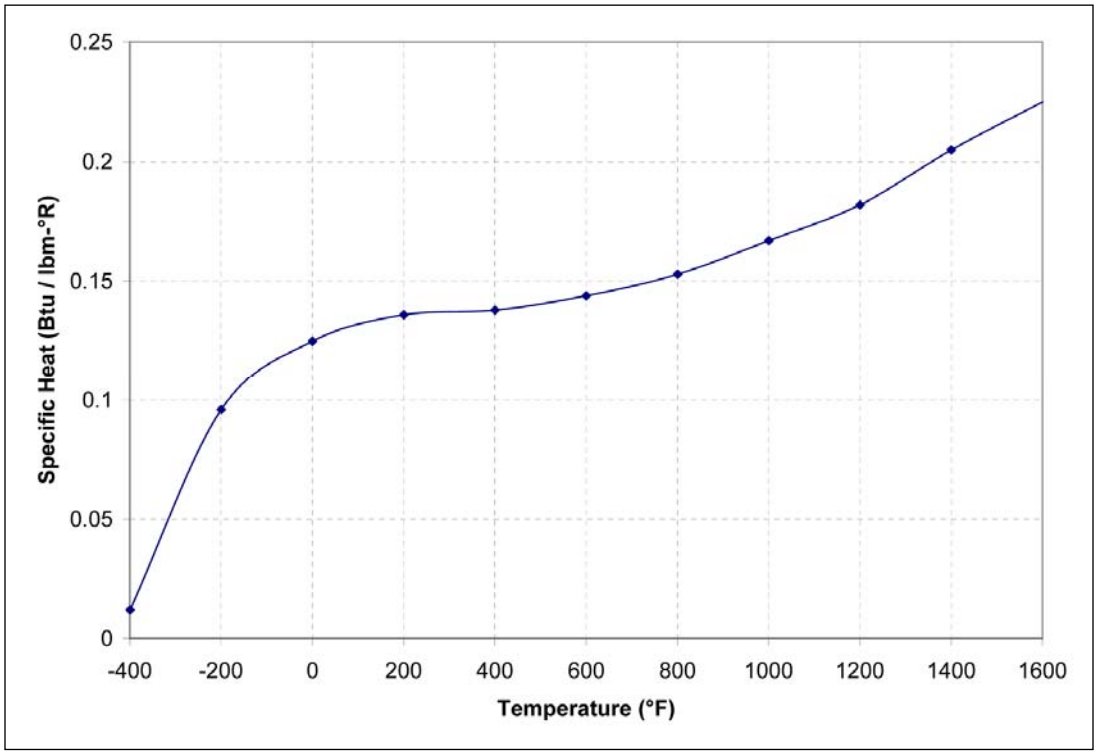


Figure 2.1-50. ***Specific heat of titanium vs. temperature.***[17]

Assuming an initial temperature of 80°F (27°C) and that the specific heat at 1,600°F (871°C) does not change until the melting temperature is reached, the amount of sensible heat required to reach the melting temperature of titanium is 564 Btu/lbm, or 750 Btu/lbm to completely ablate the material. It is possible that

[17]Thermal Protection Materials, NASA Reference Publication 1289, December 1992.

the x-link was pre-heated by earlier exposure to entry heating as the forebody rotated. The maximum pre-heating considered reasonable was 400°F (204°C), the temperature at which the surrounding aluminum structure would begin to soften and likely release the x-link. If the x-link was pre-heated to 400°F (204°C), the amount of sensible heat per mass required to reach the melting temperature is reduced to 521 Btu/lbm and the heat of ablation per mass is reduced to 710 Btu/lbm.

Knowing the heat of ablation per unit of mass allows a first-order estimate of the required heating that must be applied to cause that burn-through hole in the starboard x-link. The dimensions of the burn-through area are 2 in. by 2 in. and the thickness of the x-link in that area is 0.25 in.

The required heating rates to cause the burn-through hole as a function of time for initial temperatures of 80°F (27°C) and 400°F (204°C) are shown in figure 2.1-51. It can be seen that the heat rates are extremely high for either case, and increase exponentially as the time required to cause the hole decreases. For comparison, figure 2.1-51 also shows the heating rate from entry aeroheating for an x-link exposed at 200,000 feet (dark blue line). This shows that the x-links would have to be exposed to sustained directional entry heating for more than 25 seconds to result in the thermal damage received.

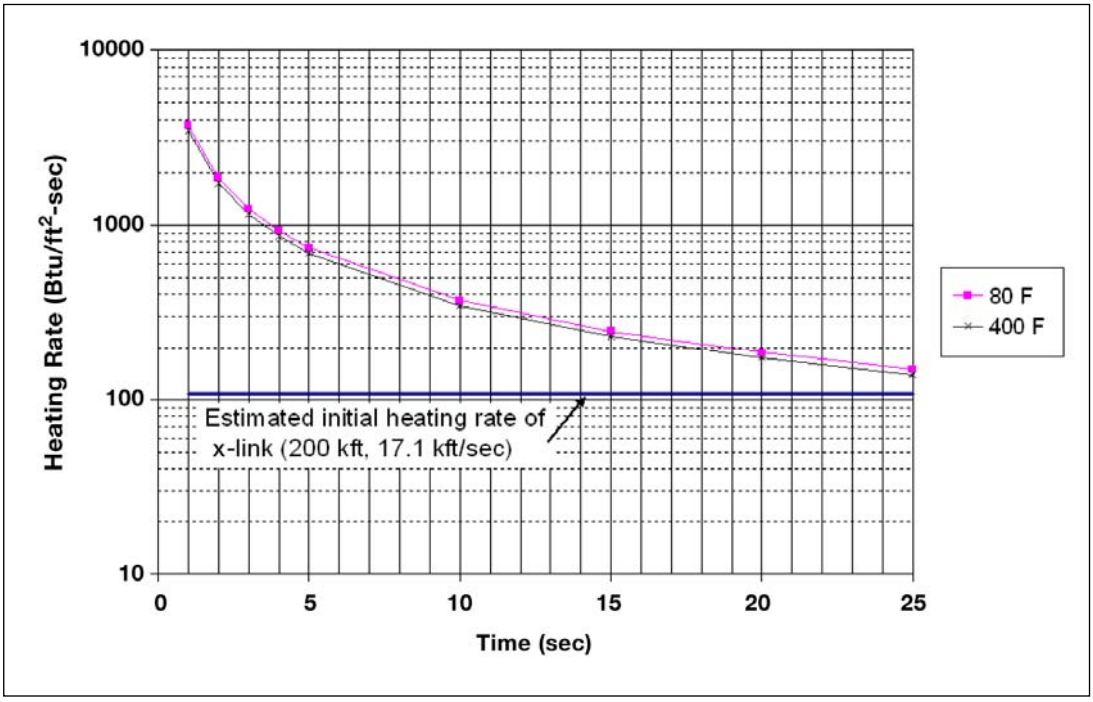


Figure 2.1-51. ***Heating rates required to ablate a titanium hole of 0.25 inch in depth.***

The question now became how such severe and similar directional heating of the two titanium x-links occurred in a short time span without melting the surrounding material, which had lower melting temperatures. Because the forebody was not likely to stabilize at a given attitude, there would be very brief durations of exposure as the x-link happened to pass through the velocity vector as the free-flying forebody rotated. The longest reasonable duration for the rotating forebody to experience peak heating was felt to be less than 3 seconds and possibly as short as 1 second. To melt a hole in 3 seconds, the required heat rate is about 1,350 Btu/ft^2-sec. To melt a hole in 1 second, the required heat rate is over 3,900 Btu/ft^2-sec. These values are an order of magnitude higher than the estimated peak heating due to aeroheating of 110 Btu/ft^2-sec.

Finding. Thermal analyses predicted that entry aeroheating alone was insufficient by an order of magnitude to produce the observed thermal damage on the x-links. Therefore, the x-links must have experienced other heating mechanism(s) in addition to normal entry heating.

This was a very surprising result, and led the team to research other heating mechanisms. Two thermal mechanisms, shock-shock interaction and titanium oxidation/combustion, could generate the observed thermal damage either separately or in tandem. Both of these mechanisms are discussed in detail in Section 2.1.7.

2.1.6.8 *Thermal analysis – payload bay door roller*

Another instance of the thermal erosion of high-temperature materials was seen in the PLBD rollers, which contain notable amounts of titanium. Initially, the interest in the PLBD rollers resulted from a search of possible sources for the titanium deposition that was found on the overhead windows (see Section 2.4).[18] A search for forward structures containing titanium showed that the nearest source of titanium material to the windows was the forward PLBD rollers. These rollers are made of aluminum, titanium, and an Inconel sleeve, and the PLBDs rest on them when the doors are closed. The forward rollers are attached to the top of X_o 582 ring frame bulkhead in close proximity to the windows on which the titanium depositions were found (figure 2.1-52). Several recovered rollers from this location showed pronounced erosion of the exposed titanium surfaces, and one roller was recovered with only part of the Inconel sleeve remaining; all titanium and aluminum inner structures were missing (figure 2.1-53). Table 2.1-2 shows the material properties of stainless steel (A286), Inconel 718, aluminum 2000 series, and titanium 6Al-4V. As shown, the Inconel and aluminum alloys used in the rollers both have lower melting temperatures than the titanium alloy.

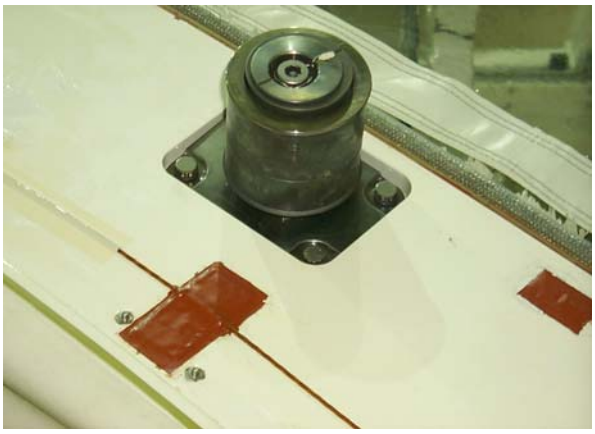


Figure 2.1-52. ***Payload bay door roller showing inner material and Inconel sleeve*.**

Figure 2.1-53. ***Recovered payload cylinder bay door roller with inner material absent and Inconel sleeve remaining*. [*Columbia* Reconstruction Database, debris item no. 10914]**

Table 2.1-2. ***Material Properties Important to Thermal Analyses***

Material	Melting Temperature (°F)	Melting Temperature (°C)
A 286 (stainless steel)	2,500	1,371
Inconel 718	2,368	1,298
Aluminum 2024	1,081	583
Titanium-6Al-4V	3,037	1,669

A computational fluid dynamics (CFD) thermal analysis of the rollers was performed to determine the flow field environment and temperature at the face of a roller for an orientation with the front of the roller facing directly into the direction of travel. Figure 2.1-54 shows the predicted heating rate and temperature distribution along the PLBD roller at Mach = 10.5.

[18]J. D. Olivas, L. Hulse, B. Mayeaux, S. McDanels, P. Melroy, G. Morgan, Z. Rhaman, L. Schaschl, T. Wallace, and C. Zapata, *Examination of OV-102 Thermal Pane Window Debris – Final Report*, KSC-MSL-2008-0178 (in press).

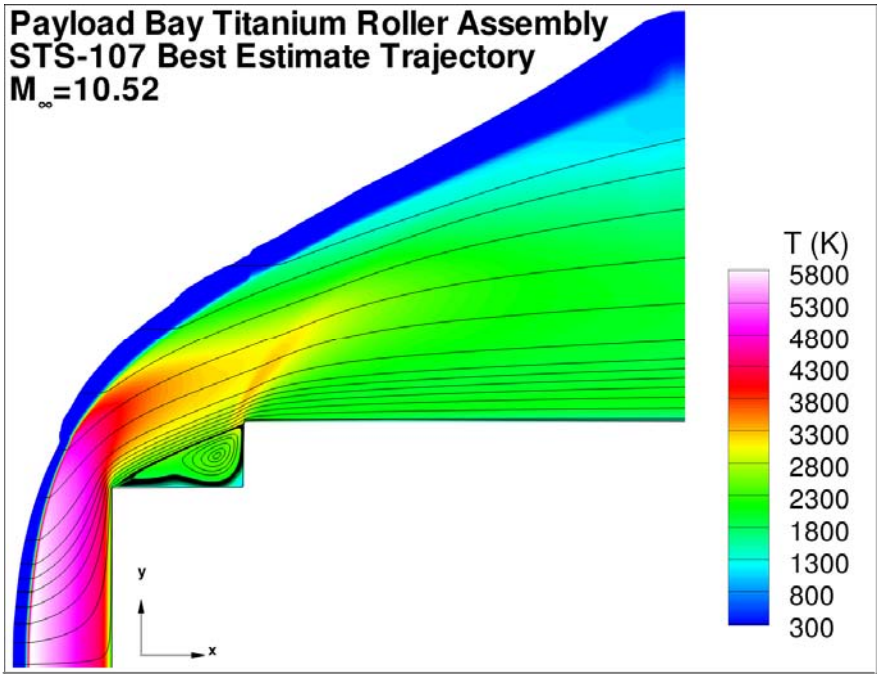


Figure 2.1-54. ***Computational fluid dynamics analysis of heating at the tip of the payload bay door roller for orthogonal geometry into the direction of travel at M = 10.5.***

It can be seen from figure 2.1-54 that for Mach numbers greater than 10, the possibility exists that radiative equilibrium surface temperatures in excess of the melting temperature of titanium (~3,000°F) (1,662°C)) can be achieved. Ablation then could occur by pressure forcing away the molten titanium on the surface.

When ORSAT was used to analyze the case of the titanium rollers, the calculated heating rates across the front face compared very well to the results from the CFD case, given the same assumptions regarding geometric orientation and free-stream conditions (figure 2.1-55) for Mach 7.5 to 10.5.

In figure 2.1-55, the heating rate increases along the front face from the center to the edge because the flow is accelerating around the corner of the front face of the roller. The first spike comes from the flow going over the edge of the inner cylinder, and the second spike comes from the flow over the (outer) sleeve cylinder. If a boundary with grid representing the payload bay forward wall had been used in the CFD analysis, it may possibly have shown a shock-shock interaction that might account for the hole found in the surrounding X_o 582 ring frame bulkhead structure on either side of the roller.

The next question was whether the observed amount of material loss was reasonable for the timeframe. As mentioned before, heating only occurs when the object is facing into the direction of travel, yet it is not believed that the forebody stabilized in a specific attitude. Additionally, the nature of the window deposition indicates that the titanium and titanium/aluminum deposition event was continuous and not cyclical. Therefore, the entire titanium/titanium aluminum ablation event had to occur in a relatively short period of time. Given an attitude change of approximately 36 deg/sec (0.1 rev/sec), the maximum duration of heating considered reasonable was about 3 seconds. In any attitude other than directly into the velocity vector, heating is less but is still present. Therefore, the CFD analysis only showed the peak heat was possible. It seems probable that some additional mechanism would be required to deeply erode the rollers.

The analysis of the x-links (see previous section) also suggests that there might be mechanisms other than simple entry aeroheating at work. The first mechanism evaluated was shock-shock interaction.

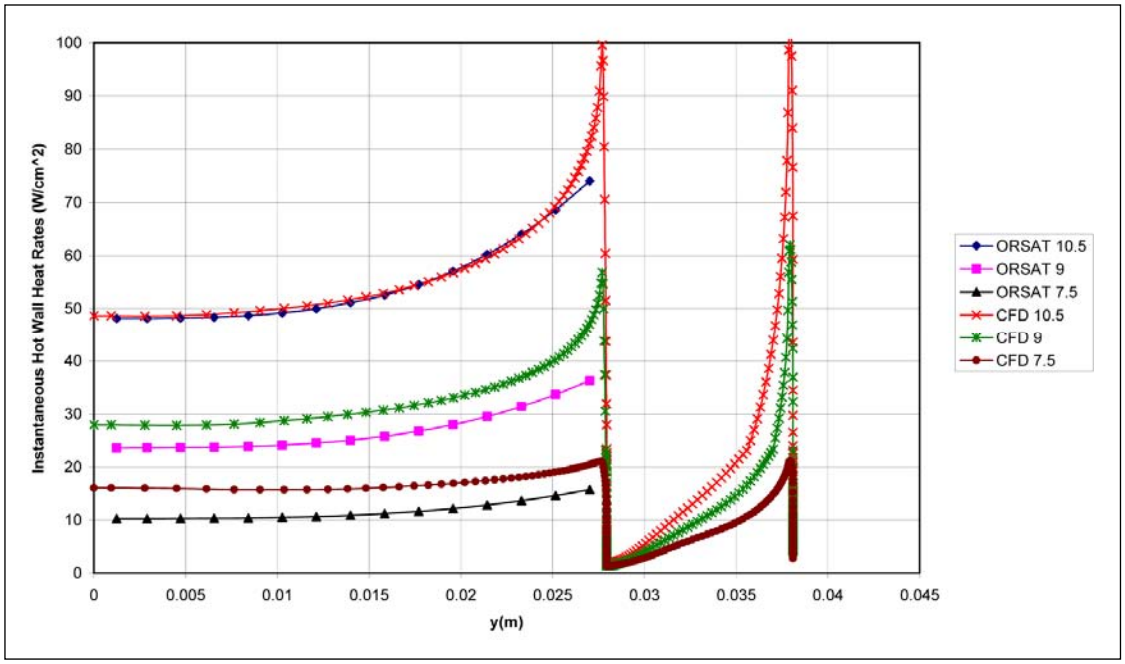


Figure 2.1-55. ***Comparison of the object reentry survival analysis tool and computational fluid dynamics heating rates for the front face of the payload bay roller for various Mach numbers*****. Y axis is radial distance from the center of the roller.**

2.1.7 Shock-shock interaction and combustion

Shock-shock interaction occurs when two shock waves intersect. Figure 2.1-56 shows an example of one interaction on the orbiter leading edge.

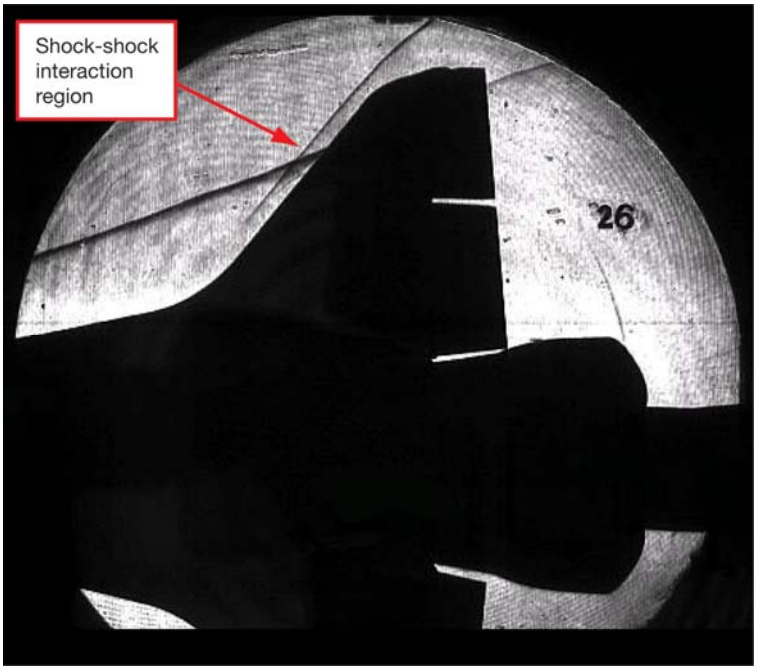


Figure 2.1-56. ***Schlieren photograph of the orbiter showing a shock-shock interaction region for nominal entry*****.**[19]

[19]This profile was taken from orbiter aerothermodynamic heating tests at the Calspan-University of Buffalo Research Center, shock tunnel facility at Buffalo, New York at a Mach number of 10.

These shock-shock interactions influence a vehicle's aerodynamics and increase the heat transfer rate and pressure where the interaction impinges on the vehicle's surface. There are six types of shock-shock interactions.[20] These types are defined by where the incident and bow shock waves intersect, the strength of the incident shock, and the angle on the blunt leading edge surface with respect to the impinging shear layer. Shock-shock interaction was examined to determine whether it could explain the thermal damage observed on the x-links and PLBD rollers.

The best candidate for the extreme heat rates is called a Type IV shock-shock interaction. Type IV shock wave interference can cause stagnation rates to be amplified by a factor of 38 for a Mach number of 16, which is slightly higher than the Mach number at orbiter breakup. This amplification factor could increase the baseline heating rate to almost 4,000 W/cm^2, so it would only take about 1 second to burn-through the titanium. These Type IV shock wave interference scenarios cause a jet that is very small and will result in highly localized heating.

Type IV shock waves are highly dependent on geometry relative to the airflow. Due to the chaotic nature of the vehicle breakup, it was not considered reasonable to attempt to determine the precise geometry for the PLBD rollers or the x-links that would result in a Type IV shock wave. However, experts did determine that there was no reason to rule out shock-shock interactions based on geometry.[21]

Therefore, theoretically a shock-shock interaction can help explain the type of damage seen on the rollers and starboard x-link. However, two questions still remained. The first was the original mass loss rate question from the PLBD rollers. The second question was why surrounding materials with lower melt temperatures, such as aluminum and the Inconel roller sleeves, remained relatively intact in such an intense environment. The aluminum condition could be explained by the fact that it has much great thermal conductivity than titanium. With sufficient mass, it is capable of withstanding higher heating rates because it can create a more effective heat sink. Since most of the CM is made of aluminum, the aluminum portions of the structure may have been very effective at shunting heat away from locally exposed regions during a period of high localized heat exposure. However, the thermal conductivity of Inconel is much lower, closer to that of titanium, and the relative lack of damage to the Inconel cannot be explained by heat shunting. This introduced the possibility that the titanium may have reacted different chemically to the environment than the other materials around it.

Crystallographic assessment by the JSC Materials and Processing Office, the JSC Astromaterials Research Office, the Kennedy Space Center (KSC) Failure Analysis and Materials Evaluation Branch, and the White Sands Test Facility on the nature of the oxide formation on the window indicated that deposition occurred at a temperature that was well above the melt temperature of the titanium alloy. The crystallographic and microscopic work further indicated that the oxide species were fluid once deposited on the windows and did not experience solidification during transportation from the source to the deposition site. These additional clues further implied some significant secondary mechanism.[22]

One important difference between titanium and the other materials is that titanium is highly reactive with O_2. In fact, titanium is often used as an igniter or a promoter, much like magnesium. Oxidation is an exothermic reaction, and the chemical reactions that occur can introduce heat into the material through bulk material conversion from a molten metallic state to an oxidized state. This could explain why the Inconel (which is not particularly reactive in terrestrial O_2 environments) did not thermally erode, although the melt temperature is more than 600°F (316°C) lower than that of titanium (Table 2.1-2), and why the titanium alloyed material reacted and deposited on the windows earlier than other materials.

It is very difficult to predict the combustion process of metals with the analytical tools used for this investigation. The ORSAT code used for much of the aerothermal analysis in this study does have a

[20]Barry Edney, "Anomalous Heat Transfer and Pressure Distributions on Blunt Bodies at Hypersonic Speeds in the Presence of an Impinging Shock," The Aeronautical Research Institute of Sweden, 1968.
[21]Allan Wieting, "Shock-Shock Wave Interference Heating Possibilities," March 2007.
[22]J. D. Olivas, M. C. Wright, R. Christoffersen, D. M. Cone, and S. J. McDanels, *Crystallographic oxide phase identification of char deposits obtained from space shuttle* Columbia *window debris,* Acta Materialia, 2008 (in press).

primitive function for determining oxidation, so the phenomenon is not unforeseen. However, there are many limitations to the model. The ORSAT function only accounts for the mass loss of the material as it is consumed by oxidation, not by a combustion reaction. ORSAT accounts for this by creating an additional component of heat that is formed by the reaction of the metal and molecular O_2. During vehicle deceleration from high Mach numbers, O_2 is dissociated into its monatomic form (O) which is much more chemically reactive than molecular O_2. A literature search and survey of major testing facilities proved that investigation into the effects of high enthalpy-low pressure environments on bare titanium has been very limited in the past.[23]

The shock-shock interaction discussed previously was not easy to test. However, simple experiments could be conducted, using an arc-jet facility, to understand whether titanium combustion could have occurred in the *Columbia* entry environment.

A selection of enthalpy-pressure test points was chosen based on the predicted trajectory and ballistic number of *Columbia* and the free-flying forebody. A series of tests was conducted at the Boeing St. Louis Large Core Arc Tunnel (LCAT) plasma arc facility.[24] This testing showed that in some higher enthalpy conditions the titanium test article eroded within a few seconds and exhibited behavior that is characterized as combustion. Although mass loss rates are not known, this type of oxidation can become a sustained reaction and does not necessarily require the continued exposure to heating to continue.

An analysis was done using ORSAT to see whether the model could account for the thermal effects seen in the arc-jet test without combustion. In lower enthalpy cases, the model matched arc-jet test results very well. However, at the higher enthalpy cases, there were incongruities that could not be accounted for by entry heating alone. [25] The conclusion was that titanium combustion was possible for the *Columbia* entry environment. This appears to be the first documented characterization of this material physical property.

Additionally, the arc-jet test team noted that geometry also appeared to play a possible role in the initiation of oxidation. This provides the intriguing possibility that momentary shock-shock interactions, which might not erode titanium significantly on their own, may have acted as a trigger for oxidation, combustion, and heavy erosion.

The fact that combustion was only seen at higher enthalpies and pressures suggests that if oxidation occurred, it was most likely when the forebody was still intact as the smaller individual objects are not likely to generate the heat required. The fact that the rollers deposited material on the windows combined with the similar directional heating on both x-links implies structural integrity of the forebody, which is consistent with this conclusion.

In conclusion, some of the thermal mechanisms experienced during the *Columbia* entry were different from those of a nominal entry, implying the possibility of shock-shock impingement, titanium oxidation, or a combination of both effects resulting in rapid and selective melting of titanium prior to the aluminum around it.

Conclusion A13-1. Titanium may oxidize and combust in entry heating conditions dependent on enthalpy, pressure, and geometry.

Conclusion A13-2. The heating from a Type IV shock-shock impingement and titanium combustion (in some combination) likely resulted in the damage seen by the forward payload bay door rollers and the x-links.

---

[23]William Rochelle, "Survey of Titanium Testing at NASA, DOE, and DOD Test Facilities," ESCG-4380-06-AFD-MEMO-0011, April 20, 2006.
[24]J. D. Olivas, B. Mayeaux, P. Melroy, and D. Cone, Study of Ti Alloy Combustion Susceptibility in Simulated Entry Environments, AIAA, 2008 (in press).
[25]W. Rochelle, J. Marichalar, M. Larin, A. Dobrinsky, J. Dobarco-Otero, and Ries Smith, "Comparison of LCAT Arc-Jet Titanium Plate Test Data with Aerothermal Predictions," ESCG-4380-07-AFD-MEMO-0012, April 20, 2007.

Recommendation A13. Studies should be performed to further characterize the material behavior of titanium in entry environments to better understand optimal space applications of this material.

2.1.8 Synopsis of motion and thermal analyses

In summary, the motion and thermal analyses of the vehicle and forebody produced significant results and findings. The loads experienced by the intact orbiter and the free-flying forebody were estimated based on a reference trajectory and attitude analyses. The forebody and CM of the orbiter are not expected to trim into a single aerodynamic attitude upon breakup of the vehicle. Thermal analyses results were multifaceted. In some cases, thermal models showed good agreement with debris condition, such as for the crew helmets. In other cases, geometry and protection from other structure proved too complex to model accurately. Further research is warranted to investigate other thermal mechanisms (shock interaction, combustion) to provide greater understanding of aerothermal effects on entry.

2.2 Orbiter Breakup Sequence

This section discusses the breakup sequence of the orbiter following the LOC. The CAIB concluded that the orbiter was shedding debris throughout entry.[1] LOS occurred at GMT 13:59:32. RGPC on-board data indicate that the Freon coolant loops in the PLBDs were still intact at GMT 14:00:04.8, indicating that the radiators and PLBDs were still intact. After GMT 14:00:04.8, no GPC data could be recovered. The on-board MADS/OEX recorder was powered until GMT 14:00:18, the time known as the CE. Based on video analysis, at GMT 14:00:18 the orbiter was no longer intact.[2] At this point in the entry, the orbiter was estimated to be at an altitude of 181,000 feet and traveling at Mach 15.

The SCSIIT relied upon analysis of ground-based video, analysis of debris recovery locations, ballistic analysis, and structural analysis to reconstruct the sequence of events experienced by *Columbia*. This section provides the framework for the sequence of structural failure and covers large-scale events that are related to orbiter structure and the separation of the forebody from the intact orbiter. Section 2.4 discusses events specifically related to the forebody breakup. These sections overlap in certain areas. Detailed description of the analysis techniques are contained in this section and are not repeated in Section 2.4.

The orbiter breakup sequence discussion is presented by type of analysis. The first is the *video analysis.* Analysis of ground-based videos consisted of reviewing all ground-based video of *Columbia*'s entry. Ground-based video analysis allowed a time-tag to be assigned to specific visual events that were seen in ground-based video. Considerable elements of the analysis were performed for the CAIB[3] although the SCSIIT performed additional analysis. On-board video is discussed in Sections 3.2 and 3.4.

The next analysis presented is the *ballistic analysis*, which used ground location, size, shape, and mass of recovered debris items. As an individual piece of debris was shed, it took on a unique ballistic trajectory based on its ballistic number (see Appendix). Heavier items with larger ballistic numbers travel farther downrange (in this case, east), while lighter items with smaller ballistic numbers decelerate quickly and achieve terminal velocity, traveling essentially straight down to the ground. Therefore, comparing the impact location alone of any one piece of debris relative to another will not provide information about the time or sequence of release; a detailed ballistic analysis comparing their trajectories is required. The modeled debris was iteratively connected to the reference trajectory (see Section 2.1) until the calculated ground impact longitude matched the actual recovered longitude of the item. This is a lengthy process and is heavily dependent on understanding the aerodynamic characteristics of the object in question. It is easiest to do this analysis on simple shapes, such as spheres and boxes. As a result, only select items have ballistic assessments. Conducting ballistics analysis on several debris items that came from the same general zone on the orbiter (such as the right wing, or aftbody) could provide a general release time for that specific structure. These times can be correlated with video events to produce a time-based breakup sequence.

The team elected to perform ballistics only on key items of debris and *cluster analysis* on the remaining objects. Cluster analysis assumes that when the recovery locations of a large number of debris items from the same structural element (e.g. tail, wings, payload bay, CM) are considered, the debris will have a similar range of ballistic numbers. Evaluating the centroid of clusters of structural elements *relative* to each other can provide a relative sequencing of key events. Specifically, because the centroid of the cluster for one

[1] *Columbia* Accident Investigation Board Report, Volume I, August 2003, p 12.
[2] *Columbia* Accident Investigation Board Report, Volume II, Appendix D.7, Working Scenario, October 2003, p. 209.
[3] *Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, October 2003.

structural element X is west of the centroid of recovered debris from structural element Y, X was released before Y. Cluster analysis of the debris field consisted of plotting the latitude and longitude of recovered debris items that originated from a specific location (such as one of the wings) on the orbiter, identifying the centroid of the debris cluster, and comparing that to the cluster centroid of items from a different source (such as the other wing). A cluster identifies a ground recovery zone or "footprint" of debris from a specific source.

The conclusion was that comparing the locations of these clusters does, in most cases, produce accurate results of *relative* sequencing. It does not provide a time-based sequence. The relative sequence may not exactly coincide with the ballistically determined time-based sequence, which is considered more accurate. The inherent uncertainty contained in the ballistic time sequence due to the reference trajectory assumptions is an error range of ±5 seconds. Other errors in the ballistic release times include the effects of cascading failures and the effects of aerodynamic lift, both of which bias release times later than the actual release time.

The *Columbia* Reconstruction Database (CRD) was critical to both the ballistic and the cluster analyses because it contains the records for ground recovery locations. When items were recovered in the field, typically their Global Positioning System (GPS) latitude and longitude were recorded. All recovered debris (approximately 84,000 pieces, or 39% of the orbiter[4]) were entered into the CRD, which is located at KSC. For details regarding the debris collection and processing, reference the *CAIB* Report, Volume II, Appendixes D.10 and D.11, and the Reconstruction Report.

Finally, *structural analysis* was based upon detailed engineering knowledge of the orbiter and forebody structures compared to the condition of the recovered debris. Given the knowledge gained from the video analysis, ballistic analysis, and subsequent cluster analyses, the overall structural analysis provided the most likely scenarios for failure modes of the orbiter.

The following is a summary of findings, conclusions, and recommendations from this section:

Finding. Ground-based video was a vital resource of data for understanding the accident, especially after telemetry was no longer available.

> ***Recommendation A11.*** All video segments within a compilation should be categorized and summarized. All videos should be re-reviewed once the investigation has progressed to the point that a timeline has been established to verify that all relevant video data are being used.
>
> ***Conclusion L3-2.*** The breakup of both *Challenger* and *Columbia* resulted in most of the X_o 582 ring frame bulkhead remaining with the crew module or forebody.
>
> ***Recommendation L3-1.*** Future vehicles should incorporate a design analysis for breakup to help guide design toward the most graceful degradation of the integrated vehicle systems and structure to maximize crew survival.

2.2.1 Ground-based video analysis

There was no NASA ground-based video imagery of the entry and breakup of *Columbia*. Shortly after the *Columbia* accident, NASA issued a request to the public to submit any photographic stills or videos taken of the vehicle's launch and entry that might aid in the mishap investigation. More than 170 videos and 1,500 stills as well as verbal accounts of the entry were submitted by the public to NASA. Most submitted video data had a variety of limitations for analysis, e.g., changing zoom factors and poor tracking. However, these videos were invaluable and contributed significantly to an understanding of the events of the last few minutes of *Columbia* and the crew.

[4]STS-107 *Columbia* Reconstruction Report, NSTS-60501, June 2003, p. 143.

The ISAG generated a master timeline that was comprised of two timelines; a "western" and an "eastern" timeline.[5] The western timeline spans the orbiter's trajectory from off the coast of California to New Mexico. A 2-minute-and-12-second gap in ground-based imagery coverage occurred as *Columbia* traveled from south of Santa Fe, New Mexico to Palo Pinto (just west of Dallas, Texas), GMT 13:57:31 to GMT 13:59:43. Video from the eastern timeline ends in Louisiana, where the engine powerheads impacted the ground.

Late in the SCSIIT investigation, the team discovered an additional video that provided useful information that had not been included in previous analyses because it was not classified as important initially. After the mishap timeline had been established, it was more evident that this video provided unique data for an important timeframe. This video, which is referred to as NBC (National Broadcasting Corporation) (EOC2-4-0076-B), adds several seconds of good-quality imagery to the eastern timeline and begins at GMT 13:59:32.5 (± 1 second), just after LOS at GMT 13:59:32. The gap in video coverage was reduced by 11 seconds. More importantly, this video provided insight into the LOC. The vantage point of the NBC video is directly under the flight path of the orbiter. Unfortunately, when the forebody separates after the CE, its trajectory is above other debris pieces, so this video adds little additional information after the CE.

Video frames shown in this section are taken from various videos that capture several events. Each frame represents 1/29.95 of a second. The frames shown come from compressed video; the frames are not representative of the higher-quality frames that were used for evaluation. The "times" shown in the frames reflect the time code synchronization protocol of the Society of Motion Picture and Television Engineers (SMPTE), not GMT times. Even in the few videos where the GMT code is displayed, the time is only as accurate as the time to which the camera was set. The GMT times reported below the video frames shown in this report have been time-synchronized and should be accurate to within approximately 1 second. The one exception is the Apache video. The source of Apache was a military helicopter and the time was GPS-based and, therefore, highly accurate (<0.3 second).

Errors associated with video event times are impacted by the magnification of the FOV, resolution, and viewing geometry. These factors can introduce some error into the timing. A combination of error sources can lead to an accuracy of ±1 second for defined events within a single video or between videos, although the actual error may be better or worse by up to an additional 1 second. This is more accurate than the ballistic trajectory times, which have a minimum error of ±5 seconds. As a result, the video analysis was used to determine the timing of key events.

Finding. Ground-based video was a vital resource of data for understanding the accident, especially after telemetry was no longer available.

> ***Recommendation A11.*** All video segments within a compilation should be categorized and summarized. All videos should be re-reviewed once the investigation has progressed to the point that a timeline has been established to verify that all relevant video data are being used.

2.2.1.1 *Significant events seen in the video*

Eastern timeline video coverage starts at GMT 13:39:32.5, immediately after LOS. There are four significant events for which there is video coverage: LOC, CE, CMCE, and Total Dispersal (TD). The CMCE and TD are discussed in Section 2.4. Significant debris shedding events are visible throughout the timeline.

Although SMPTE times that are seen in the frames are video-specific relative tape times, they can be correlated to GMT times if a visual event can be synchronized to a known GMT. The bright "dot" seen in the videos is the envelope of hot gases surrounding the orbiter during entry. Both the "dot" and the trail left by the orbiter are typical and expected visual signatures during entry. The apparent "star" around the orbiter is a lens flare, a common image effect due to the orbiter's brightness and the lens system of a camera. This "star" is seen in most videos.

[5] *Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, October 2003.

Loss of control

Figure 2.2-1 is the first frame of video for the eastern timeline. Figure 2.2-2 shows GMT 13:59:37, 135 frames (4.5 seconds) after figure 2.2-1. Figure 2.2-3, which shows 13 frames (approximately 0.5 second) after figure 2.2-2, correlates to LOC at GMT 13:59:37. The width of the trail increases at this time, which likely indicates a change in the flight condition. In addition to the change in the width of the trail, the trail appears to pulse or "corkscrew" over a period of less than 1 second (figure 2.2-4). This is consistent with the motion analysis, which suggests that the orbiter went into a flat spin.

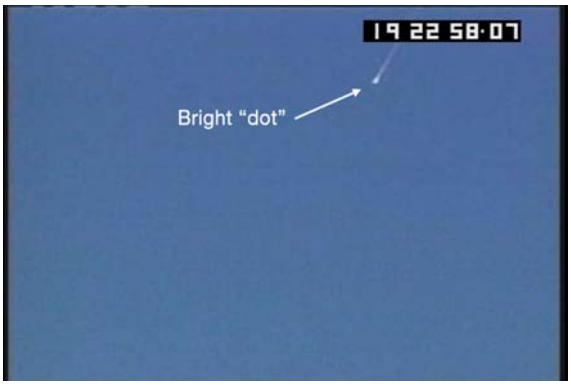


Figure 2.2-1. ***First frame of video of the Eastern timeline, GMT 13:59:32.5*****.**

Figure 2.2-2. ***GMT 13:59:37.00*****.**

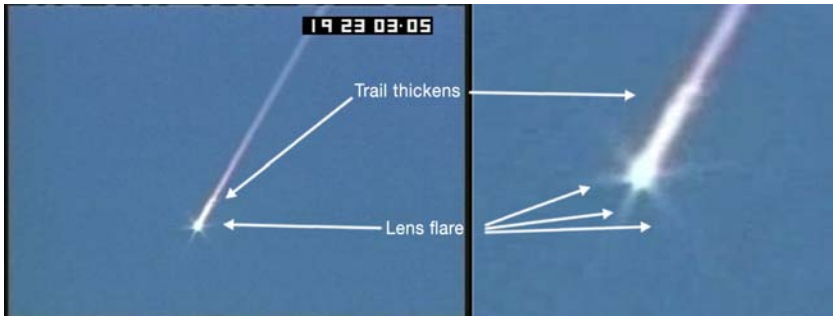


Figure 2.2-3. ***GMT 13:59:37.43, 13 frames after figure 2.2-2*****. Right image is a zoomed view of the left image.**

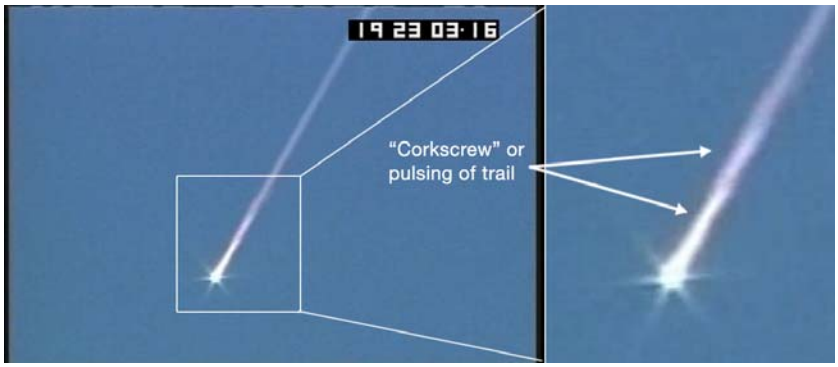


Figure 2.2-4. ***GMT 13:59:37.80*****.**

Catastrophic Event

The second significant event that is seen in video is the CE. Figure 2.2-5 shows the CE; there is no change in magnification between frames. The video of the CE is a distinct visual event in which the orbiter envelope brightens significantly and the trail width doubles. Additionally, some color change to orange is evident just prior to the CE. The color change is not readily visible in the still image frames that are taken from the video. In videos of the *Challenger* accident, an orange color was also seen in the videos and was assumed to be hydrazine, the material that is contained in the Orbital Maneuvering System (OMS) pods and RCS tanks. It is unknown whether the orange color seen in *Columbia*'s trail correlates to OMS or RCS tanks, or to some other cause.

Shortly after the brightening event and color change, a split trail is seen (figure 2.2-6). This strongly supports that the orbiter broke into multiple pieces at the time of the CE. The NBC video shows that the orbiter has an intermittent split trail prior to the CE. Prior to the CE, the split trails are tied to the separation of pieces of debris, and a singular main body trail continues after the separation of the debris. The split trail seen after the CE is constant; it does not dissipate and does not resolve back to a singular trail (figure 2.2-7).

Figure 2.2-5. ***The Catastrophic Event is depicted in these three frames of video that cover 0.1 second.*** **There is no change in the magnification/zoom factor. The third frame represents GMT 14:00:18.3.**

Figure 2.2-6. ***Split trail after the Catastrophic Event*****. [GMT 14:00:18.6]**

Figure 2.2-7. ***Persistence of the split trail 23 seconds after the Catastrophic Event*****. [GMT 14:00:41.1]**

Due to camera effects of saturation of pixels, magnification settings, distance to the orbiter, and angle of view of the orbiter, visual separation of the major pieces of the orbiter is not identifiable until 8 seconds after the CE. The visual separation of the orbiter into separate identifiable pieces is first made evident by the appearance of a second "star" beside the first "star." Figure 2.2-8 uses reversed color ("inverted") images of the orbiter to more readily show the "star" (lens flares). The dark lines on the images are shown in an effort to illustrate the lens flares.

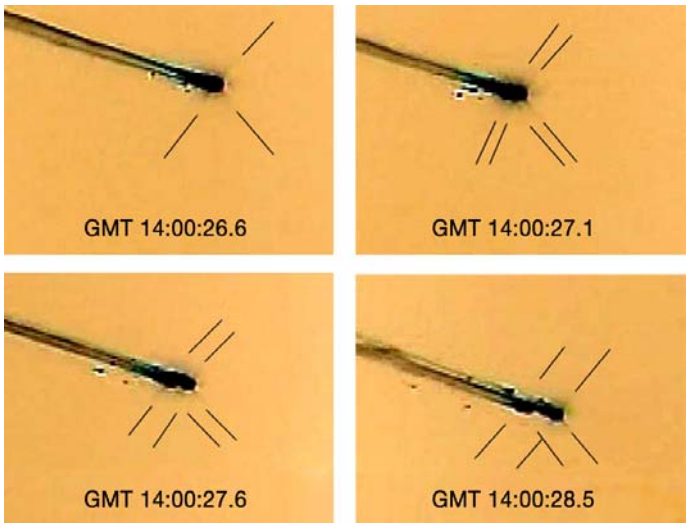


Figure 2.2-8. ***Color-inverted video images of the start of the double star event (GMT 14:00:26.6).*** **Black lines have been added to more clearly identify the two separate objects.**

The forebody breakup was identified at GMT 14:00:53 (see Section 2.4).

Other Major Structure Identification

Once the key events had been identified, an effort was made to tie specific debris shedding events with specific recovered debris. In particular, the team wanted to identify the forebody in the video. Heavier objects have a higher ballistic number and slower deceleration than lighter objects. Heavy objects, therefore, would be visible longer than other objects and could possibly be identified by their distinct trajectory in the video compared to other objects. The aftbody is the heaviest part of the orbiter, followed by the CM. The next heaviest item was SPACEHAB, which was located in the payload bay.

Six entry videos were found to offer the most information for debris shedding identification (Table 2.2-1). Videos are referred to by the name listed in the table. Figure 2.2-9 shows the debris tree created from these videos. The end time for a debris piece generally indicates when the object left the FOV of the video rather than the object breaking into pieces too small to be seen by the camera.

Table 2.2-1. ***Videos Used to Create the Debris Tree in Figure 2.2-9***

Reference EOC No.	City in Texas	Name	Latitude	Longitude
EOC2-4-0024	Arlington	Arlington	32.7	–97.1
EOC2-4-0209-B	Hewitt	Hewitt	31.4	–97.2
EOC2-4-0221-4	Mesquite	WFAA4/Mesquite	32.8	–96.6
EOC2-4-0221-3	Fairpark	WFAA3	32.8	–96.7
MIT-DVCAM	Fort Hood	Apache	31.2	–97.6
EOC2-4-0077	Burleson	NBC	32.5	–97.3

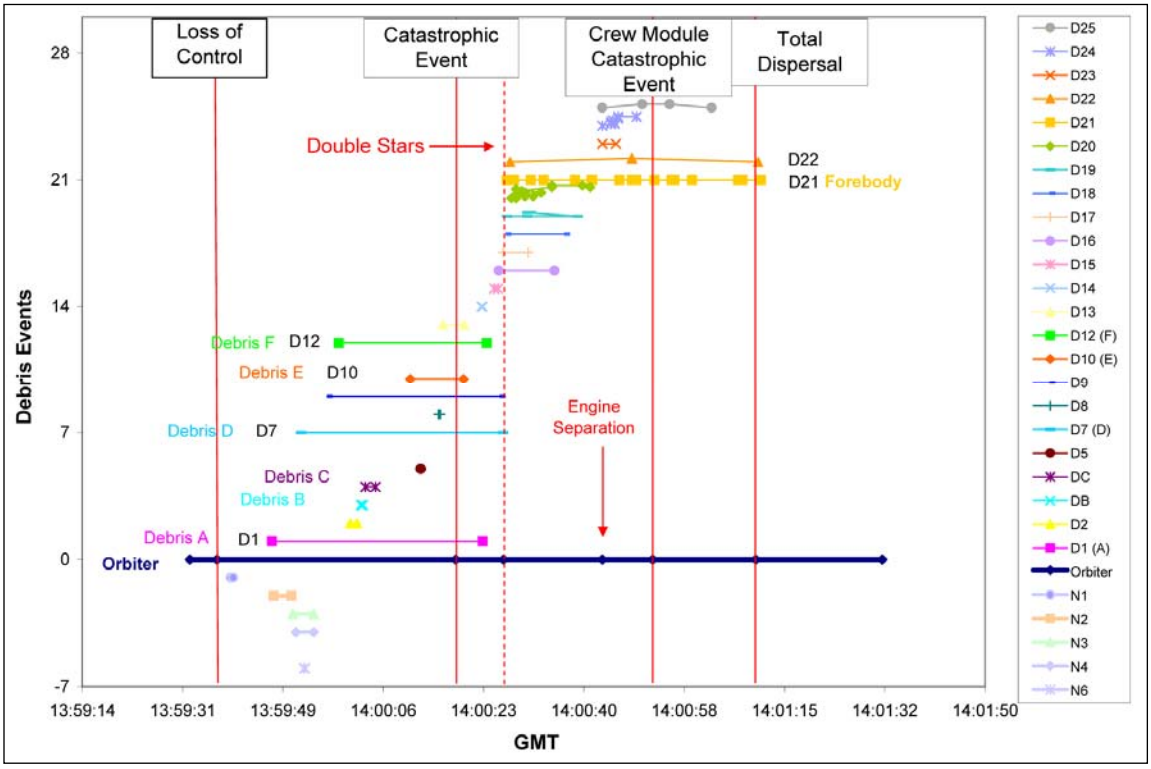


Figure 2.2-9. *Debris tree*.

The points that are labeled Debris A, B, etc. are the names that correlate to the master timeline provided to the CAIB.[6] These were the starting point for the SCSIIT video analysis. When other videos were reviewed, they received a different numbering system. For example, debris shedding events that were identified in Hewitt were labeled D1, D2, D3, etc. In the NBC video, debris events were labeled N1, N2, N3, etc. When an object could be positively correlated with the originally identified debris events (A, B, C, etc), they were merged into that line, showing the full range across all videos for how long a specific debris object was visible in any video. If an object could not be clearly correlated with another video, it was kept separate in the debris tree and labeled accordingly.

The points that are seen on the various horizontal lines typically represent a significant piece separating from that debris piece. In some cases, these significant pieces are graphed with the parent piece. All the major events are marked on the timeline. Debris D through F and Debris 8 through 12 might have some overlap as objects but, due to different angles, were difficult to correlate.

Two pieces of debris (Debris A and, shortly afterward, Debris D) are seen in the NBC video (and some other videos) well before the CE. Debris A "flashes," brightening and completely disappearing repeatedly, which is suggestive of tumbling. At times, the track and movement of the piece possibly suggests that it generates lift. At one point, the single flashing debris disappears and multiple smaller flashing debris appear, suggesting that Debris A broke into pieces. Debris D, however, paces the orbiter, suggesting that it is an object with significant mass and a high ballistic number. Given that the left wing and OMS pod were known to be structurally degrading in this timeframe, Debris A may possibly be the left OMS pod cover or a piece of the left wing, and Debris D may possibly be the left OMS pod, which is fairly heavy and substantial.

[6] *Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, October 2003.

The aftbody is a part of the line labeled "orbiter" in the figure, since it was traced back to the intact orbiter and was the most persistent piece of the orbiter after the CE. It was easily identifiable because the engines separate from it late in the video.

Debris item 21 (D21) is identified as the forebody. This identification is based upon the mass of the forebody and trajectory in the video. Ballistic and debris field evidence later confirmed this assessment.

The points seen on the debris item 21 line represent all events that could be correlated to more than one video. Because of various camera viewing angles, correlation of debris pieces was difficult. Cross-correlation with three other videos showed that brightening events could be seen in many videos while few specific debris events could be cross-correlated.

The last time at which the main engines visually appeared to be a single unit is GMT 14:00:50.6; this is defined as the starting point for engine separation. Figure 2.2-10 shows the engines at GMT 14:00:53, the three pieces below the forebody which are circled in green. Their separation is already well under way. Considering that visual separation of the orbiter pieces after the CE took about 10 seconds, the actual time of separation of the engine components from one another may have been much earlier than GMT 14:00:50.6. Available data do not allow for a more accurate time of separation.

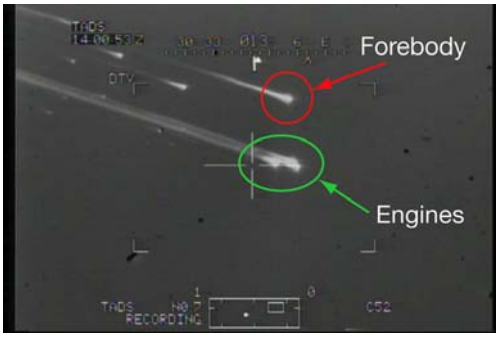


Figure 2.2-10. ***Clear visual separation of the three main engines*****.**

In summary, specific debris shedding events were very difficult to tie to specific recovered debris with a few exceptions. The forebody was positively identified, as was the aft section/engines. Debris A is possibly the left OMS pod cover. Debris D is possibly the left OMS pod. Debris D22 is suspected to be the SPACEHAB. The sequence, as derived from the video analysis, is presented in Table 2.2-2. It should be noted that the OMS pod time represents a time at which the debris piece is already clearly separated from the orbiter. The actual separation time could not be identified. Both forebody and (potentially) SPACEHAB are identified as separating from the rest of the orbiter at the CE. The times shown below indicate the video-based time of structural breakup.

Table 2.2-2. ***Orbiter Breakup Event Sequence and Times Determined by Video***

Event	GMT
LOC	13:59:37
OMS pod cover	13:59:46
OMS pod	13:59:51
Orbiter breakup (CE)	14:00:18
SPACEHAB breakup (D22)	14:00:48
Main engines separate	14:00:50
Forebody breakup (D21)	14:00:53

2.2.2 Ballistic analysis

Ballistic analyses were conducted on selected recovered debris objects to help define the orbiter breakup sequence. The ballistic analysis determined an object's approximate time of release from the vehicle given certain initial conditions. It should be noted that this type of analysis does not account for serial (cascading) debris events where a large object is released on its own trajectory and the object then breaks up into multiple smaller objects. Therefore, assessing an individual object has inherent uncertainty. For objects that left the vehicle in almost the same configuration as they were recovered (i.e., an individual item of crew equipment), the ballistic numbers and times of release have a higher confidence level. However, conducting ballistics

analysis on several debris items from a specific zone of origin on the orbiter can provide a general release time for that specific structure.

The process is called "ballistic analysis" because all the objects are assumed to become ballistic upon release. The trajectory of a ballistic object is unique to the properties of that object and is not controlled by power or directional steering. A good example would be a ball fired from a cannon. The cannonball's trajectory is determined by the forces acting on it: its momentum (mass × velocity), the drag from the atmosphere, Earth's gravity, and winds (figure 2.2-11).

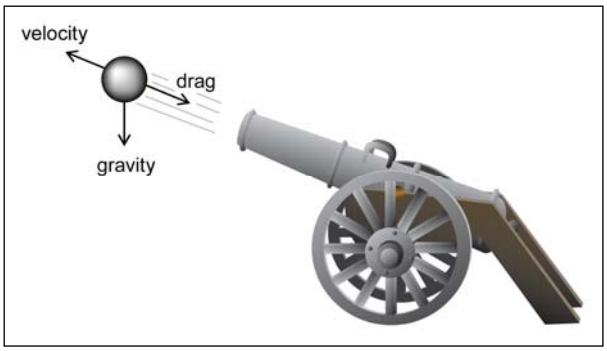


Figure 2.2-11. ***A cannonball is an example of a ballistic object.***

The modeling tools used in the ballistic analysis were Snewt and SORT (see Chapter 4).

Objects were selected for ballistic analysis based upon weight, shape, original location on the orbiter, and possible relevance to the breakup sequence. For each debris object, a reference area, reference length, and c.g. location were specified. The c.g. location of some analyzed debris items was already known. For others, the c.g. location was measured from the debris recovered, or the c.g. location was estimated. The aerodynamic coefficients were generated at different orientations through an alpha (angle-of-attack) and beta (sideslip angle) sweep or at a significant number of random orientations. The ballistic number can be determined using aerodynamic coefficients. It is possible to average the ballistic number of the various orientations to come up with an average ballistic number. The more "regularly" shaped the object is (sphere, cube, rectangular prism, flat plate, etc.), the more precisely the average ballistic number can be computed. For most of the objects analyzed, the average ballistic number was used because it was assumed that the objects were tumbling and, therefore, were constantly changing their orientation. Very few objects had directional thermal erosion, indicating that they had aerodynamically trimmed in a specific orientation. For these objects, the ballistic number for that orientation was used. The average ballistic number assumption can introduce errors in both directions for release time because the actual ballistic number may have been either larger or smaller than the assumed average.

After the aerodynamic coefficients were calculated, the object's approximate release time could be determined. As previously stated, there is an error bar of ±5 seconds in release times due to the uncertainty in the reference trajectory used as a baseline for the item to be released from.

Release times for large structures were determined using the estimated release time of their subcomponents. However, these estimated times do not take into consideration cascading failures of these large objects. The recovered debris objects were separated into categories of large orbiter structure, such as left wing, OMS pod, forward bulkhead, aft bulkhead, etc. Each category of objects recovered on the ground made a cluster along the ground track. It was first thought that objects in the westernmost part of a given cluster would have the lower ballistic numbers, and objects to the east would have the larger ballistic numbers. However, it was found from examining the database of recovered items that each debris cluster included many low-ballistic-numbered objects, meaning that cascading failure was common. Items may in fact be part of a larger object that separated earlier, for which a trajectory is not known. Only the trajectory of the individual item can be computed. Cascading events generally result in calculated release times for individual items that are later than the actual release times.

Another key assumption in the ballistic analysis was that the analyzed objects did not have lift. To determine whether an object could have generated lift, the shape and especially the orientation of the object during flight has to be known. Since that knowledge was not available, drag-only or ballistic flight was assumed. In general, an object with lift will travel a greater downrange distance than that same object on a purely ballistic trajectory, causing an error in release time that is biased to a later time than actual release.

Since cascading failures and lift would send an object farther downrange (east) than expected from a simple ballistic trajectory, a decision was made to use items from the westernmost (earliest) edge of each cluster and to use easily modeled items (such as flat plates) when possible to estimate the release time of the large orbiter structure. A conscious effort was made to choose items such as structural skin rather than tiles or reinforced carbon-carbon (RCC), which were continually shed during the entire entry. If three or more debris items were found that gave the same estimated release time, confidence was increased that the large orbiter structure came off at that time. The estimated time of release of large structures is referred to as the major structural release time.

For further information on ballistics, see Appendix A.

2.2.2.1 *Major structural release times*

The estimated release time is the calculated time at which an object was released from the reference trajectory, given the previous assumptions and conditions. Table 2.2-3 shows the estimated release times for large orbiter structures. After the CE, the vehicle was no longer intact and was in at least three major pieces (forebody, midbody/right wing, and aftbody). Some of the subsequent release times reflect separation from those major pieces, not the intact orbiter.

Table 2.2-3. *Release Times of Large Structures*

Major Structural Release Time (GMT)	Vehicle Structure	Time of Release (GMT) and Debris Object Number[7]
13:59:48	Left OMS Pod (skin, structure, honeycomb)	13:59:48 (78899) 13:59:57 (85446) 14:00:16 (84132)
14:00:04	Left Wing (RCC, upper and lower wing skin)	14:00:02 (70391) 14:00:05 (81331) 14:00:05 (11525)
CE		
14:00:26	Midbody Fuselage	14:00:20 (82427) 14:00:33 (82172) 14:00:35 (38315)
14:00:27	Right Wing (RCC, skin, structure)	14:00:24 (68702) 14:00:26 (24508) 14:00:28 (8172) 14:00:30 (49833)
14:00:31	SPACEHAB	14:00:29 (65045) 14:00:33 (7641) 14:00:57 (22900)

[7]The debris object numbers are the numbers that are assigned to each recovered object when its recovered location, description, weight, dimensions, etc. was entered into the *Columbia* Reconstruction Database for tracking and identification.

Table 2.2-3. *Release Times of Large Structures* (Continued)

Major Structural Release Time (GMT)	Vehicle Structure	Time of Release (GMT) and Debris Object Number[8]
14:00:34	Vertical Tail (structure, drag chute panel)	14:00:26 (26078) 14:00:27 (77800) 14:00:38 (85279) 14:00:39 (1633) 14:00:40 (45837) 14:00:48 (52092)
14:00:34	Aft Fuselage (Carrier Panel, structure)	14:00:32 (43091) 14:00:33 (31308) 14:00:36 (79178)
14:00:36	Internal Airlock soft stowage items	14:00:36 (65900) 14:00:36 (31297)
14:00:43	Tunnel Adapter structure	14:00:42 (64966) 14:00:44 (69606)
14:00:54	Aftbody forward bulkhead (X_o 1307)	14:00:52 (12877) 14:00:52 (83678) 14:00:53 (14559) 14:00:57 (64156)

Large structural pieces, such as the left OMS pod or left wing, were not released from the orbiter at a single time as is implied by the major structural release time. The objects in this section should not be considered as being released from a single complete orbiter, but from (at minimum) three separate objects. Table 2.2-4 compares the sequence suggested by video against the sequence suggested by ballistic assessment.

Table 2.2-4. *Comparison of Video and Ballistic Sequence*

Video Sequence (relative)	Ballistic Sequence (time-based)
OMS Pod	OMS Pod
	Left Wing
Forebody/Midbody/Aftbody (CE)	Midbody
	Right Wing
SPACEHAB	SPACEHAB
	Vertical Tail
	Aftbody
Main Engines	Main Engines
Forebody	Forebody

The differences between the two sequences can be explained by cascading events where major portions of the vehicle, such as the aftbody and the forebody, remained intact for some time. One conclusion from this comparison is that the relatively lighter components making up the midbody, including the wings and payload bay, disintegrated fairly rapidly and as individual objects while the aftbody, SPACEHAB, and the forebody maintained some integrity.

[8]The debris object numbers are the numbers that are assigned to each recovered object when its recovered location, description, weight, dimensions, etc. was entered into the *Columbia* Reconstruction Database for tracking and identification.

2.2.3 Cluster analysis

This section addresses the cluster analysis that shows how the pattern of recovered debris aided in understanding the sequence of the orbiter breakup.

An evaluation of the debris field was conducted to match debris sources (orbiter structural zone) with debris recovery locations. In all cases for the debris maps that follow, the orbiter travels from approximately the upper left corner (northwest) to approximately the lower right corner (southeast). Although debris was seen falling away from *Columbia* in ground-based video as far west in its trajectory as off the coast of California, the farthest west confirmed *Columbia* debris was recovered in Texas.

The location of debris on the ground is influenced by when the debris separated from the orbiter and how far the object traveled downrange. As a general rule, heavier objects travel farther downrange (to the east, in this case) than lighter items of similar shape and size. In addition, wind effects and, possibly, lateral forces exerted during orbiter breakup had some effect as evidenced by the lateral dispersion of objects.

The center of a large debris group was considered reasonably accurate for relative sequencing, with ballistics allowing for refinement and correction, if needed. Cascading debris failures and lift generation will result in biasing the debris cluster farther east (artificially elongated at the trailing end) and greater lateral spread for the same reasons as the effects on the ballistics analysis. Wind, which at the time of the mishap was prevailing from the southwest, can also affect the debris field cluster shape and centroid.

However, cross-referencing ballistic release times across a range of clusters showed that as few as five objects in a debris group can allow some conclusions to be drawn *if the objects have approximately the same range of ballistic numbers*. When five or more items were available to determine a cluster, there was fairly high confidence in comparing that cluster to another.

Knowledge of the topographic conditions was important in some instances, where water hazards or thick brush interfered with debris recovery efforts. Some graphs show that fewer debris items were recovered in certain areas, suggesting a decrease of debris when, in fact, the debris field thinning is related to field conditions and obstacles that impeded the ability to locate the debris items.

2.2.3.1 *Debris clusters*

The orbiter is made up of multiple structural elements (figure 2.2-12). The recovery location data of the debris were divided into many of these structural elements to determine the apparent sequence in which the orbiter failed.

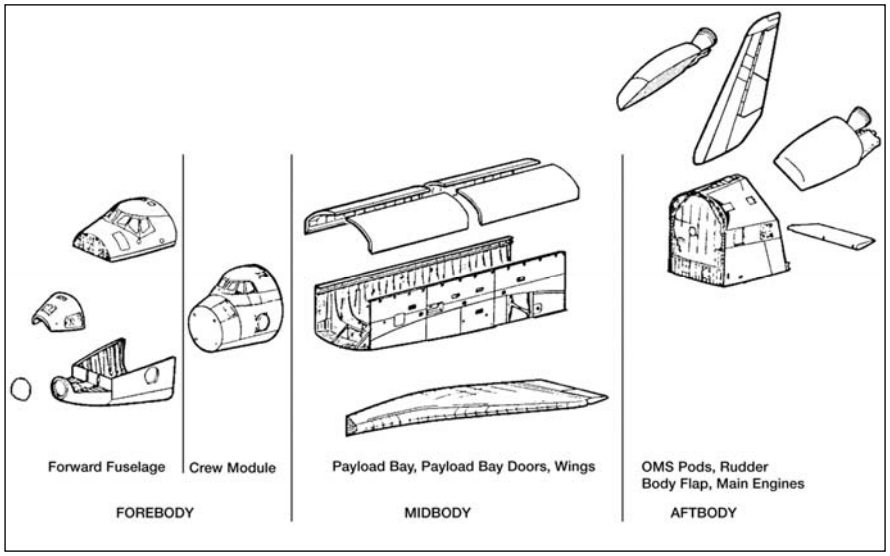


Figure 2.2-12. ***Depiction of the orbiter forebody, midbody, and aftbody elements*****.**

First, recovered debris longitude data were sorted by respective structural sources. Then, each item became a data point placed into longitude "bins," each having a range of 0.5 degree. Each bin was then graphed by what percent of the recovered debris fell into what longitude bin. The numbers of debris items in a group ranged from fewer than 100 into the thousands. One bias introduced is that the data were not corrected for these differences in total number when comparing datasets; only the percent of the total recovered from that zone is compared. It is unknown how this difference affects the data overall.

Details of each debris field cluster are presented in the following sections.

Wings

Figure 2.2-12 identifies where the wing structure is in relationship to the other orbiter structural components. TPS tile is included in these figures, as some tile adhered to structure and it was not easily possible to separate out individual tiles. Figure 2.2-13 shows a ground debris map that is similar to that produced in the CAIB Report, Volume I, p. 75; however, figure 2.2-13 shows all the debris for the wings whereas the CAIB map shows only the RCC panels. As the CAIB reported, the left wing failed before the right wing. This is implied in figure 2.2-14 as well. Overlay mapping of debris at their correct latitudes and longitudes prevents easy recognition of their actual ground footprints. The bars shown on the map and in later ground debris maps represent the main cluster of the debris. If coverage extends beyond the map shown, an arrow indicates that the footprint continues off the map. The bar shown above each histogram indicates the longitude range covered by the overlay mapping.

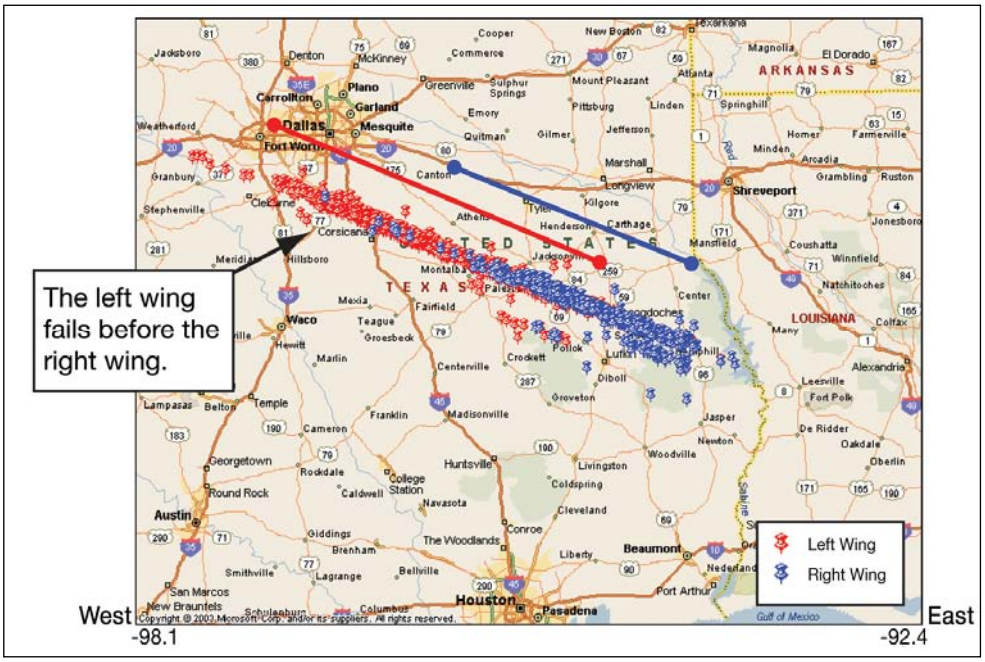


Figure 2.2-13. ***Debris field of the left and right wings.***

Tail (vertical stabilizer)

Figure 2.2-14 illustrates the debris of the tail section in relation to the wing debris. The bulk of the tail debris impacted west of the bulk of the right wing debris. Figure 2.2-15, which is a histogram of longitude vs. the number of debris items, shows this as well. The lines above the histogram indicate the approximate area of the histogram that the debris map covers. Figure 2.2-15 suggests that the tail possibly began to depart between the left wing and the right wing.

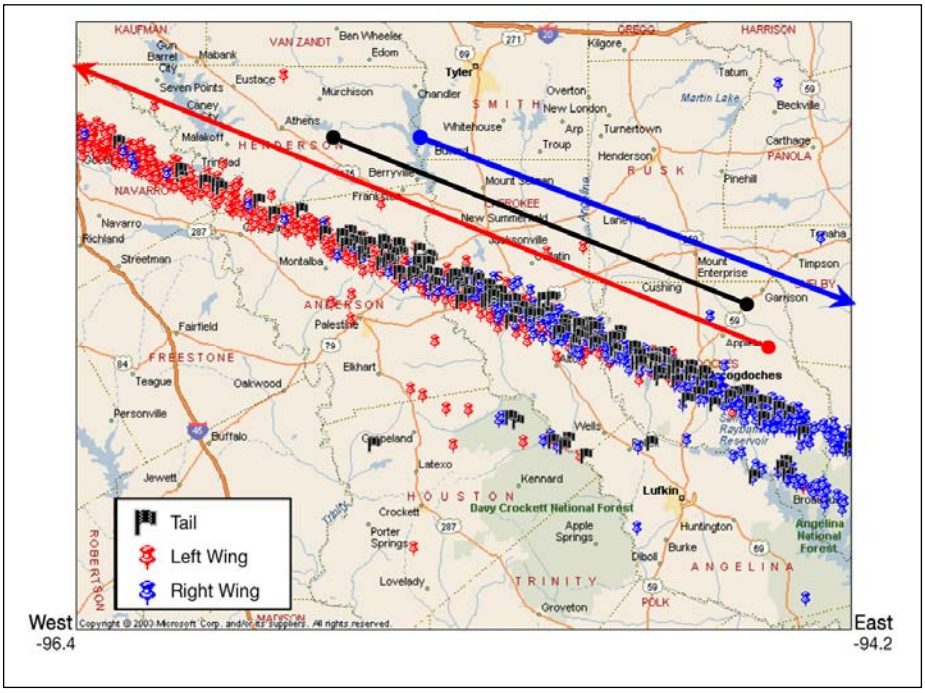


Figure 2.2-14. ***Partial debris field of the left wing, right wing, and tail.***

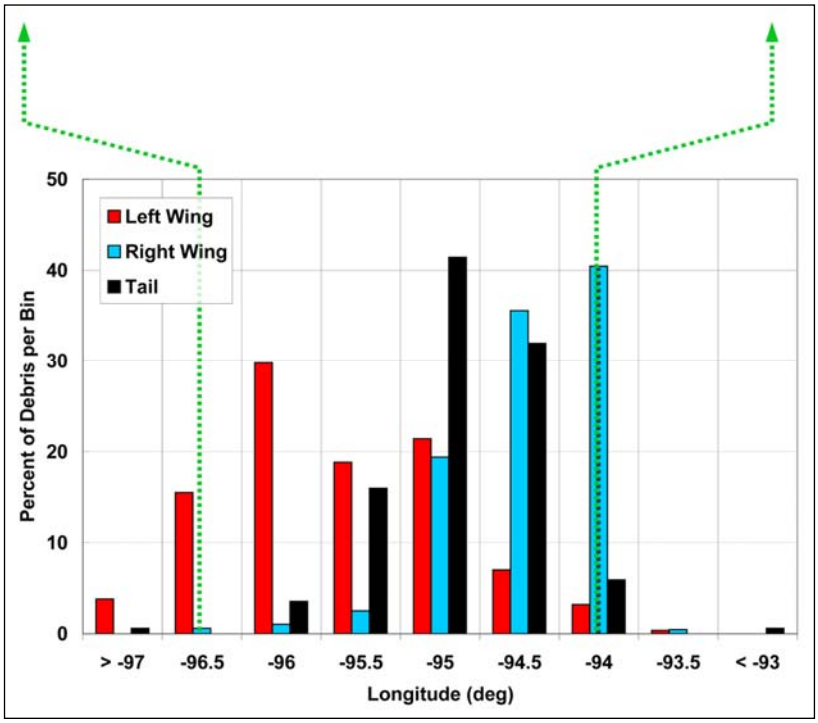


Figure 2.2-15. ***Histogram of the left wing, right wing, and tail debris.*** **The green dotted lines are the projection onto the histogram of the longitude range shown in the map.**

Midbody

Figure 2.2-12 identifies where the midbody is related to the other orbiter structural components. The midbody consisted of the mid fuselage, payload bay, and PLBDs (SPACEHAB is not included in this chart). The midbody began losing structure and shedding objects after the left wing but prior to the tail (figures 2.2-16 and 2.2-17). The midbody debris shown is comprised predominantly of PLBD structure, although pieces of the midbody are scattered throughout the entire debris field.

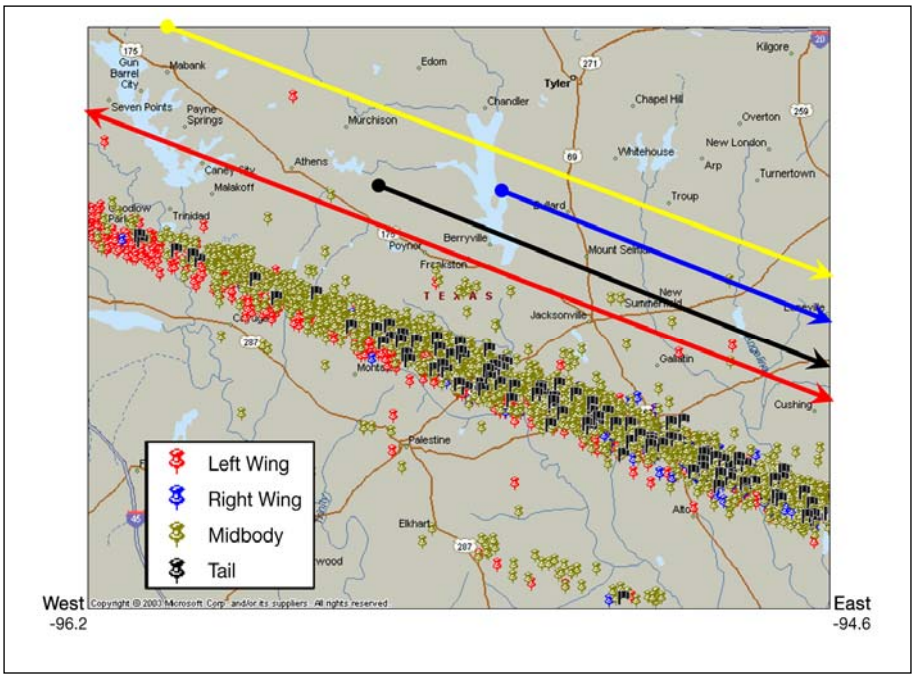


Figure 2.2-16. ***Partial debris field of the left wing, right wing, midbody, and tail.***

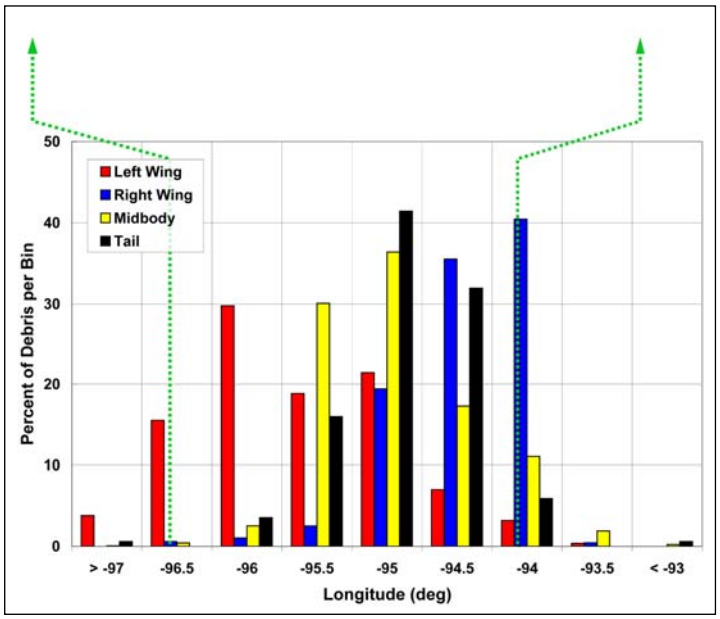


Figure 2.2-17. ***Histogram of the percentage of debris found vs. longitude for the left wing, right wing, midbody, and tail.*** **The green dotted lines are the projection onto the histogram of the longitude range shown in the map.**

Figure 2.2-18 shows the on-ground sequence of the midbody structural debris with right (starboard) and left (port) midbody origins marked where known. Beside the chart is a schematic of the midbody with some positions marked in X_o coordinates for reference. The X_o terminology refers to the X position in inches in the orbiter coordinate frame, where the X axis runs the length of the orbiter from fore to aft in the orbiter X-axis measurements. The nose of the orbiter is at X_o 236″ as the coordinate system extends beyond the nose. X_o 582″ is the demarcation between the forebody and the midbody. The port lower wing chine and midbody floor skin stretches from X_o 582″ to X_o 807″ (just forward of the SPACEHAB location). The wing attachment region spans X_o 807″ to X_o 1365″. The error bars shown on some pieces indicate the range of X_o positions that a piece of debris covered or could originate from, if the range was noted in the database.

The westernmost piece of ground debris from the midbody structure with a known X_o location was a port location just aft of SPACEHAB. However, this appears to be an outlier data point. Figure 2.2-18 indicates that based upon recovered midbody structure, the midbody first suffered a major failure ranging from about X_o 582″ to X_o 919″.

The next grouping of debris predominantly originates from X_o 776″ (towards the aft of SPACEHAB) to aft of the Fast Reaction Experiments Enabling Science, Technology, Applications, and Research (FREESTAR) payload near X_o 1124″, although there are a few pieces near the forebody bulkhead (X_o 582″).

The final cluster is from the aft region of the midbody. As debris originating from near the midbody aft bulkhead (X_o 1307″) appears farthest east in the debris field, this suggests that some midbody payload bay structure remained attached to the aftbody.

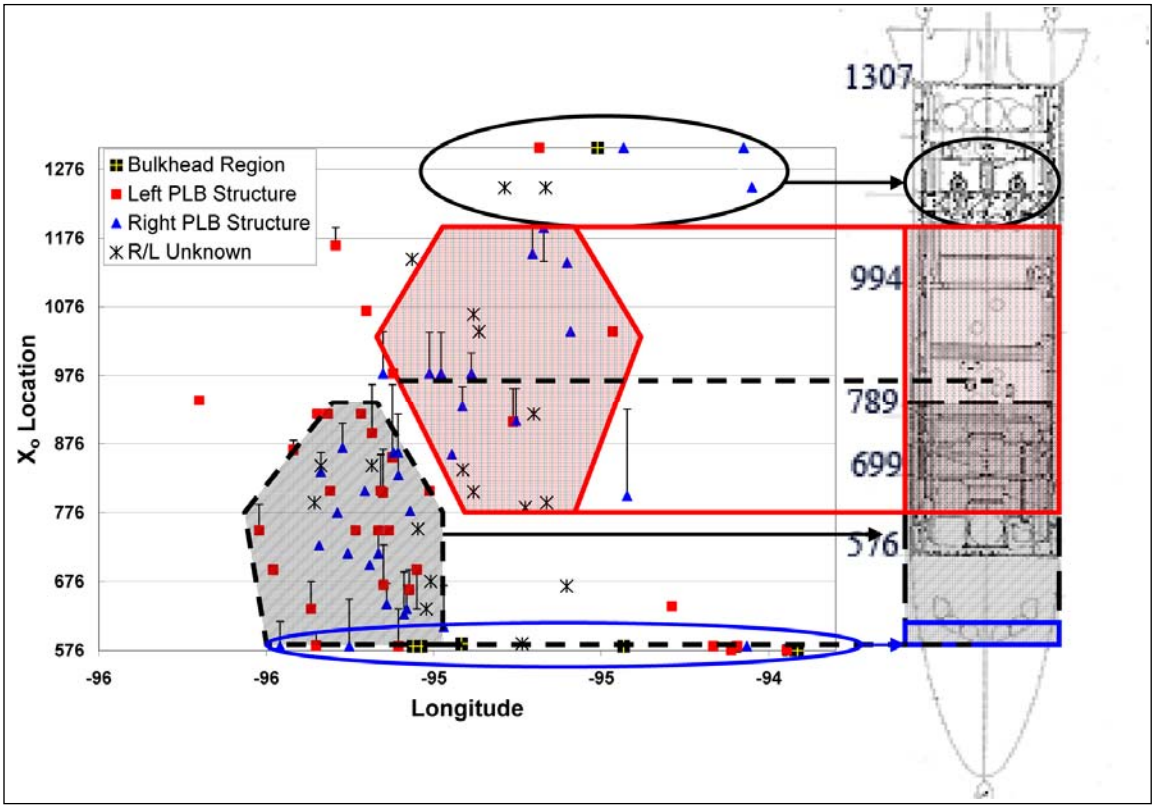


Figure 2.2-18. ***Longitude vs. the X_o origin of the structural debris of the mid fuselage. To the right is a drawing of the mid fuselage with key X_o positions marked.***

The data in figure 2.2-18 cannot be used to determine a port or starboard initiation of breakup of the midbody. Size of individual items was also examined with the assumption that ballistic numbers of objects of roughly the same size and shape will land in the order in which they separate from the orbiter. The number

of objects is small, so this may be an invalid assumption. For smaller-sized midbody debris (≤500 in^2), port debris appears west of starboard debris. For larger-sized debris (≥1000 in^2), starboard debris appears west of port debris. Overall, it is indeterminate whether the port or starboard side failed first based upon ground debris.

SPACEHAB debris was also evaluated (not shown here). SPACEHAB structural debris is concentrated near the forebody debris field, suggesting that the element maintained structural integrity for some period of time as did the forebody.

The midbody *structural* debris (vs. objects contained within the midbody such as SPACEHAB) and the right wing debris fields increase in density at the same time. As shown above, the port and starboard midbody structural debris originating from the same X_o locations begin appearing in the same longitude range (95.5 to 95W). However, in the next longitude range sector (95 to 94.5W), predominantly starboard pieces were recovered. This is the same longitude range sector where the largest number of right wing debris items was recovered. This suggests that a portion of midbody structure and right wing may possibly have initially remained together as a unit when *Columbia* experienced the CE.

Aftbody

The midbody ground debris analysis indicates that at least some of the midbody may have remained with the aftbody. As observed on video, the main engines (the heaviest individual components of the orbiter, with a correspondingly high ballistic number) remained a cohesive unit until GMT 14:00:50.6. Aftbody shedding did occur throughout the period following the CE. Main engine pieces, which were among the easternmost debris objects recovered in the debris field, were found in Fort Polk, Louisiana. Figures 2.2-19 and 2.2-20 show the distribution of the debris field of the aftbody and the left and right wings relative to each other.

Forebody

The forebody breakup sequence is discussed in Section 2.4. The forebody is composed of the FF, CM, nose cap, nose landing gear, and forward RCS.

The debris field shows that the CM and FF failed at nearly the same time, with their debris footprints overlaid on one another. This supports that the forebody remained an integral unit until structural failure, which began at the CMCE. The debris field showed that about 87% of the forebody structural debris appears suddenly; degradation may have occurred, but it was minor until the start of total forebody structural failure. Video corroborates this, showing only minor intermittent debris loss from the forebody until its failure.

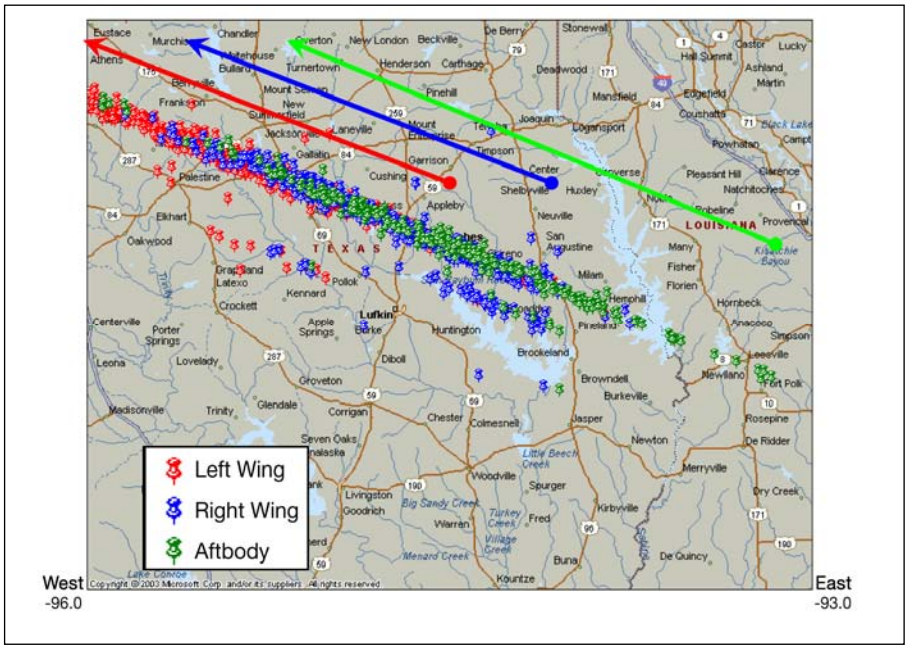


Figure 2.2-19. ***Partial debris field of the left wing, right wing, and aftbody*****.**

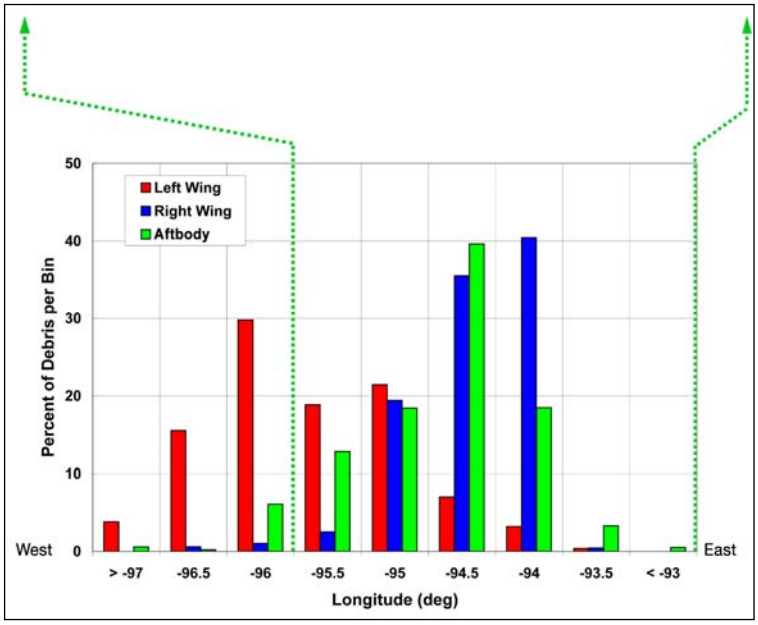


Figure 2.2-20. ***Histogram of debris of the wings and aftbody (including orbiter main engines)*****. The green dotted lines are the projection onto the histogram of the longitude range shown in the map.**

In summary, figure 2.2-21 shows an overall assessment of debris clusters. The westernmost longitude bin of each group that exceeded 30% of the recovered debris group is referenced with an arrow; the label above the bin identifies the debris group. It was observed that when the number of recovered debris items from one orbiter structural zone reached approximately 30%, the next group began to appear in rapidly increasing numbers. The significance of this observation is not known. It may be an artifact of data processing, or it may be characteristic of a cascade effect of structural breakup. This is a topic that may warrant further research and investigation.

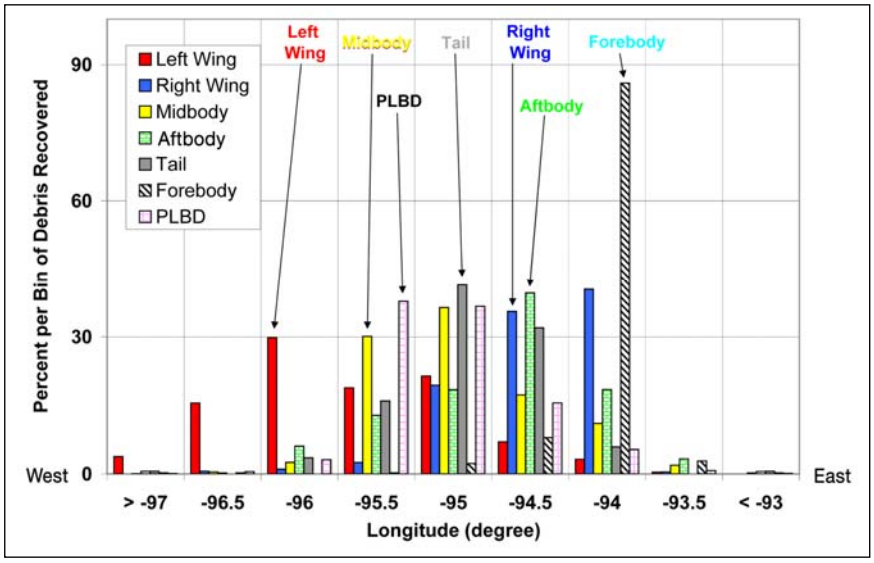


Figure 2.2-21. ***Histogram of the relative order of the major debris groups recovered.***

Cluster analyses show that following the left wing, midbody items (predominantly PLBD pieces) departed the vehicle. The tail, the right wing, and the aftbody shed a significant number of pieces, followed by SPACEHAB and the forebody. Table 2.2-5 shows a comparison of sequencing from cluster analysis compared to video and ballistic analyses.

Table 2.2-5. ***Comparison of Orbiter Breakup Sequence from Video, Ballistic, and Cluster Analysis***

Video Sequence (relative)	Ballistic Sequence (time-based)	Cluster Sequence (relative)
OMS Pod or Left Wing	OMS Pod	Left Wing
	Left Wing	
Forebody/Midbody/Aftbody separation (CE)	Midbody/PLBDs	Midbody
	Right Wing	Vertical Tail
SPACEHAB	SPACEHAB	
	Vertical Tail	Right Wing
	Aftbody breakup	Aftbody breakup
Main Engines separation	Main Engines separate	
Forebody breakup	Forebody breakup	SPACEHAB/Forebody breakup

This relative sequence of cluster-based groups does not match the ballistics reported time-based sequence (Table 2.2-3) of a few debris groups. Specifically, ballistics shows the tail failing before the right wing. This is probably the result of a cascading failure. Based on video, ballistic, and cluster analysis, the conclusion was that the left OMS pod and left wing departed first. It is possible that the PLBDs departed next. At the CE, the orbiter appeared to separate into three main components: forebody, midbody including the right wing, and aftbody. The midbody/right wing and aftbody failed in cascade following the primary breakup event. The forebody failed last.

2.2.4 Structural analysis

2.2.4.1 *Background*

This section provides a discussion of an analysis and scenario for the orbiter breakup. In addition to the data presented in previous sections, this analysis also draws on the orbiter's design strengths and weaknesses.

This structural assessment predominantly focuses on events leading up to the CE and the forebody release at the CE, since that was the area of most interest to the team. Little to no assessment was done by the SCSIIT structures team on the wings or aftbody since they were not considered relevant enough to this study to warrant the resources.

There are difficulties in understanding and supporting the various theories concerning the orbiter vehicle breakup. Except for the left wing, the vehicle was essentially structurally intact with most systems functioning normally at the time of the last telemetry received (GMT 14:00:05). No recorded on-board data were recovered after GMT 14:00:19, 1 second after the orbiter breakup began. Ground-based video footage does not provide adequate detail to determine the exact sequence of events other than as previously outlined.

Some background regarding the orbiter structure is necessary for this discussion. Orbiter structure is mainly constructed of aluminum components such as riveted skins and stringers, integrally machined plates, honeycomb sandwich panels, frames, bulkheads, and trusses. The exceptions are the OMS pod skins and PLBDs, which are made from graphite epoxy honeycomb sandwich panels, and the aft fuselage thrust structure, which is diffusion-bonded titanium that is reinforced with a boron-epoxy laminate. The payload bay is bounded by a forward bulkhead, the X_o 582 ring frame bulkhead (adjacent to the aft bulkhead of the CM), and an aft bulkhead (located at X_o 1307).

As detailed in figure 2.2-22, the FF is a semi-monocoque structure consisting of aluminum (2024) skins, stringers, longerons, bulkheads, and frames. Its structural purpose is to withstand the loads from the nose landing gear, CM, aerodynamic loads, and CM venting pressure. It also supports the associated portion of the TPS. The forward RCS is considered an integral part of the FF. The FF skin panels were designed primarily based on stiffness requirements to minimize local skin deflection to avoid the cracking or loss of the thermal protection tiles.

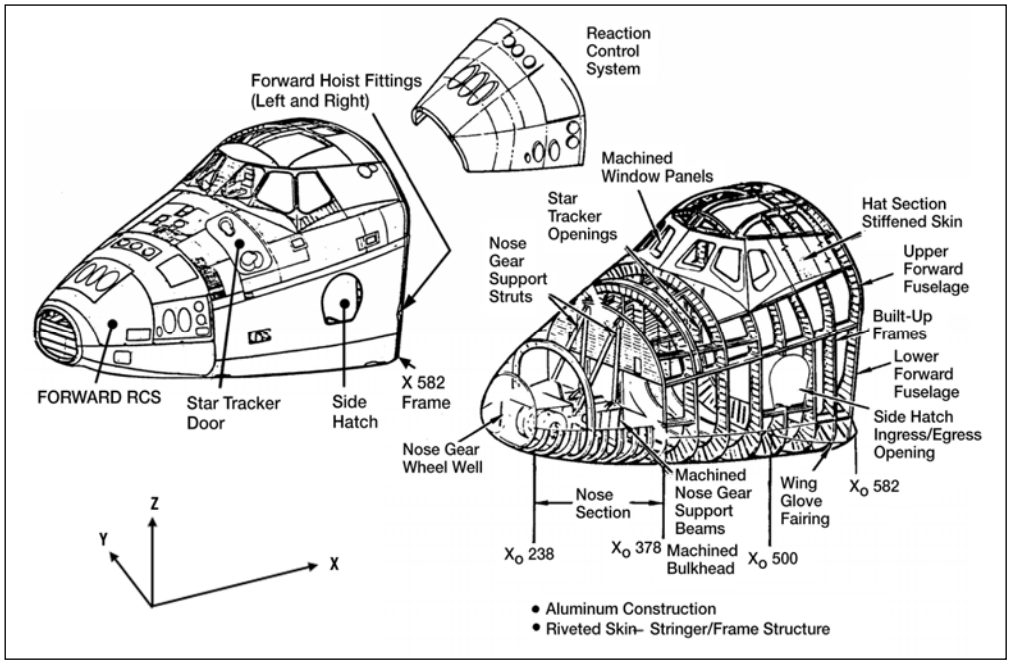


Figure 2.2-22. ***Forward fuselage structure*****.**

The CM is an airtight pressurized compartment that is constructed from welded aluminum panels (2219 aluminum alloys) with integral stringers, frames, and longerons. It is enclosed and protected by the FF skin panels and the TPS. It is supported and suspended inside the FF by a combination of fittings and linkages.

The CM is supported within the FF at four main attach points made of titanium (figures 2.2-23 and 2.2-24). Side links provide secondary stabilization but do not provide significant structural support. Two major attach points are at the aft end of the flight deck floor level. These links support loading in the X (longitudinal) direction. They also carry 52% of the Z (vertical) direction loading and 34% of the Y (lateral) direction loading. Because these two links carry loads in the X, Y, and Z directions, they are called xyz links, but are generally referred to simply as "x-links." The x-links bridge the boundary of the X_o 576 bulkhead (aft bulkhead of the CM) and the X_o 582 ring frame bulkhead (the forward bulkhead of the midbody (figure 2.2-25)). The x-links physically connect the CM, the FF, and the midbody structure.

The other links are the z-link and y-links. The z-link (vertical load reaction) is on the centerline of the CM forward bulkhead. The y-links (lateral load reaction) are on the lower portion of the X_o 576 bulkhead; they attach to the X_o 582 ring frame bulkhead. These attach fittings are designed to crash requirements to 9 G in –X (within a 30-deg cone), and ±3 G in the Y and Z axes.

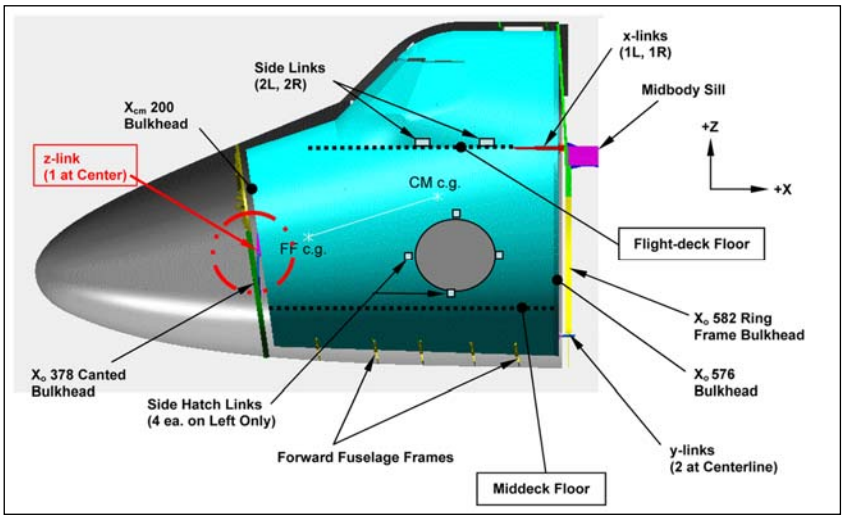


Figure 2.2-23. ***Thirteen links to the crew module inside the forward fuselage.***

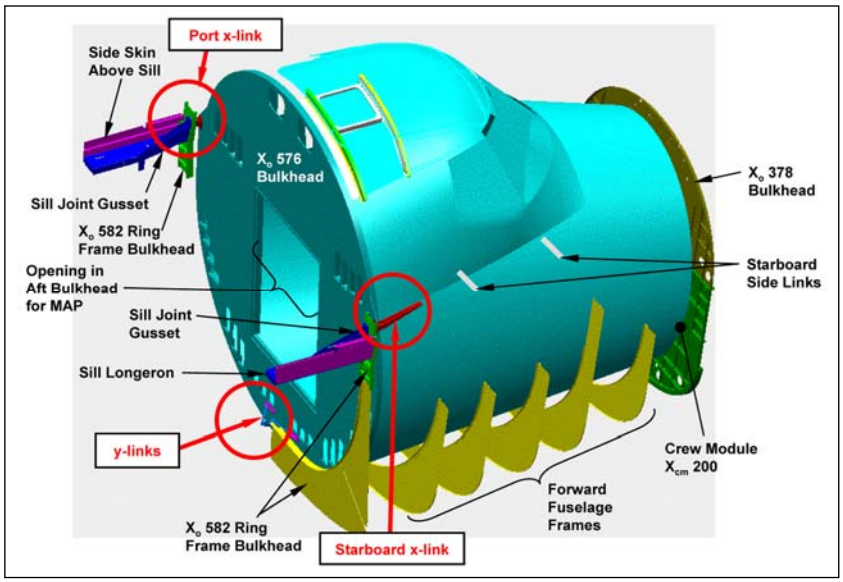


Figure 2.2-24. ***Crew module support links, starboard-side view.***

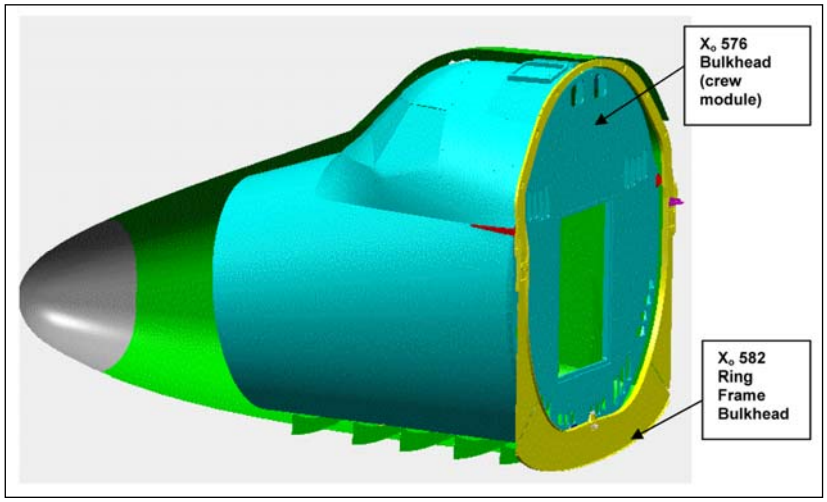


Figure 2.2-25. ***Picture of the X_o 576 bulkhead and the X_o 582 ring frame bulkhead relative to each other.***

2.2.4.2 *Forebody structure prior to the Catastrophic Event*

It is unlikely that any part of the FF experienced major structural failures under the estimated loading before the CE. A motion analysis was performed to compute loads at key locations after the LOC of the orbiter (see Section 2.1). Figure 2.2-26 shows that the estimated G loading at the CM increased to slightly more than 3 G just prior to the CE. Acceleration along the vehicle's Z axis (Gz) is the major component. This analysis assumed intact PLBDs. Loss of the doors may have increased loads in unknown ways such that the motion may have been affected. Assuming that the PLBDs were still intact, the results indicated that inertial loads were within the orbiter vehicle design limits.

Stress analysis also shows that it is unlikely that any of the major CM support links failed before the CE. Even if the temperature at the titanium links were elevated to about 600°F (316°C), the links still would have a high margin of safety before failure.

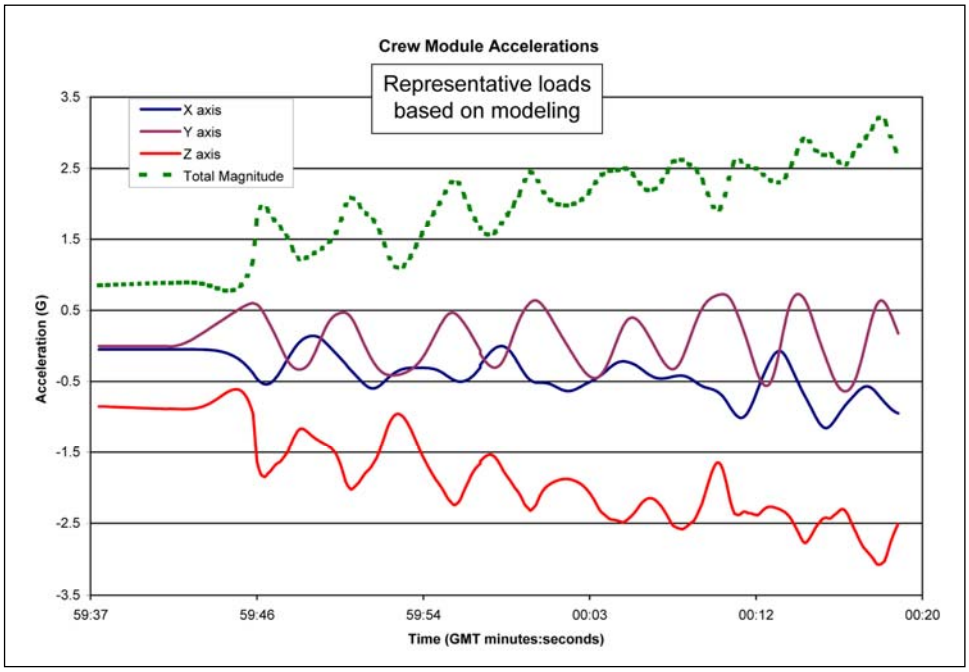


Figure 2.2-26. ***Resultant acceleration (G) vs. time prior to the Catastrophic Event in the*** **orbiter** ***coordinate frame.***

There was no evidence from the ballistic or debris cluster analysis that the forebody suffered any major breach before it separated from the midbody. Figure 2.2-21 (in the debris cluster assessment section) shows that forebody debris does not begin to appear in the debris field until other midbody debris is present in large quantities.

Most of the FF debris components that were recovered west of the main forebody debris field were TPS fragments and external items such as star tracker doors that came from the forward section of the FF, probably due to warping of the skin and aerodynamic loads as the forebody rotated (figure 2.2-27). Approximately 87% of the remaining FF structural debris has the same footprint as the CM debris footprint. This implies that the significant FF structural breakup occurred nearly coincidental with the CM shell breakup.

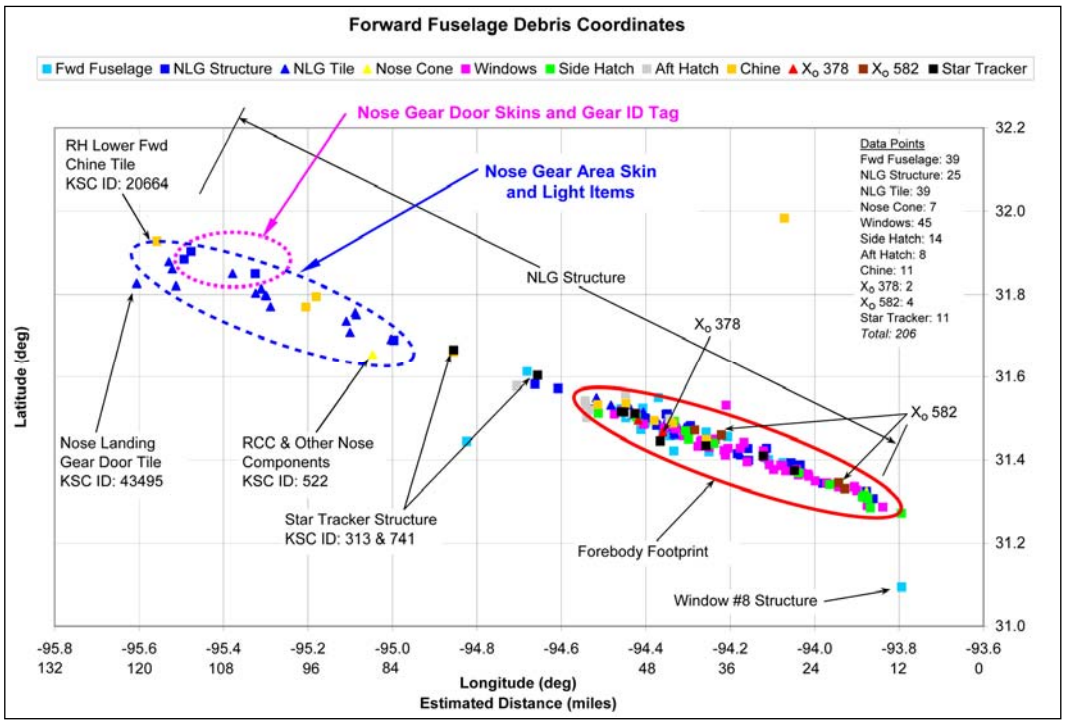


Figure 2.2-27. ***Forward fuselage debris field*.**

Two exceptions are the nose landing gear skin items and one landing gear nametag foil that was recovered farther west. These are two very thin pieces of nose landing gear skin and one thin foil piece with very low ballistic numbers, similar to that of the TPS debris.

2.2.4.3 *Payload bay doors*

During normal entry, the PLBDs and the sidewalls of the midbody are not exposed to high-temperature air flows. After the vehicle lost control, the PLBDs were exposed to abnormal midbody flexing. Also, as the orbiter changed attitude, parts of the vehicle that were not designed for high heating were periodically exposed to the velocity vector and the full force of entry heating. From a structural standpoint, the PLBDs are the most likely place for the first orbiter structural failure that is unassociated with the left wing degradation. RGPC data indicate that the Freon loops in the doors were intact at GMT 14:00:05. Since the Freon radiator loops are an integral part of the doors, this time bounds the earliest time at which the doors could have failed.

Figures 2.2-28 and 2.2-29 show the debris field coordinates for the recovered PLBD debris and a comparison of the PLBD vs. payload bay structures. Ballistics analysis showed very late release times (well past the CMCE)

that were considered suspect and assumed to contain errors due to lift generation and/or cascading events. Evaluation of the PLBD debris vs. payload bay structures shows that the debris field was extremely long, suggesting a very complex breakup and cascading event.

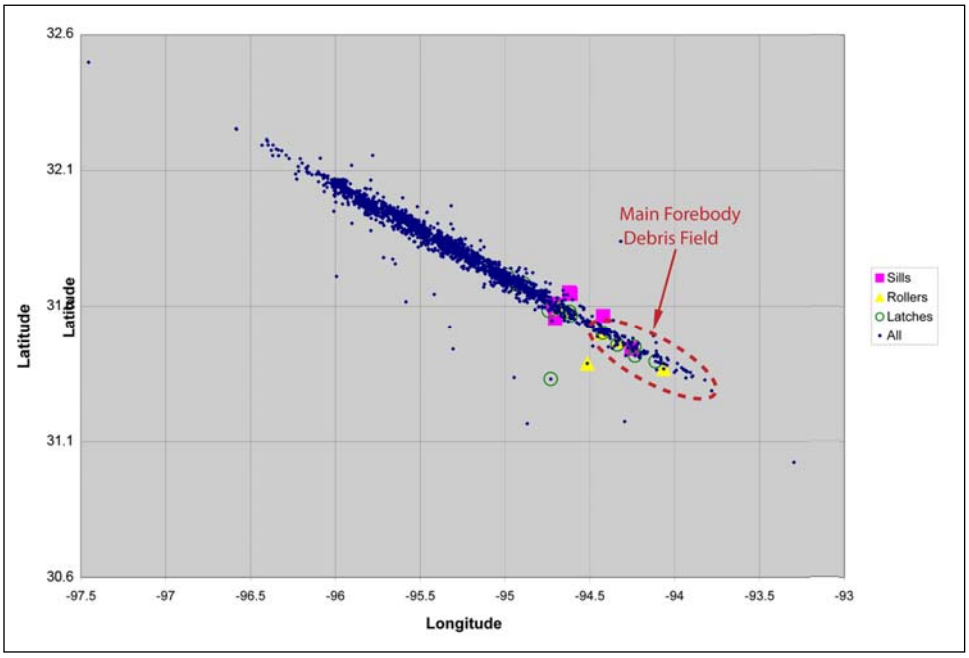


Figure 2.2-28. ***Payload bay door debris footprint relative to the main crew module debris field.***

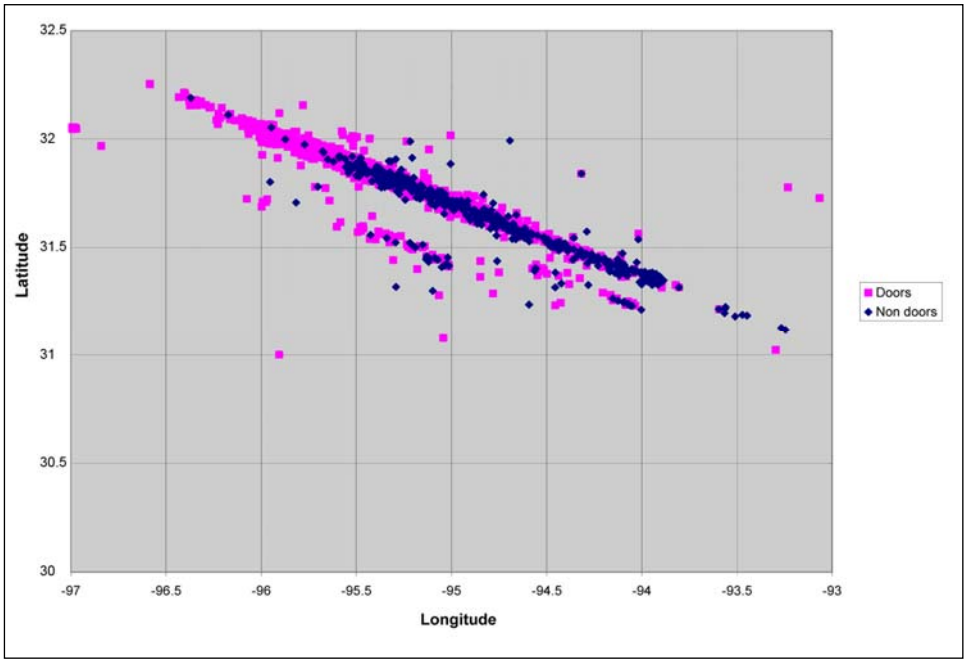


Figure 2.2-29. ***Payload bay door debris footprint relative to the payload bay structure.***

The PLBDs could have begun failing as early as immediately after the last available RGPC data, at GMT 14:00:06, or about 12 seconds before the CE. If this were the case, the debris cluster evidence could be interpreted to indicate that the sills and structural elements of the doors remained intact as a part of a larger section of structure after the departure of the doors.

2.2.4.4 *Forebody separation event*

Comparison to *Challenger*

Very little *Challenger* data regarding the CM structure were found. Most data were gathered by locating individuals who had performed some part of the assessments. In many cases, anecdotal reports were all that were available. It was anecdotally reported to the team that in the *Challenger* mishap, the CM separated from the orbiter between the X_o 576 and X_o 582 ring frame bulkheads. Figure 2.2-25 shows a picture of the two bulkheads relative to each other. This anecdotal information was subsequently found to be incorrect.

During the CAIB investigation, all *Columbia* debris were evaluated, but only items deemed significant to the cause of the accident were placed on the reconstruction grid and easily accessible. Significant efforts were carried out to identify CM structural debris, but not much was done on the midbody structure. As a result, the initial CSWG debris review resulted in the belief that no material had been recovered from the X_o 582 ring frame bulkhead, although significant elements had been found of the X_o 576 bulkhead. This belief was incorrect. X_o 582 ring frame bulkhead items were recovered; they were simply not readily available for inspection. This oversight led to the assumption that the failure mode was the same for *Columbia*, and this assumption was reported to the CAIB.[9]

However, the SCSIIT review of *Challenger* debris photographs clearly showed that part of the X_o 582 ring frame bulkhead was recovered with the *Challenger* X_o 576 bulkhead debris (figure 2.2-30).

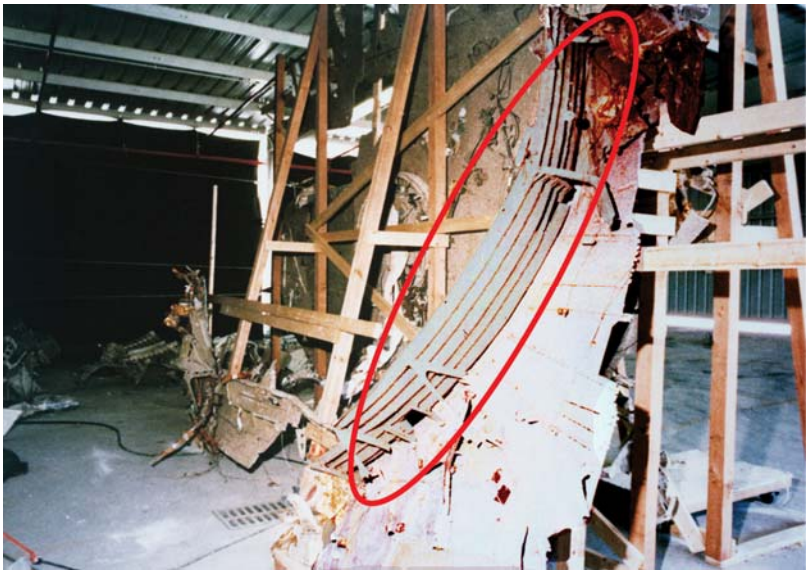


Figure 2.2-30. *Recovered X_o 576 bulkhead of the* Challenger *crew module showing a portion of the X_o 582 ring frame bulkhead* (circled in red).

This discovery led to a detailed search through *Columbia* debris for any recovered portions of the X_o 582 ring frame bulkhead. It should be noted that the X_o 582 ring frame bulkhead contains less mass than the X_o 576 bulkhead (a ring shape rather than a flat plate (figure 2.2-25)) and so significantly less debris would have been expected. As a result of this extensive search, conducted with the assistance of the *Columbia* Research and Preservation Team, four items of X_o 582 ring frame bulkhead debris were found. Plotting the ground coordinates on the debris field (figure 2.2-28) showed that these recovered elements of the X_o 582 ring frame bulkhead impacted the ground well within the recovered forebody debris cluster. This strongly

[9]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 77.

suggests that most of the X_o 582 ring frame bulkhead stayed with the forebody and only broke up when the forebody broke up. This is consistent with what appears in the *Challenger* photograph.

Load path assessment

With or without the PLBDs, stress concentrations and the highest structural loading would have occurred in the areas adjacent to the X_o 582 ring frame bulkhead (attachment of the forebody to the midbody) and X_o 1307 bulkhead (attachment of the aftbody to the midbody). At the X_o 1307 bulkhead, the wings and wing carry-through structures provide the area with some additional strength. Structural assessment shows that the weaker link is at the X_o 582 ring frame bulkhead area because this area must react to a more concentrated loading from the CM through the two x-links attaching to the midbody sill longerons.

By design, the forward splice of the X_o 582 ring frame bulkhead is much stronger than the aft splice because this bulkhead was built integrally with the FF shell and attached to the FF by multiple longitudinal frames or longerons, while the aft of the bulkhead is spliced to the midbody by only two sill longeron joints and the lower/side skin splice.

Without the PLBDs, the weak zone between the forebody and the midbody is the sill joint gusset on the aft side of the X_o 582 ring frame bulkhead (figure 2.2-31). This gusset is an offset structural connection that transfers the major X-axis loads from the CM to the midbody sill longeron via the x-links.

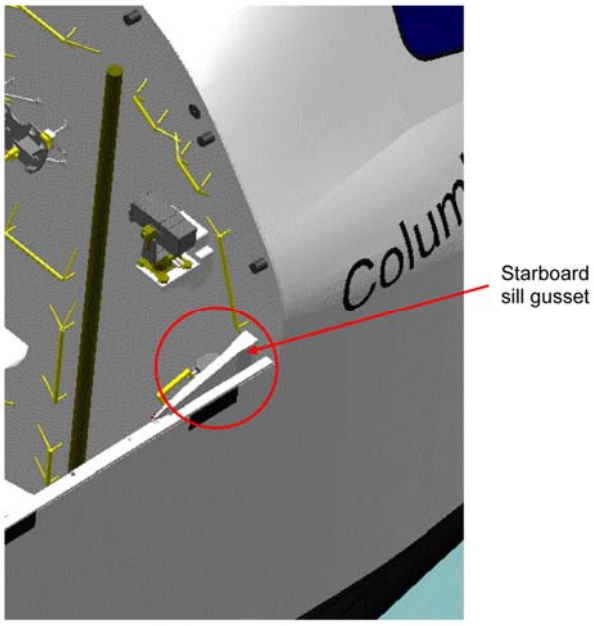


Figure 2.2-31. ***Close-up of the starboard sill gusset area.***

This assessment agrees with the debris field assessment finding that the X_o 582 ring frame bulkhead stayed with the forebody at the CE.

Midbody and X_o 582 ring frame bulkhead structure comparison

The midbody and forebody skin debris that originated from near the X_o 582 ring frame bulkhead was evaluated for confirming evidence. Specifically, the two forward hoist fittings that were used to lift the orbiter during ground processing and were aligned with the X_o 582 ring frame bulkhead, and the lower midbody skin from Bay 1, the forward-most section of the payload bay that is adjacent to the X_o 582 ring frame bulkhead, were evaluated.

The recovered starboard forward hoist fitting (figure 2.2-32) is still attached to FF skin. This is evidence that the midbody skin separated from the FF structure aft of the X_o 582 ring frame bulkhead. It was recovered

from the westernmost portion of the forebody debris field. The green-colored Koropon primer shown in the photograph is similar to the condition of the side skin a few feet above it (figure 2.2-33), which was recovered near the center of the forebody debris field. These elements most likely stayed with the forebody after the CE and only departed at the forebody breakup event. Temperatures above 400°F (204°C) degrade the primer's appearance, and high temperatures (>900°F) (486°C) will completely ablate it. The presence of Koropon indicates that the breakup of these elements was caused by mechanical overload rather than thermal effects. The presence of intact primer indicates that the FF TPS protected this area from entry heating until the forebody breakup at the CMCE.

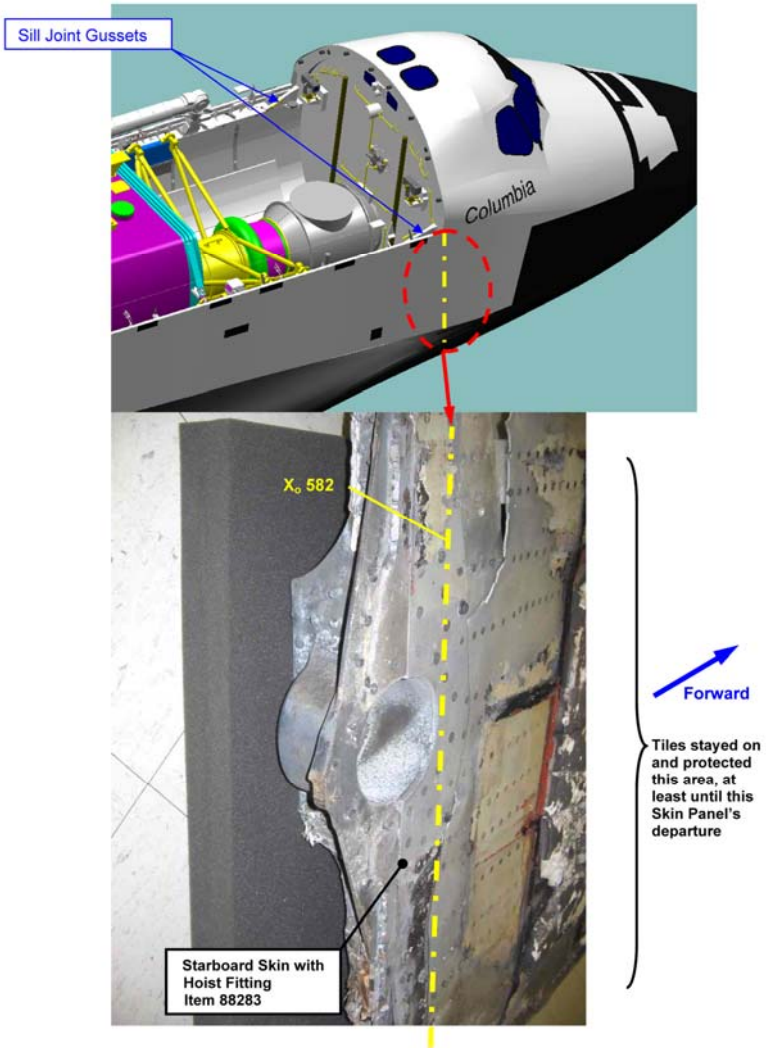


Figure 2.2-32. ***Forward fuselage skin still attached to the starboard hoist fitting.***

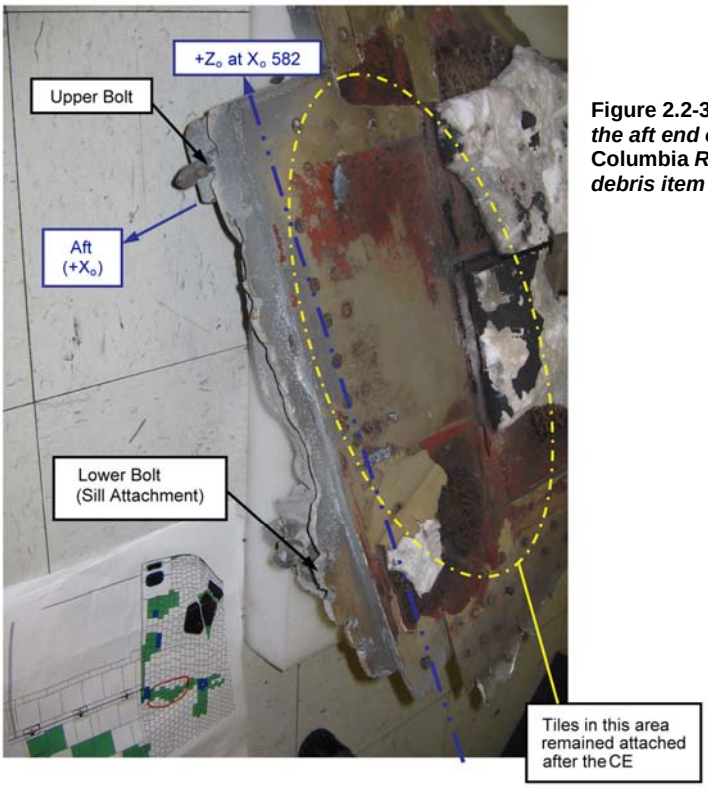


Figure 2.2-33. ***Thermal erosion on the aft end of the starboard-side skin,*** **Columbia** ***Reconstruction Database debris item no. 2436*****.**

The condition of the recovered forward portside hoist fitting (figure 2.2-34) indicates that the Xo 582 ring frame bulkhead likely broke away from the midbody skin via mechanical failure, prior to any thermal effects. The Bay 1 side skin pulled away because of mechanical failure. The midbody skin splice severed the fasteners from the titanium hoist fitting by mechanical force, some Hilok bolts were cut off, and some flush bolt heads pulled through the skin splice. This damage, which may have occurred at the CE, suggests that the forebody yawed left and pitched down relative to the midbody. This motion may also have broken the port x-link lug (discussed in the next section). This kind of mechanical failure could only happen when the hoist fitting was still attached to the FF, when sufficient mass existed to exert this level of force.

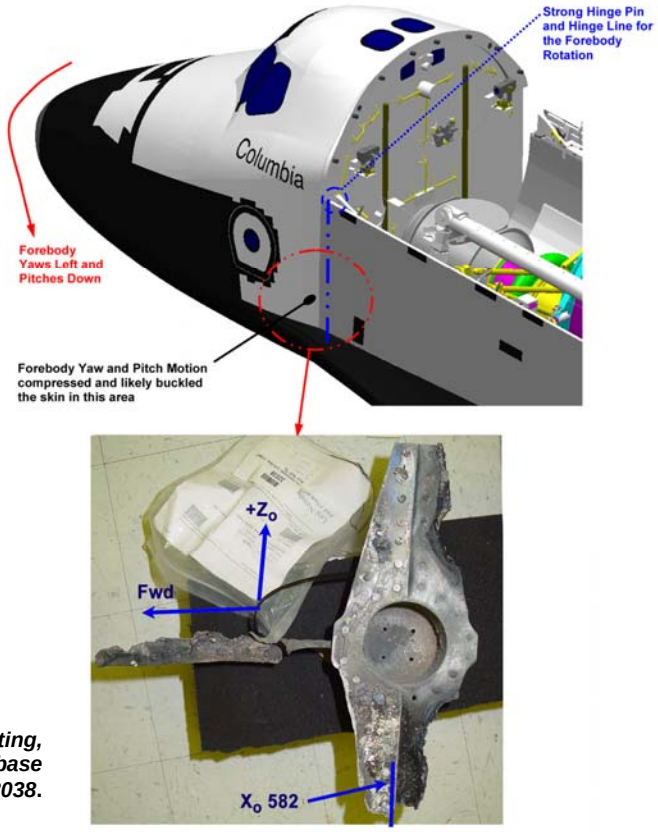


Figure 2.2-34. ***Forward port hoist fitting,*** **Columbia** ***Reconstruction Database debris item no. 32038*****.**

Unlike the starboard hoist fitting, the skin panel forward of the port hoist fitting was not recovered. One possible explanation is that the FF skin on the forward side of this item was deformed or wrinkled by compression loading at the CE when the forebody yawed left and pitched down. This may be supported by the fact that the port forward hoist fitting was recovered in the middle of the forebody debris field while the starboard hoist fitting was found in the western end of the debris field. Wrinkled skin most likely debonded TPS tiles, so this area would be subjected to thermal flow damage on at least the skin-side surface starting at the CE, then on both sides after it departed from the forebody after the CMCE. Alternatively, the port area may have simply stayed intact longer and received more entry heating following the CMCE.

The recovered midbody bottom skin pieces from Bay 1 (the most forward bay in the midbody, immediately aft of the X_o 582 ring frame bulkhead (figures 2.2-35, 2.2-36, and 2.2-37)) show little or no heat damage to the skin at the splice aft of the X_o 582 ring frame bulkhead. The damage is mainly mechanical breakup of the frame to skin splice.

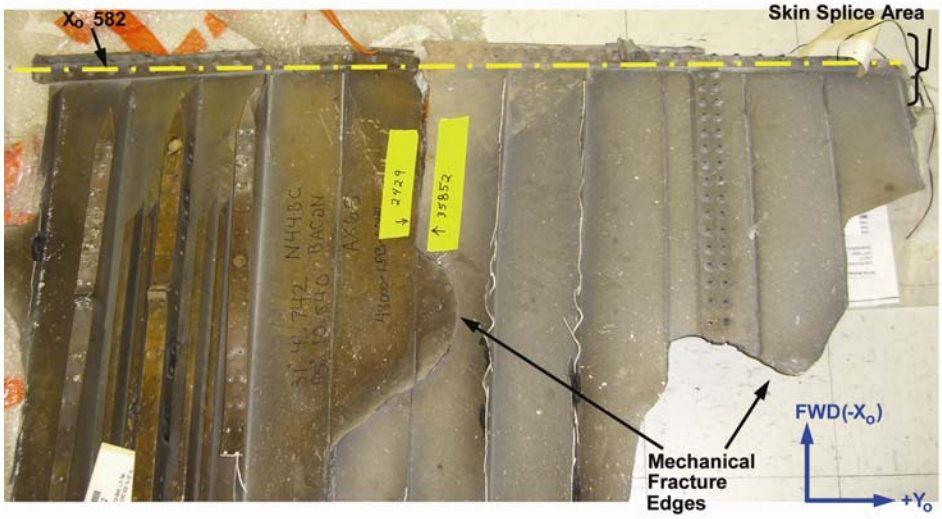


Figure 2.2-35. ***Bay 1 midbody bottom skin,*** **Columbia** ***Reconstruction Database debris item nos. 2429 and 33852*****.**

Figure 2.2-36. ***Bay 1 midbody skin,*** **Columbia** ***Reconstruction Database debris item no. 88290*****.**

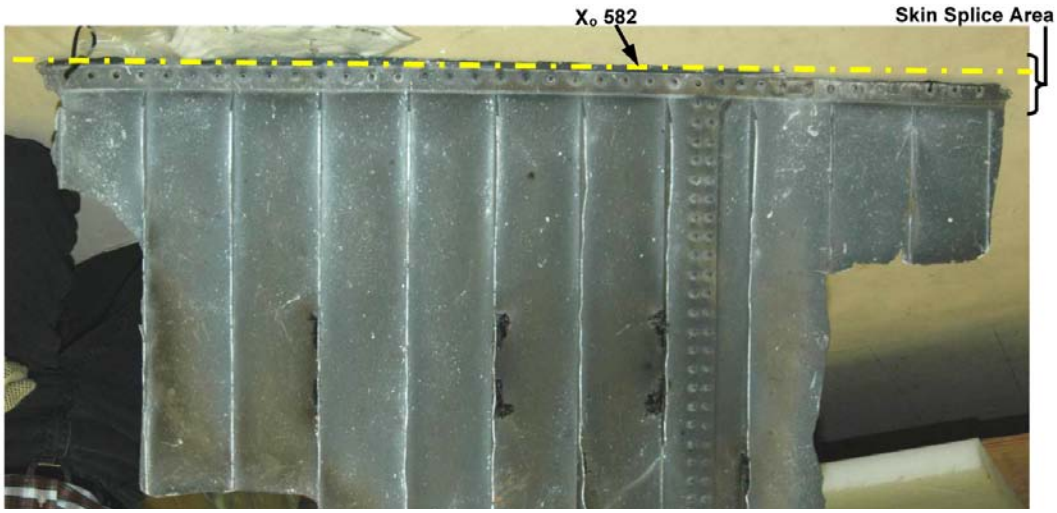


Figure 2.2-37. ***Bay 1 midbody skin,*** **Columbia** ***Reconstruction Database debris item no. 14908.***

The *Columbia* Reconstruction Team noted that thermal damage to the midbody was not generally severe.[10] The thin skin and stringers of Bay 1 experienced some heating but not sufficient to melt the material. One scenario that may explain this is that this bay corresponded to the X location of the structural failure and disintegrated quickly during or immediately after the CE. Even if the bay skin remained with the midbody at the CE, without the X_o 582 ring frame bulkhead frame aerodynamic loads would peel off the Bay 1 skin quickly. This is consistent with the interpretation that the forebody and midbody separated in this area.

Crew module attach fittings

The primary attach fittings of the CM to the forebody are the x-links, the y-links, and the z-link (figures 2.2-23 and 2.2-24). All are made of titanium. The x-links also connect the FF and CM to the midbody and will be addressed here. The y-links and z-link will be discussed in Section 2.4.

The x-links are normally protected by the PLBDs, which also provide stiffness for the orbiter. By design, the x-link attachment to the CM is stronger than its attachment to the FF and midbody because it is attached to the extension webs of the CM flight deck floor and X_o 576 bulkhead.

Comparison of the recovered port and starboard x-links (figure 2.2-38) shows two important differences. First, the starboard x-link was recovered with a portion of the X_o 582 ring frame bulkhead and starboard sill attached, while the port x-link failed forward of the X_o 582 ring frame bulkhead. Second, the starboard x-link experienced much more thermal erosion than the port x-link. It is clear that the directionality of the thermal damage (aft to forward flows above the x-links, and forward to aft flows along the bolt heads) is the same on both links. This orientation was shown to be unlikely due to a common trim attitude when free-flying (see Section 2.1). This strongly suggests that they were in the same relative orientation (still attached to forebody structure) when exposed to heating. The common thermal pattern on the x-link body is discussed in Section 2.4.

[10]STS-107 *Columbia* Reconstruction Report, NSTS-60501, June 2003, p. 55.

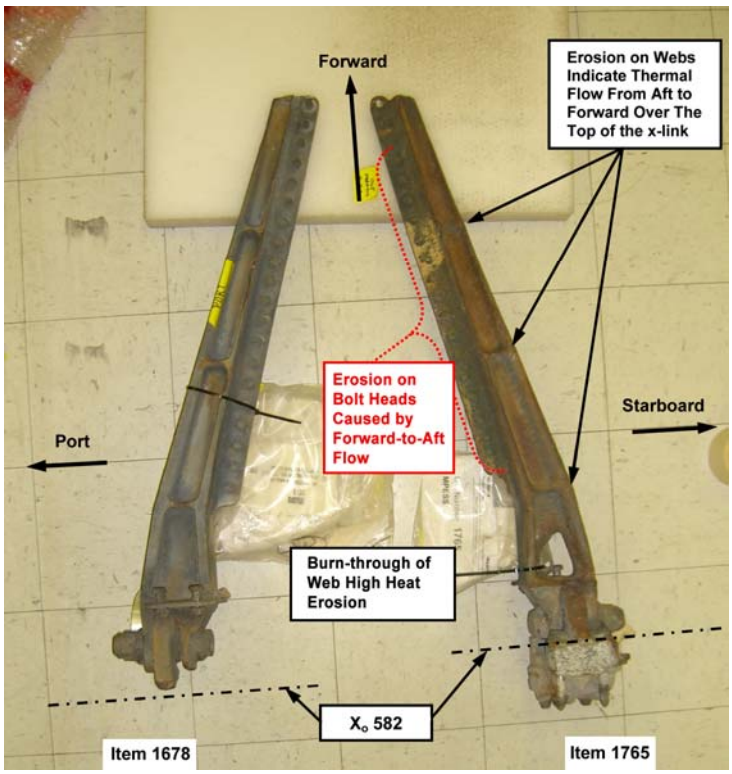


Figure 2.2-38. ***Port and starboard x-links*.**

Starboard x-link assessment

Figures 2.2-39 and 2.2-40 show the heavy thermal erosion on the sill joint gusset that is attached to the starboard x-link. The portion of side skin splice from just outboard of the starboard x-link was also recovered. Heavy heat erosion was noted on the midbody side (aft side of X_o 582 ring frame bulkhead), but was not significant elsewhere on the debris. This skin panel was recovered near the center of the forebody debris field. Together, this confirms that the CM stayed inside the FF shell until the forebody breakup, and suggests that the trailing edge (midbody side) was exposed to heating.

The structure aft of the X_o 582 ring frame bulkhead on both the starboard covering side skin and the starboard x-link was heavily damaged by heat. The heat erosion pattern on bolt heads on the aft side shows that the higher (+Z) and outboard (+Y) bolts experienced more heat erosion than the lower/inboard bolts. The resulting shape of eroded bolt heads indicates

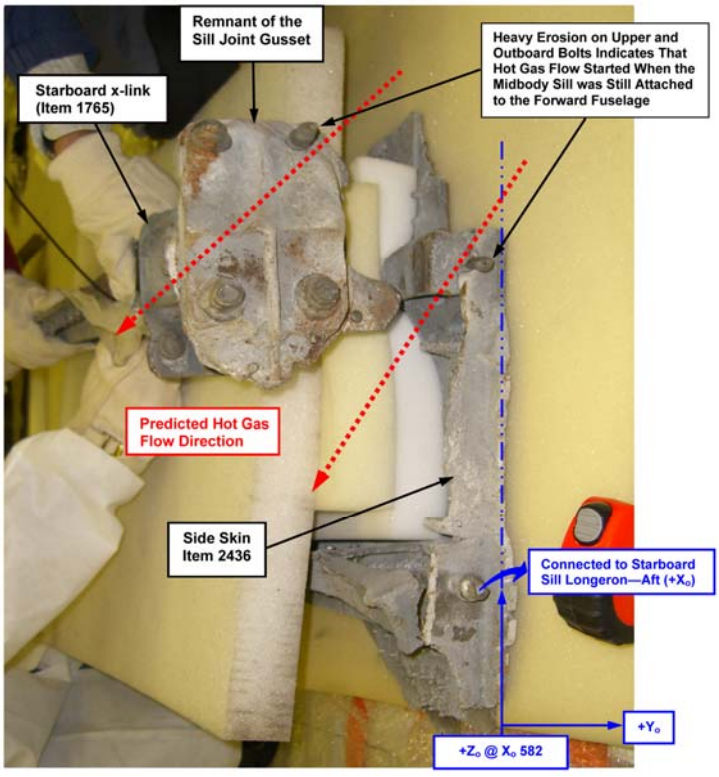


Figure 2.2-39. ***View looking forward at the starboard x-link and side skin*.**

that hot gas flowed down and inboard (figures 2.2-39 and 2.2-40). This suggests that the aft side of the starboard x-link area was exposed to hot gas while the starboard midbody sidewall/sill was still attached to the X_o 582 ring frame bulkhead and FF.

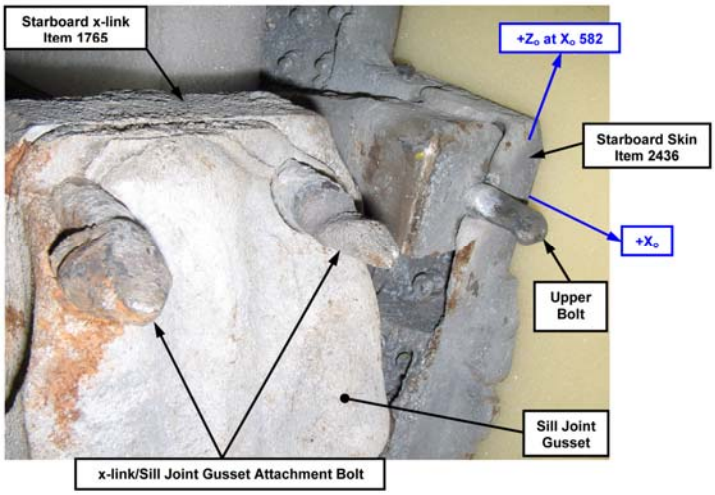


Figure 2.2-40. ***Detail of thermal erosion on the upper bolts at the starboard x-link and skin.***

Hot gases flowing over the starboard sill and X_o 582 ring frame bulkhead could elevate the temperature of the starboard structural joint between the sill and the FF behind the starboard x-link while the sidewall shielded the lower portions of these structures. It appears that the sill and the sidewall were still attached to the FF when this thermal erosion took place. The condition of the remaining bolt heads indicates that these bolt heads and shanks did not fail by mechanical loading.

To summarize the assessment, the point of the orbiter that was weakest under dynamic rotational loading was likely to be the sill gusset immediately aft of the x-link and the X_o 582 ring frame bulkhead. Debris shows that for the starboard sill, the failure of the midbody sill joint occurred at a location aft of the X_o 582 ring frame bulkhead, perhaps near the middle of the gusset (figures 2.2-41 and 2.2-42). Since orbiter load limits were apparently not exceeded, this implies that thermal degradation was required.

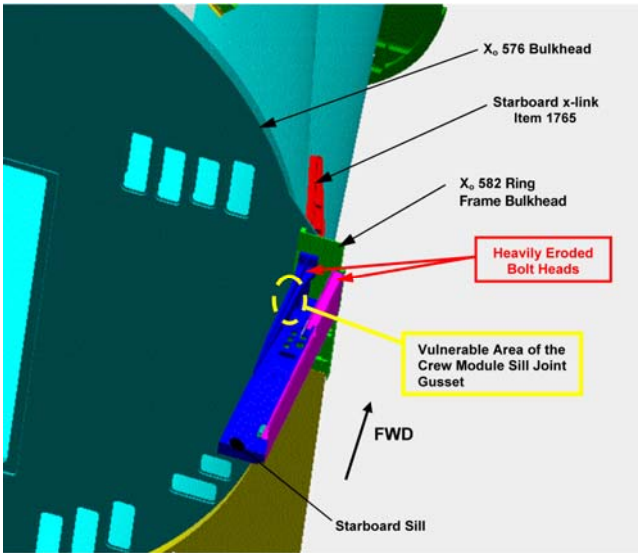


Figure 2.2-41. ***Weak links – sill joint gusset, starboard side.***

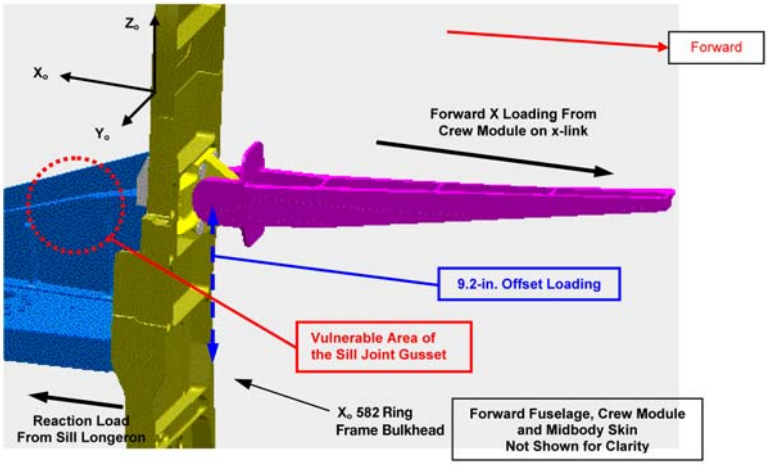
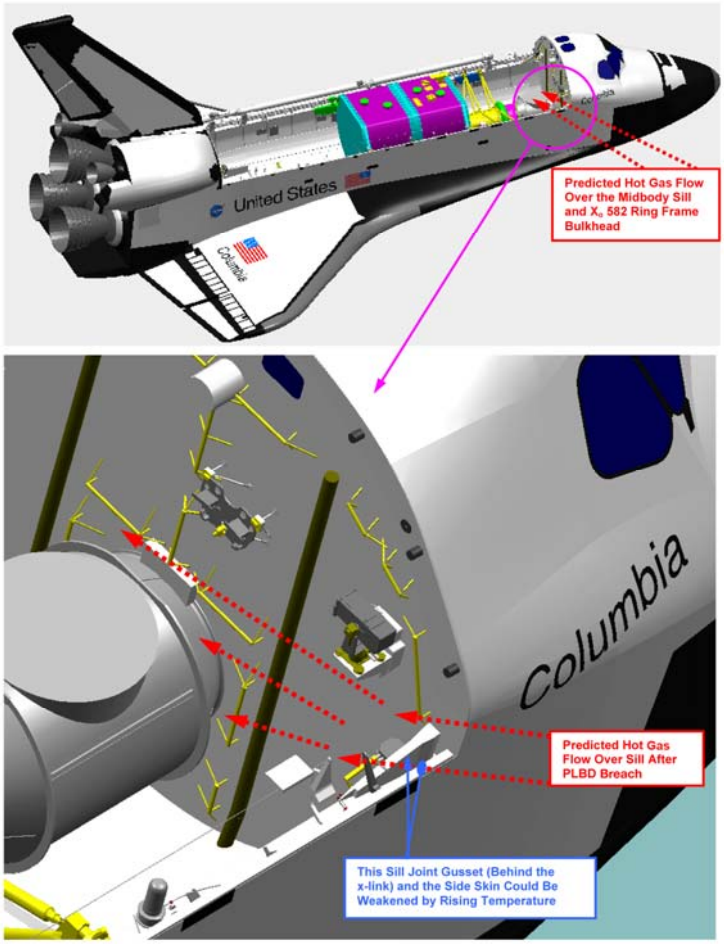


Figure 2.2-42. ***Structural loading from starboard x-link to sill longeron.***



This thermal degradation could have occurred prior to the CE if the PLBDs departed or were compromised in that area (figure 2.2-43). Debris evidence shows that the starboard sill joint experienced more thermal damage than the port sill joint. This suggests the possibility that rising temperature on the starboard sill area weakened the sill joint until it failed mechanically. Although orbiter motion generally resulted in "belly-into-the-wind" orientation (see Section 2.1), with the large pitch and roll oscillations hot gas could flow beyond the belly of the orbiter, up and over the sill. However, hot gas exposure to the sill area could only be short and intermittent (<3 seconds at a time). Debris field and ballistic assessment of PLBD debris is inconclusive. However, a local failure or compromise on the starboard PLBD near the sill could have resulted in this scenario without evidence being apparent in these assessments.

Figure 2.2-43. ***Hot gas flow heats up starboard sill and the adjacent x-link.***

It is unlikely that the temperature increase on the starboard-side sill joint was the primary cause in the orbiter breakup. High heating most likely played a contributing role for the starboard sill joint failure. The main cause for the sill joint failure was probably mechanical loading exceeding the capability of the thermally weakened structure. A failure in this location would likely cascade structural failures along load paths throughout the vehicle.

As illustrated in figure 2.2-44, failure of the starboard sill joint area would trigger a separation of the forebody away from the midbody. Starting from the starboard side, the midbody skin splice would fail progressively, opening from starboard to port at the forebody aft bulkheads.

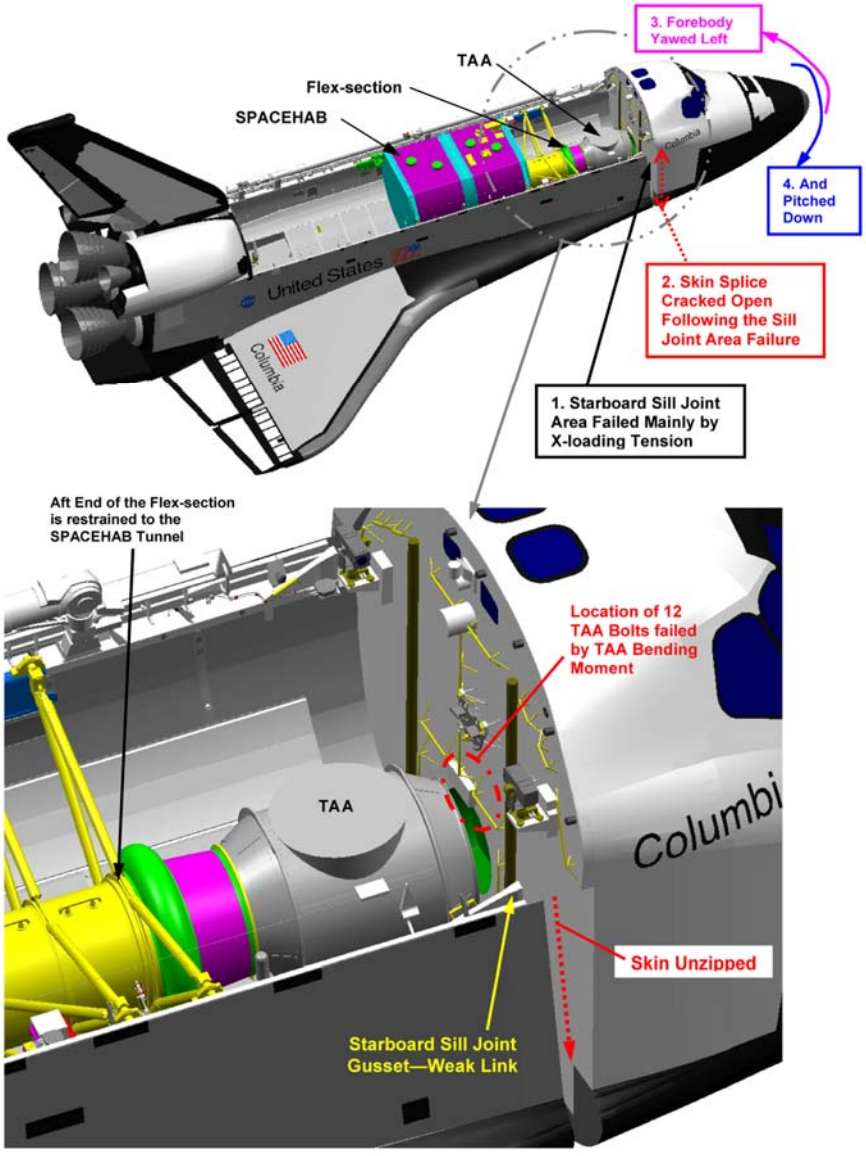


Figure 2.2-44. ***Forebody departed the midbody at the Catastrophic Event.***

The other possibility is that a portion of the starboard midbody remained with the forebody after the CE and failed during subsequent motion of the forebody. However, it is unlikely that the forebody separated the midbody farther aft of the starboard sill joint area unless the farther aft area had been damaged earlier because of other thermal or mechanical causes, for which there is no clear evidence.

Port x-link assessment

A portion of the X_o 582 ring frame bulkhead and portside sill fitting, which has a lug that supports the port x-link, was recovered and inspected. The lug broke near its base mainly because of a bending load caused by the x-link and the forebody yawing to the left (figure 2.2-45). This yawing motion was identified by the yielding on the outboard flange of the clevis on the aft end of the port x-link (figure 2.2-46).

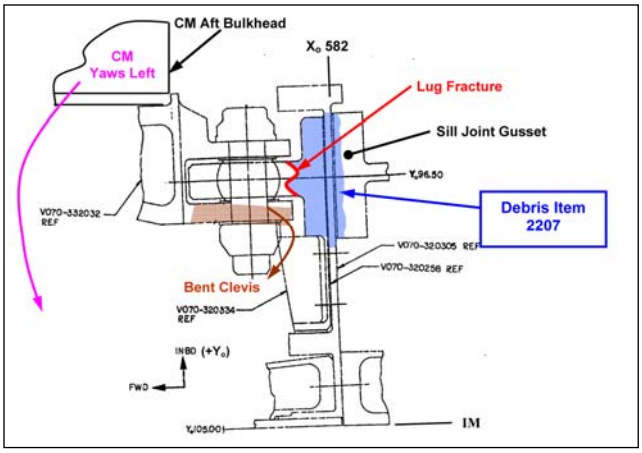


Figure 2.2-45. ***How the port x-link failed, cross-section view looking down.***

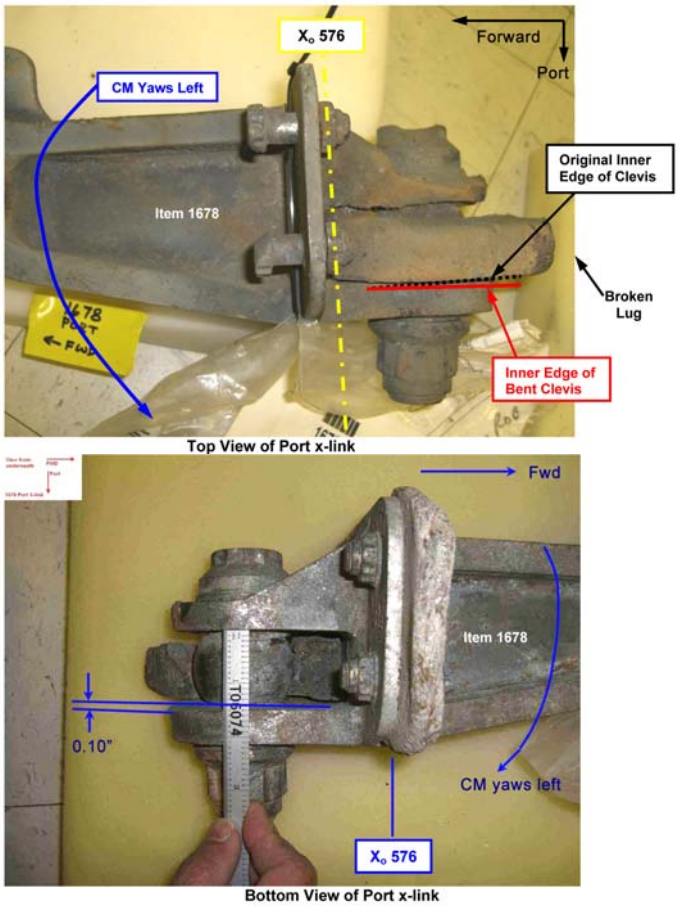


Figure 2.2-46. ***Views of the port x-link with about a 0.10-inch yield on the outboard side of the clevis.***

As shown in figure 2.2-47, the outboard upper bolt head was gouged and deformed by an impact from the broken lug, which remained in the port x-link clevis. This indicates that the forebody not only yawed to the left but also shifted to the left when it moved away from the midbody.

Figure 2.2-47. Columbia *Reconstruction Database debris item no. 2207, which includes the remainder of the lug fitting that supports the aft end of the port x-link*.

This implies that when this lug failed, the port x-link was still attached to the CM and the lug still had the structural support of the midbody sill or a large portion of the portside midbody/sill attached. On the portside, the FF and x-link separated from the midbody sill on the *forward* side of the X_o 582 ring frame bulkhead. The remaining part of the portside X_o 582 ring frame bulkhead and the sill joint gusset experienced little heat erosion (figures 2.2-48 and 2.2-49). These two items were found at longitude 94.0W and 94.3W, which are in the central portion of the midbody debris field (figure 2.2-16). It is possible that they stayed with the portside midbody sill, which is one of the heaviest parts of the midbody shell. No ballistic assessment was performed to confirm or deny this theory.

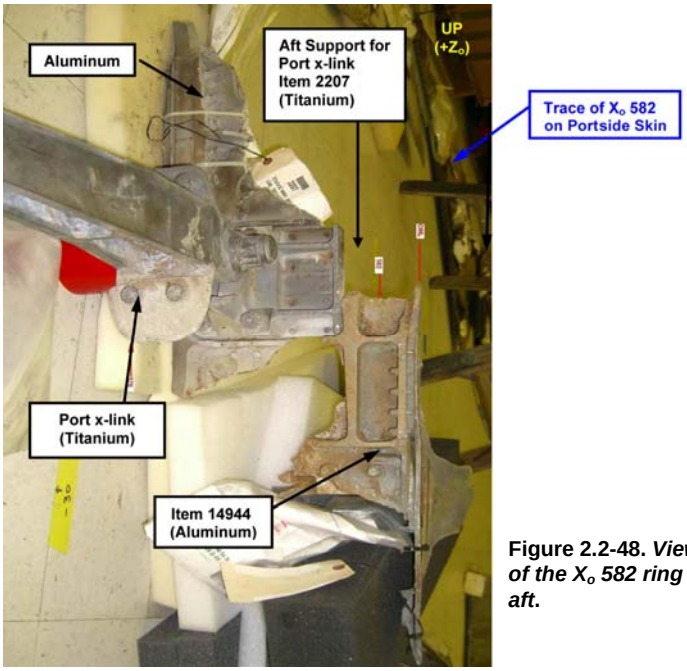


Figure 2.2-48. ***View of the port x-link and a portion of the X_o 582 ring frame bulkhead debris items, looking aft.***

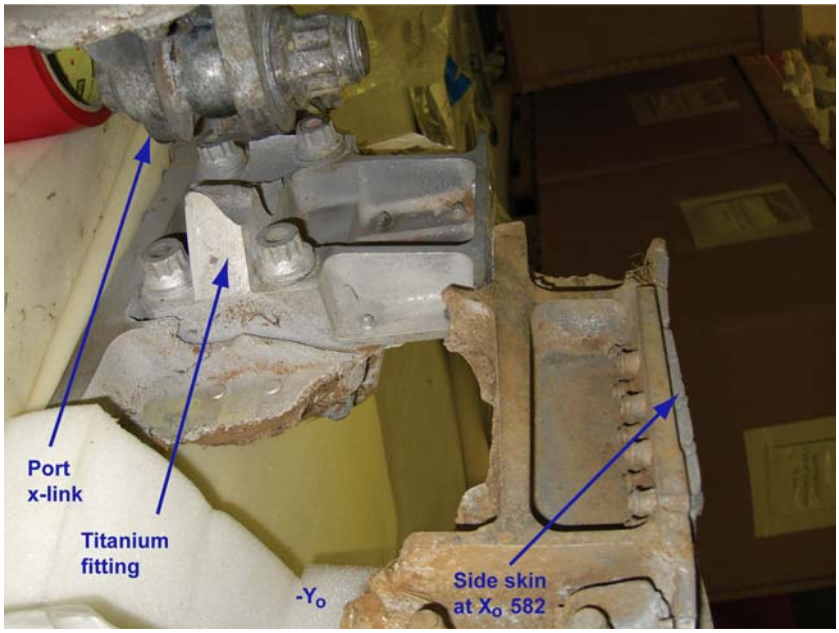


Figure 2.2-49. ***View looking up and aft at the port x-link.***

These pieces provided a more subtle understanding of the separation event between the forebody and the midbody.

The conclusion was that the forebody separation started *aft* of the X_o 582 ring frame bulkhead on the starboard-side sill area, and then spread across the fuselage along the aft portion of the X_o 582 lower ring frame bulkhead. The breakup progressed toward the portside sill while the forebody yawed to the left and pitched down, causing the portside lug fitting to fail by bending to the portside. This resulted in failure *at* the X_o 582 ring frame bulkhead on the portside of the bulkhead, leaving a portion of the bulkhead with the midbody.

Immediately after the CE, the starboard CM x-link pulled forward through the X_o 582 ring frame bulkhead as soon as the CM began to move forward inside the FF shell. The CM port x-link was unsupported by midbody structure after the CE and was only attached to the CM. Interestingly, the starboard side of the *Challenger* X_o 582 ring frame bulkhead also remained with the X_o 576 bulkhead. However, because the breakup events had different causes, this is assumed to be a coincidence.

Conclusion L3-2. The breakup of both *Challenger* and *Columbia* resulted in most of the X_o 582 ring frame bulkhead remaining with the crew module or forebody.

Tunnel adapter

The tunnel adaptor assembly (TAA) provided the internal pressurized path for the crew to move between the middeck to the SPACEHAB in the payload bay (figure 2.2-50). The tunnel adaptor attached to the middeck access panel (MAP), which is a large removable panel in the X_o 576 bulkhead.

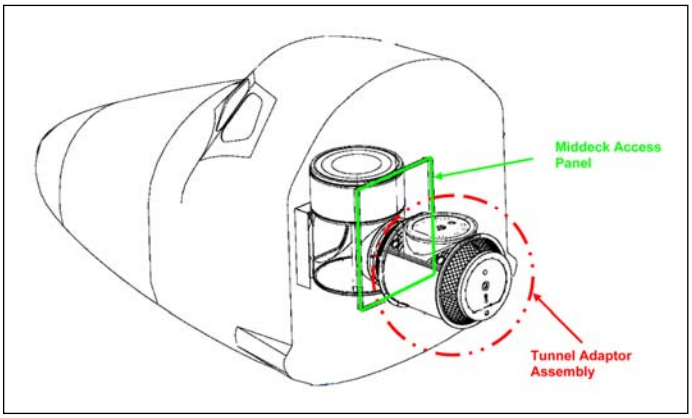


Figure 2.2-50. ***Tunnel adaptor assembly and middeck access panel.***

One note of importance is that a breach in the TAA would cause the immediate rapid depressurization of the internal airlock, since there was no hatch between it and the TAA volume. The airlock inner hatch to the middeck was closed, so this would not result in the depressurization of the main CM, only the airlock.

Some items that were stowed in the airlock were recovered in the western portion of the debris field, prior to the main body of forebody debris. These items were either soft goods that were stowed in bags taped or strapped to the wall or floor, or bonded items (no airlock structure), which indicates that although the airlock rapidly depressurized, the structure remained intact. A headrest pad (used for ascent only) was stored in the internal airlock for entry according to the crew's entry stowage plan. Ballistic analysis was performed on this one item because of its easily described shape. This analysis determined that this item separated from the airlock/TAA compartment at GMT 14:00:36. Ballistic analysis was also performed on two small pieces of TAA structure. The release time for both pieces was computed to be GMT 14:00:43.

The debris field analysis shows that most of the TAA was recovered west of the main forebody debris field, suggesting that the TAA did not remain with the forebody but departed before the CMCE.

As shown in figures 2.2-51 through 2.2-54, the 12 high-strength bolts (180,000 pounds per square inch (psi)) that attached the forward ring of the TAA to the MAP failed with shanks still remaining in the inserts. This suggests that this area was subjected to high tension and bending while the remaining bolts were not subjected to the same loading. It implies that this TAA joint reacted to a bending moment (M_{TAA}) plus aft tension loading (figure 2.2-54). When the aft end of the TAA was restrained by the SPACEHAB tunnel to the midbody, the forebody yawing left and pitching down relative to the midbody could create the same bending effects. This suggests that the TAA broke away from the MAP when the forebody separated from the midbody at the CE or shortly afterwards.

Figure 2.2-51. ***Tunnel adapter assembly and the manufacturing access panel debris, view looking forward at "B" hatch opening.***

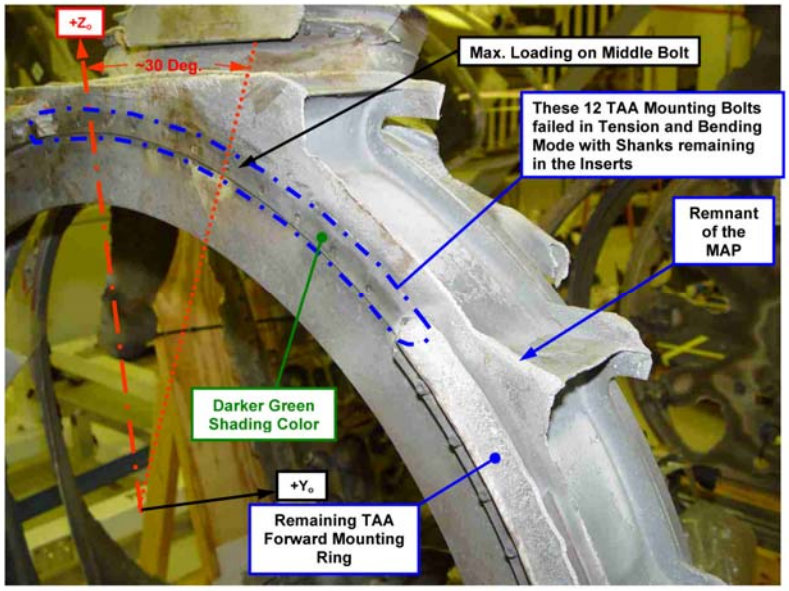


Figure 2.2-52. ***Tunnel adapter assembly debris showing that the bolt shanks failed mainly by tension and bending.***

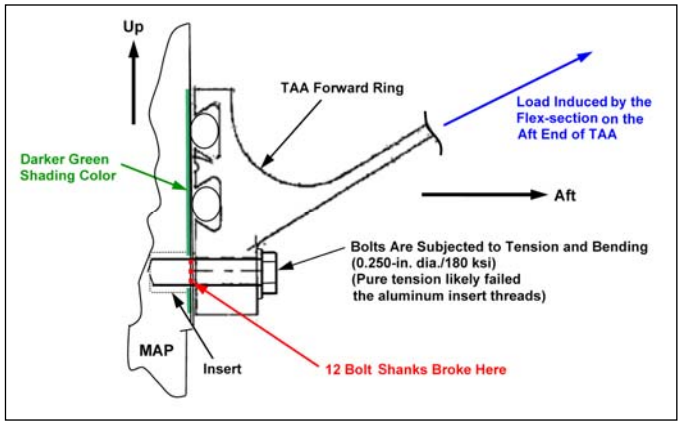


Figure 2.2-53. ***Cross section through tunnel adapter assembly mounting bolt.***

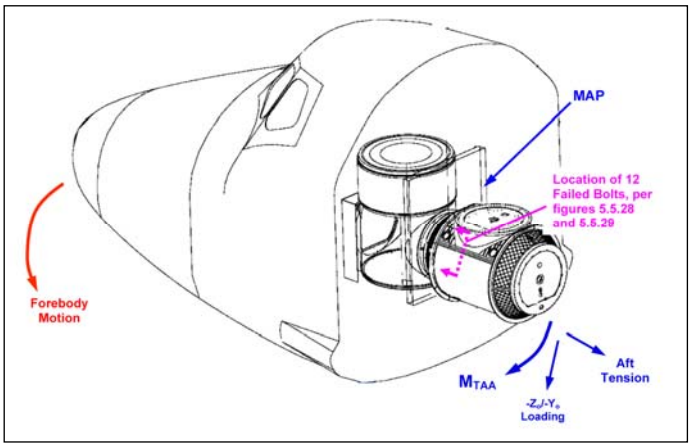


Figure 2.2-54. ***Predicted motion based on debris evidence.***

Darker shading color on the surface of the MAP, as was noted on figures 2.2-51 and 2.2-52, indicates that this area was shielded by the missing portion of the TAA ring and suggests that the MAP surface outside of the TAA ring was subjected to thermal flow before the TAA departure. If the TAA departed at the CE, the MAP or the aft bulkhead of the CM was exposed to the thermal flow for some period of time prior to the CE (figure 2.2-44). This would agree with the theory that the PLBDs were compromised prior to the CE.

The failure of these TAA mounting bolts also indicates that it is unlikely that the X_o 576 CM aft bulkhead suffered any major structural failure at the CE, although it was subjected to CM internal pressure and apparently some thermal exposure. By design, the X_o 576 bulkhead can handle 24 psi of internal pressure. The TAA aft loading required to fail the TAA mounting bolts would impose less load on the X_o 576 bulkhead than the maximum cabin internal pressure load case. The large mass of the bulkhead would also effectively absorb and diffuse heat.

As shown in figure 2.2-50, the MAP is a significant element of the X_o 576 bulkhead. The bulkhead is linked to the flight deck floor, the middeck floor, and the two partitions on both sides of the airlock. It is not likely that the MAP would fail without causing a massive failure of the whole aft section including the X_o 576 bulkhead and the airlock, affecting the floors and partitions (figure 2.2-55). This supports the conclusion that no significant damage occurred to the bulkhead at the CE.

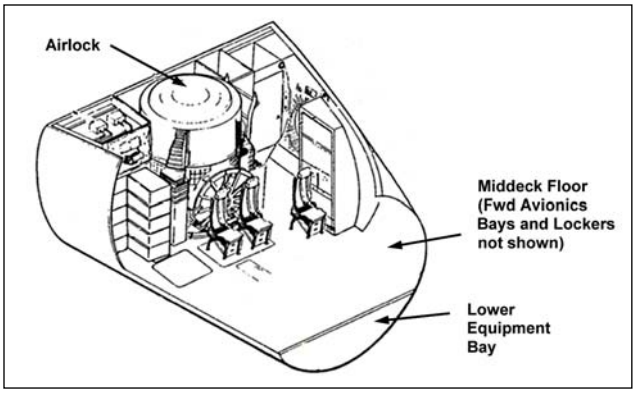


Figure 2.2-55. ***Airlock in the middeck*.**

Extreme thermal erosion at the lower edge opening on the MAP where the forward end of the TAA is attached indicates that after the TAA broke away from the MAP, hot gas flowed in over the lower edge of the "B" opening in the MAP. This thermal flow also caused heat erosion to the two aft corners of the stowage platform, which was mounted to the airlock floor (figure 2.2-56). This is consistent with the rotational motion of the forebody post-CE, when such an exposure could have occurred.

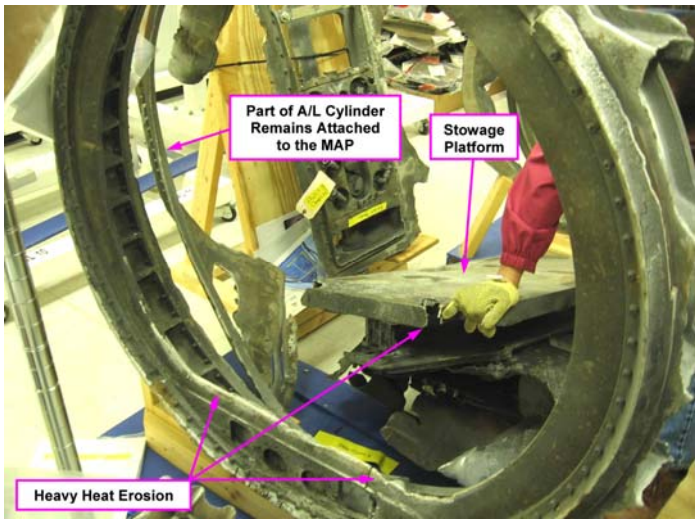


Figure 2.2-56. ***View looking forward at the "B" hatch opening on the X_o 576 bulkhead*.**

2.2.5 Synopsis of orbiter breakup sequence

As reported by the CAIB, the left wing was gradually failing until orbiter breakup. At GMT 13:59:37, the orbiter lost control. Around GMT 13:59:49, the left OMS pod probably departed. It is possible that the PLBDs departed next or at least degraded enough to allow hot gas to flow over the starboard sill as the orbiter rotated. The orbiter then broke up at GMT 14:00:18 (CE) into aftbody, forebody, and midbody/right wing components.

It is likely that the initial failure was at the weakest area of the overall structure, just aft of the X_o 582 ring frame bulkhead. This failure progressed from starboard to port by unzipping the skin splice at the X_o 582 ring frame bulkhead. The forebody yawed left and pitched down, hinging at the portside x-link. This motion broke the lug fitting that supports the aft end of the port x-link, leaving the base of the lug and the local X_o 582 ring frame bulkhead portside with the midbody. The aft and right wing attachments successively failed as abnormal load paths were propagated through the orbiter from the forebody separation. The next failure was the TAA, depressurizing the airlock. The forebody remained intact at orbiter breakup (see Section 2.4).

> ***Recommendation L3-1.*** Future vehicles should incorporate a design analysis for breakup to help guide design toward the most graceful degradation of the integrated vehicle systems and structure to maximize crew survival.

2.3 Crew Cabin Pressure Environment Analysis

An analysis of the crew cabin pressure environment in *Columbia* was critical to formulating an understanding of what happened to the crew. This analysis was particularly important to acquiring insight into how the crew cabin environment affected the crew's ability to make decisions, at what point during the depressurization the crew's ability would have been permanently compromised, and when the crew members would have lost consciousness. To formulate this understanding, various aspects of the accident were analyzed. These included telemetry, ground-based videos, the debris, medical evidence, and structural analysis. This section draws on information described in detail in other areas of this report (structural analysis, medical findings, video analysis, etc.) to determine the timeline of the cabin depressurization. The timeline includes times for when the cabin depressurization began, when the depressurization was complete, and the rate of depressurization. Related to this analysis is determining the location(s) of the CM breach(es).

The conclusions relative to the cabin depressurization timeline are provided below:

Conclusion L1-1. After loss of control at GMT 13:59:37 and prior to orbiter breakup at GMT 14:00:18, the *Columbia* cabin pressure was nominal and the crew was capable of conscious actions.

Conclusion L1-2. The depressurization was due to relatively small cabin breaches above and below the middeck floor and was not a result of a major loss of cabin structural integrity.

Conclusion L1-3. The crew was exposed to a pressure altitude above 63,500 feet, indicating that the cabin depressurization event occurred above this altitude.

Conclusion L1-5. The depressurization incapacitated the crew members so rapidly that they were not able to lower their helmet visors.

2.3.1 Depressurization timeline boundaries

The depressurization timeline boundaries were identified with reconstructed telemetry and ground-based video. RGPC-2 data indicate that *Columbia*'s cabin pressure was normal (~14.7 psi) until GMT 14:00:04.826. Therefore, cabin depressurization started no earlier than (NET) GMT 14:00:05. No visual events in ground-based videos were identified positively as evidence of cabin depressurization. However, the videos show that at GMT 14:01:10, the CM image vanishes while it was still clearly in the camera's FOV. The image loss was due to the CM being broken into subcomponents that were too small and dispersed to be visible on the video. This event was defined as TD. After this time, the CM no longer had any structural integrity. Thus, cabin depressurization was complete no later than (NLT) GMT 14:01:10. This analysis establishes the absolute NET and NLT times for the start of the cabin depressurization and the completion of the depressurization. The effort to narrow these boundaries to the maximum extent possible is discussed in subsequent sections.

2.3.2 Start of depressurization

Medical evidence and debris field and ballistic analyses were used to determine the NET and NLT times for the beginning of the cabin depressurization.

Medical evidence suggests that the cabin pressure condition at CE was within the bounds of human survival (see Section 3.4). Therefore, the cabin depressurization started NET the CE at GMT 14:00:18.

> ***Conclusion L1-1.*** After loss of control at GMT 13:59:37 and prior to orbiter breakup at GMT 14:00:18, the *Columbia* cabin pressure was nominal and the crew was capable of conscious actions.

Recovered objects originating from inside the CM are a positive indication of a breach in the CM. The 20 westernmost debris items originating from within the CM were evaluated.[1] All items were small (< 8 in.) in size and none were structural elements of the CM. The debris item located farthest west in the debris field was a piece of reflective tape from a crew helmet. Because this item was not a good candidate for ballistic analysis, the next farthest west item, a mission patch, was used for ballistic analysis. All crew members wear a mission patch attached by VELCRO® to the ACESs; virtually all other patches are stored in plastic-wrapped packages of 25 patches, which are stowed in a stowage compartment below the middeck floor called Volume E. Of the 20 debris items, nine were patches that probably originated from Volume E (Table 2.3-1).

Table 2.3-1. *Westernmost Crew Module Debris*

Item description	CM location
Launch/entry helmet reflective tape	Crew helmet
STS-107 mission patch fragment	Volume E (below middeck floor)
STS-107 mission patch fragment	Volume E
Harness or parachute webbing piece	Crew
STS-107 mission patch fragment	Volume E
STS-107 mission patch fragment	Volume E
Payload patch	Volume E
STS-107 mission patch fragment	Volume E
Payload patch	Volume E
Life raft spray shield fragment	Crew parachute pack
Panel illuminator fragments	Flight deck panel
Middeck ceiling luminous panel	Middeck ceiling
STS-107 mission patch fragment	Volume E
Air duct fragment	Middeck port
Payload checklist fragment	Middeck locker MF43K
STS-107 mission patch fragment	Volume E
Life raft reflective tape	Crew parachute pack
In-flight maintenance tool, no. 0 screw driver	Middeck locker MF43C
Sleep station closeout material	Middeck starboard
Sleep station light cover	Middeck starboard

Ballistic analysis produced a separation time of GMT 14:00:35 ± 5 seconds for the westernmost patch. Therefore, the depressurization of the CM started NLT GMT 14:00:35 ± 5 seconds.

[1] *Columbia* was traveling west-to-east, so debris that was recovered in the western portion of the debris field generally are items that departed from the vehicle earlier than items that were recovered farther east.

2.3.3 Depressurization rate

Debris items and medical evidence were analyzed in the hope that they would aid in determining the cabin depressurization *rate* and thereby aid in determining when the depressurization was complete.

Some intact packages (drink pouches and personal hygiene bottles) were recovered as was the CM cabin altimeter. Rapid depressurization tests were performed on new packages to determine the depressurization rate required to rupture these types of packages. Determining the depressurization rate sufficient to cause rupture also identifies the maximum rate that will *not* cause rupture, yielding an upper bound for the cabin depressurization rate. However, even at the maximum rate that the test chamber provided (almost 32 psi/sec[2]), the packages did not rupture, so an upper bound for the cabin depressurization rate could not be determined from these tests.

The recovered middeck cabin altimeter was disassembled and compared to a new cabin altimeter to determine whether the recovered altimeter contained any evidence of the cabin pressure environment. No pressure-related differences were noted between the recovered altimeter and the new altimeter. Neither the packages nor the altimeter analyses could provide any conclusions regarding depressurization rates.

Medical forensic evidence was studied to determine the rate of the cabin depressurization. Information on the effects of a rapid depressurization to vacuum is limited to postmortem analysis of isolated accidental occurrences and animal studies. A literature search revealed a case of apparent lung trauma occurring at slow depressurization rates.[3] Additionally, information on the fatal depressurization accident of *Soyuz 11* in 1971 revealed that although the *Soyuz 11* cabin depressurization was relatively slow (reportedly taking more than 3.5 minutes to depressurize to 0 psi), it was reported that the depressurization was fatal to the *Soyuz* crew in roughly 30 seconds.[4] Further research indicates that the specific circumstances (depressurization rates, the magnitude of the pressure differentials, absolute pressures, etc.) that result in the type of depressurization-related tissue damage seen in the *Soyuz 11* and *Columbia* crews have not been fully characterized. Because the exact scenario cannot be positively identified, no conclusions with respect to cabin depressurization rates or timing can be made from the medical findings.

The 51-L *Challenger* accident investigation showed that the *Challenger* CM remained intact and the crew was able to take some immediate actions after vehicle breakup, although the loads experienced were much higher as a result of the aerodynamic loads (estimated at 16 G to 21 G).[5] The *Challenger* crew became incapacitated quickly and could not complete activation of all breathing air systems, leading to the conclusion that an incapacitating cabin depressurization occurred.[6] By comparison, the *Columbia* crew experienced lower loads (~3.5 G) at the CE. The fact that none of the crew members lowered their visors[7] strongly suggests that the crew was incapacitated after the CE by a rapid depressurization.

Although no quantitative conclusion can be made regarding the cabin depressurization rate, it is probable that the cabin depressurization rate was high enough to incapacitate the crew in a matter of seconds.

> ***Conclusion L1-5.*** The depressurization incapacitated the crew members so rapidly that they were not able to lower their helmet visors.

[2]This rate would result in a shuttle cabin depressurization in less than half a second – a scenario that is contradicted by debris and video evidence. The shuttle cabin depressurization rate was probably an order of magnitude less than 32 psi/sec.
[3]"Survival Following Accidental Decompression to an Altitude Greater than 74,000 Feet (22,555m)."
[4]http://history.nasa.gov/SP-4209/ch8-2.htm.
[5] JSC 22175, STS-51L, JSC Visual Data Analysis Sub-Team Report, Appendix D9, June 1986.
[6]Report from Dr. Joe Kerwin to Rear Adm Truly, http://history.nasa.gov/kerwin.html, July 28, 1986.
[7]See Section 3.2, Crew Worn Equipment.

2.3.4 Depressurization complete

Because the depressurization rate was concluded to be high enough to incapacitate the crew within seconds, the depressurization complete NET time is some number of seconds after the earliest initiation of the depressurization, (i.e., several seconds after GMT 14:00:18). However, no direct debris evidence or analysis provided conclusive results that could refine the NET time for when the depressurization could have been completed.

Debris field analysis, ballistics analysis, and video evidence were used to provide the NLT time for when the depressurization completed. The presence of significant CM structural debris items in the debris field indicated a loss of structural integrity of the CM and, therefore, the inability to maintain cabin pressure. Ballistic analysis on a middeck floor panel indicates a release time of GMT 14:01:02 ± 5 seconds. More than 200 pieces of CM structure were recovered west of this item, strongly suggesting that the CM lost structural integrity prior to this time. Video evidence indicated that major changes in the appearance of the CM and significant debris shedding occurred from GMT 14:00:58 to GMT 14:00:59. It is probable that the CM lost structural integrity and was fully depressurized NLT GMT 14:00:59. The CM was estimated to be over 135,000 feet altitude at GMT 14:00:59, so the crew was exposed to a high-altitude environment. Figure 2.3-1 shows the depressurization timeline with the start of depressurization NET and NLT times and the NLT time for when the cabin depressurization was complete.

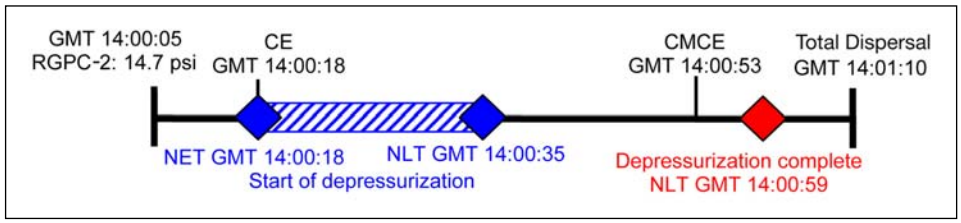


Figure 2.3-1. ***Cabin depressurization timeline.***

Conclusion L1-3. The crew was exposed to a pressure altitude above 63,500 feet, indicating that the cabin depressurization event occurred above this altitude.

2.3.5 Location(s) of the cabin breach(es)

Structural analysis was performed to evaluate the overall loads for the CM to determine whether a structural breach occurred in the CM due to forces during the vehicle LOC. The CM was a strong pressure vessel that was designed to withstand the loads of a crash landing. Aerodynamic modeling of the vehicle provided loads for the period from LOS (GMT 13:59:32) to the CE. Deceleration loads were increasing and peaked at 3.5 G at the CE. None of the loads exceeded the CM crash loads structural limits, but CM structural deformation was possible. Impacts between the CM and the FF during and shortly after the CE likely caused damage to the CM, resulting in a depressurization.[8] Internal damage from the sudden load change also accounts for objects breaking free from Volume E and other stowage areas.

Depending on the size and location of breaches in the CM, a depressurization can result in differential pressures across the flight deck and middeck floors. The middeck floor had very few openings; the vent path between the middeck volume and the lower equipment bay consisted primarily of narrow gaps around the panels and access doors. Totaling all the gaps, this venting area was approximately 50 in^2. The middeck floor structure could withstand a differential pressure of 0.32 psi without suffering deformation.[9] A cabin depressurization computer model was used to determine the maximum hole sizes that would exceed the

[8]See Section 2.4, Forebody Breakup Sequence.
[9]Rockwell internal letter SAS/AERO/88-469.

capability of the floor.[10] This model predicted that a hole larger than 12.4 in. diameter (or several smaller holes equivalent to 12.4 in.) *above* the middeck floor, or a hole larger than 4.8 in. in diameter *below* the middeck floor would result in a differential pressure across the floor greater than 0.32 psi (figures 2.3-2 and 2.3-3).

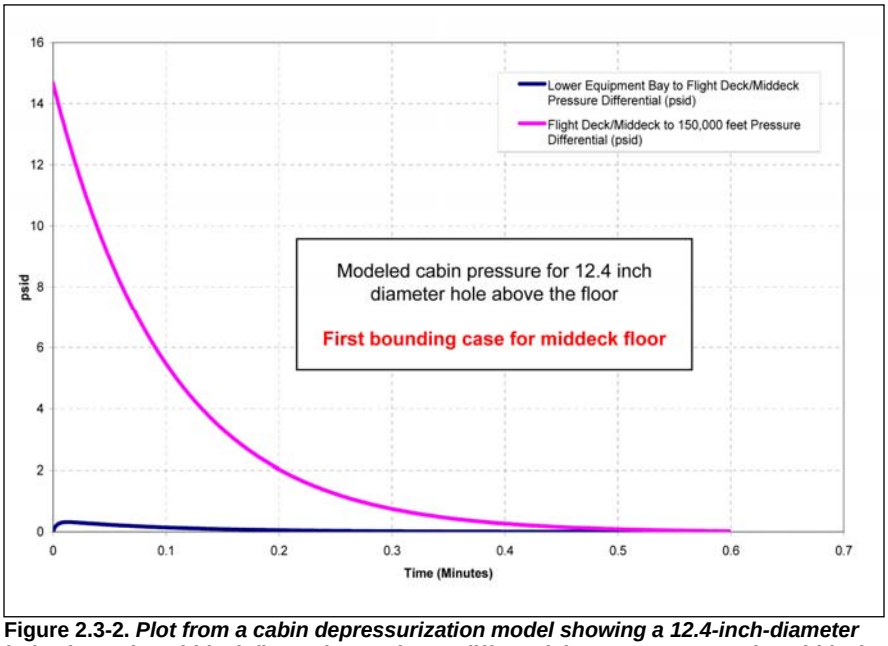


Figure 2.3-2. ***Plot from a cabin depressurization model showing a 12.4-inch-diameter hole above the middeck floor*****. The maximum differential pressure across the middeck floor is 0.317 pounds per square inch (within middeck floor capability).**

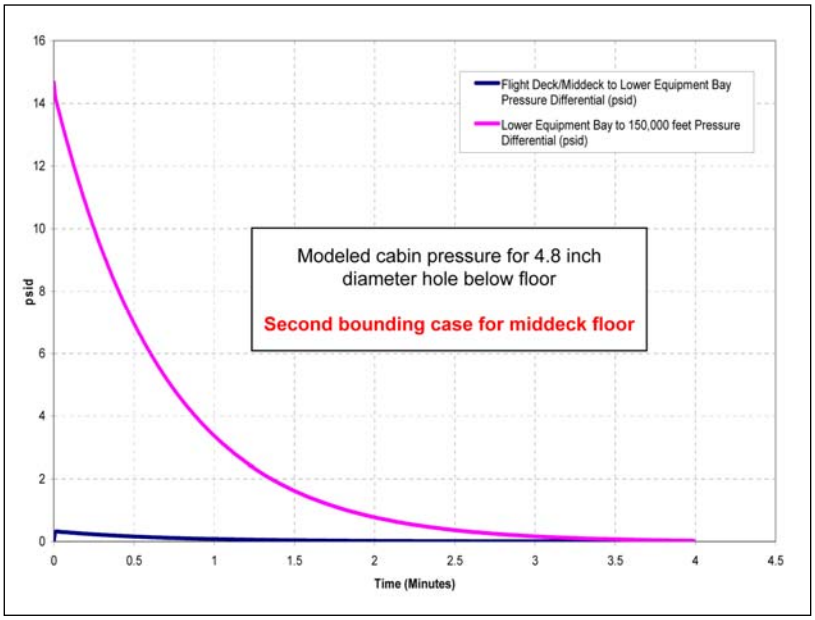


Figure 2.3-3. ***Plot from a cabin depressurization model showing a 4.8-inch-diameter hole below the middeck floor*****. The maximum differential pressure across the middeck floor is 0.319 pounds per square inch (within middeck floor capability).**

[10]The shuttle life support system includes emergency regulators that can maintain the cabin pressure at 8 psi in the event of a cabin breach. The combined maximum flow rate for the system is less than 200 lbs./hour. Cabin breaches discussed in this report were much larger (> 50,000 lbs./hour) and would have overwhelmed the system. Additionally, because the O_2 and N_2 supply tanks were separated from the forebody at the CE, the cabin depressurization analysis presented here does not include gas introduction from the emergency system.

More than 65% of the middeck floor was recovered, and there was no evidence of buckling due to differential pressure. This indicates that the CM depressurization rate did not exceed the structural capability of the middeck floor.[11] Therefore, the cabin depressurization was not caused by an instantaneous hole above the middeck floor larger than 12.4 in. diameter (or several holes, all above the middeck floor, with a combined effective hole size of 12.4 in. diameter). Additionally, the depressurization was not due to an instantaneous hole (or several holes combined) below the middeck floor larger than 4.8 in. diameter. However, holes on both sides of the floor could cause cabin depressurization to occur more quickly and still not cause differential pressures sufficient to damage the floor. The analyses assumed that the hole sizes remained constant; if the holes were to enlarge gradually, a depressurization could occur faster without exceeding the middeck floor's 0.32 pounds per square inch differential (psid) capability.

Based on the fact that the westernmost CM debris field contained items originating from stowage volumes below the middeck floor and from areas above the middeck floor, and the lack of evidence of floor deformation, it is probable that the cabin breach involved holes above and below the middeck floor. The small size of the items and the distinct lack of CM shell structure or elements of heavy internal structure suggest that the individual breaches were not large.

The CE was the most probable time at which a structural failure would occur that would result in structural warping of the CM and/or CM/FF impacts, resulting in one or more breach locations. This is consistent with the previous conclusion that the crew was conscious at the time of the CE (GMT 14:00:18).

> ***Conclusion L1-2.*** The depressurization was due to relatively small cabin breaches above and below the middeck floor and was not a result of a major loss of cabin structural integrity.

2.3.6 Synopsis of crew cabin pressure environment analysis

Prior to the CE (GMT 14:00:18), the *Columbia* cabin pressure was nominal and the crew was capable of conscious actions. The CM depressurization began NET GMT 14:00:18 and NLT GMT 14:00:35, and was due to cabin breaches above and below the middeck floor. The depressurization rate was high enough to incapacitate the crew members within seconds such that they were unable to perform actions. Although the CM lost structural integrity and was fully depressurized no later than GMT 14:00:59, it is highly probable that the depressurization was complete earlier.

[11]The flight deck floor can be damaged by a differential pressure greater than 0.81 psi. Due to the much larger venting area between the flight deck and the middeck, holes much larger than 12.4 in. diameter would be required to deform the flight deck floor. Very little flight deck floor debris was recovered, so no conclusions regarding the deformation of the flight deck floor could be made.

2.4 Forebody Breakup Sequence

This section discusses the breakup of the forebody of *Columbia*. The findings in this section are based on ground-based video analysis, ballistic calculations, cluster analysis of the debris field, and structural analysis.

The format of this section follows the format of Section 2.2, Orbiter Breakup Sequence. *It is strongly recommended that the reader review Section 2.2 before reading this section*. Complete descriptions of the types of analysis can be found in the introduction of Section 2.2 and are not repeated here.

Based on video, the forebody broke away from the intact orbiter at the CE, GMT 14:00:18. Because the forebody's ballistic number was significantly higher than that of the intact orbiter (more dense, with less drag), at the moment of separation the deceleration due to drag decreased suddenly. The loads experienced by the forebody dropped from the peak load of approximately 3.5 G just before the CE to approximately 1 G after the CE. As the forebody began its own unique ballistic trajectory, deceleration forces began to build again. Additionally, the forebody was rotating in all axes at approximately 0.1 rev/sec with an increasing rate (see Section 2.1).

The forebody breakup was initiated at the CMCE at GMT 14:00:53. This was the beginning of a sequence of events resulting in the dispersal of the forebody into multiple smaller components. NLT GMT 14:01:10, the CM had lost all structural integrity and had been broken into subcomponents. This time was defined as the TD.

The following findings and recommendation are in this section:

Finding. The *Columbia* windows remained largely intact up until the CMCE and were not a cause of cabin depressurization.

Finding. Windows 7 and 8 experienced a titanium deposition event that occurred prior to window breakup.

Finding. The most probable source for the titanium deposition on Windows 7 and 8 was PLBD rollers. These rollers were not exposed to heat flow until after the PLBDs were compromised.

Finding. All the windows had an aluminum-rich deposition, which was consistent with a turbulent process.

> ***Recommendation L3-1.*** Future vehicles should incorporate a design analysis for breakup to help guide design toward the most graceful degradation of the integrated vehicle systems and structure to maximize crew survival.

2.4.1 Ground-based video analysis

This section addresses the detailed analysis of ground-based video related to the forebody after separation from the orbiter at the CE. This section only covers major events seen in the video that occur relative to the forebody breakup, including the CMCE and the TD.

A video triangulation analysis on the motion of the free-flying forebody was discussed in Section 2.1. A different relative motion analysis related to the TD is discussed here.

Catastrophic Event to Crew Module Catastrophic Event

It was initially anticipated that the CM depressurization could be identified in videos as a halo or other visible effect. However, detailed review of the video showed a visually complex event at the CE. The forebody was not visually distinct from the rest of the orbiter's pieces until about 8 seconds after it separated from the intact orbiter. It is unknown whether a depressurization event would be visible; but if it was and it occurred during this time, any indication of the event would be lost in the merged signals of the orbiter's pieces. None of the changes in appearance at the CE or afterward could be positively identified as a depressurization event (see Section 2.3).

Crew Module Catastrophic Event

Video analysis established the precise time for the CMCE (the initiation of the breakup of the forebody) as GMT 14:00:53. Thermal and ballistics analyses of forebody debris items were consistent with this, supporting the video-based time.

At the CMCE, the video shows that the forebody began to visibly brighten and the envelope of gases around the forebody appear to increase in size. This brightening event was followed quickly by a significant debris shedding event that was likely to be the FF and the CM separating from one another.

Figure 2.4-1 shows a set of five paired images covering less than 2 seconds of time. On the left is the original image and on the right is a close-up of the forebody and CM. The first frame shows when the forebody begins to brighten. The second frame, which was taken one-third of a second later, shows the slight increase in apparent size of the envelope of gases of the orbiter. The third image, almost 1 second later than the previous image, shows the first definitive indication (although it can be recognized a few frames earlier) that the forebody is beginning to fail. The magnified image has been inverted to emphasize the split in the trail of the forebody. The next frame, which occurs about one-third of a second later, shows what are believed to be two significant portions of the FF separating from the CM. The last image, taken one-fifth of a second later, shows how quickly the FF is breaking into pieces that are too small to be seen by the camera. The CM breaks up over the next 16 seconds.

Total Dispersal

As objects separated from each other during the breakup, each took on its own trajectory based on its unique ballistic number. The high initial speed immediately resulted in a wide dispersion of trajectories as lighter and smaller items decelerated very rapidly, while heavier and larger items decelerated less and, hence, traveled farther. Evaluation of the debris appearance confirmed that very little debris-to-debris interaction (impacts) occurred. As subcomponents decelerated, the entry heating began to decrease quickly, resulting in a loss of visual signal in the video. The CM image vanishes while it was still clearly in the camera's FOV. The image loss was not due to a major deceleration taking the intact CM out of the frame, but was due to the CM being broken into subcomponents that were too small and dispersed to be visible on the video. This event was described as the TD. After this time, the CM no longer had any structural integrity.

It should be noted that cascading structural failures were still occurring following the TD, as well as frictional heating on individual objects with high ballistic numbers that decelerated more slowly. These ongoing failures could not be seen on video.

Figure 2.4-1. ***Five time-paired images covering under 2 seconds of time showing the Crew Module Catastrophic Event*****. Image on the left is the full frame, image on the right is an enlarged view.**

Figure 2.4-2 shows the last few seconds of the CM as seen in the Apache video, ranging from GMT 14:01:06 through GMT 14:01:09. The images on the right side are a magnified view of the original images on the left. The CM has been circled in red.

As the images illustrate, when structural integrity of the CM completely failed, it did so in a fraction of a second. The times for the images (figure 2.4-2), from top to bottom, are GMT 14:01:06.73, GMT 14:01:06.87, GMT 14:01:06.97, GMT 14.01.08.3, and GMT 14:01:09.67. In the next frame, the CM is no longer visible.

Figure 2.4-2. ***Apache video of the Total Dispersal, spanning from GMT 14:01:06.73 to GMT 14:01:09.67*****. The forebody/crew module is circled in red.**

Relative motion analysis

Relative motion analysis compares the rate of change of the movement of objects in the FOV of a video. Rate of change can provide an estimation of the G-load that is experienced by the bodies within a relative frame of reference; e.g., a relative difference might be that one object experiences a deceleration of 3 G

relative to another object. While the analysis indicates that the first object experiences three more Gs than the other object, it does not define how many total Gs either object is actually experiencing.

Two relative motion analyses were performed. The first regarded a triangulation of the motion of the forebody relative to the engines between the CE and the CMCE (see Section 2.1).

The second relative motion analysis was performed on the CM (identified as D21) and the aft engines post-CMCE in a stabilized Apache video (figure 2.4-3).

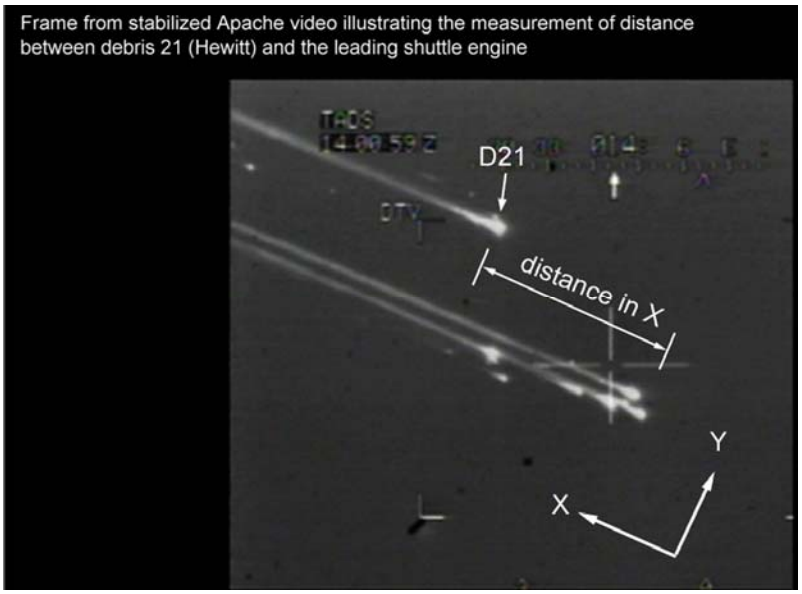


Figure 2.4-3. ***An anayzed frame from the Apache video.***

The relative motion of the CM to the engines appears to suggest a high deceleration event occurred during the breakup of the forebody (between the CMCE and the TD). Later understanding of the breakup sequence revealed that this deceleration was related to the disintegration of the CM and the resulting cloud of debris. Rather than a specific high-G event that was experienced by an intact CM, it represents the cloud of individual items separating and rapidly decelerating. Heating was sufficient to keep the individual objects visible for a short period of time, and these items were close enough together that they could not be distinguished as separate items. This also explains why the CM's visible disappearance occurred in a fraction of a second, as the deceleration passed the threshold of sufficient heat generation for visibility in the video.

2.4.2 Ballistic analysis

Refer to Section 2.2.2 for a full description of the techniques, assumptions, and limitations for ballistic analyses.

Forebody structure

Table 2.4-1 shows the debris ballistic timeline for some selected forebody structures. Ballistic analysis could not be done on all recovered debris because ballistic numbers are hard to estimate for irregularly shaped objects and the analysis is time-intensive. For some structures, there is a major structural release time. For other objects, the major structural release time is the same as the debris object release time since there is only one object.

Table 2.4-1. ***Ballistic Timeline for Post-Crew Module Catastrophic Event Forebody Events***

Major Structural Release Time (GMT)	Vehicle Structure	Debris Object Number and Time of Release (GMT)
14:00:54	FF	11119 (14:00:51) 81231 (14:00:55) 26099 (14:00:55)
14:00:57	Forward RCS strut	2170
14:00:59	Forward RCS thruster	2167
14:01:05	Thermal pane glass (outer pane)	14:00:57 (65012) 14:01:02 (68534) 14:01:04 (1574) 14:01:05 (73192) 14:01:16 (77517)
14:01:07	FF port ejection panel[1]	51987
14:01:12	Nose cap	257
14:01:13.5	Forward RCS helium tank	14:01:13 (1481) 14:01:14 (1209)
14:01:35	FF starboard ejection panel	71801

These individual release times alone did not provide significant insight into the breakup event, but will be compared to other analyses in this section. However, the times generally span the video determination for the time between the CMCE and the TD. Although a few items appear to have been released after the TD, cascading failures were expected to occur even after the TD, and this accounts for those times being later (see Section 2.2.2).

Crew module interior items

Objects that are discussed in this section originated from inside the CM but are not directly associated with a seat or a crew member. All of the items listed were selected because their shapes were easily modeled for ballistic analysis, and all are believed to have been stowed or located in the middeck. No flight deck structural debris items were good candidates for ballistic analysis because of their irregular shapes. For a detailed discussion of individual crew seat and crew equipment recovery, see Sections 3.1 and 3.2. The results of that ballistic analysis concluded that the middeck seats and equipment were released prior to the flight deck.

The release times and descriptions of these interior items are in Table 2.4-2. The mission patch (debris item no. 31539) in this table should not be confused with the separate mission patch that is identified in the cabin depressurization analysis, which was one of the most westerly objects recovered.

[1] *Columbia* was the only orbiter in the fleet with ejection hatches for the Commander and Pilot stations. The ejection seat systems were disabled and the ejection hatches were deactivated when the orbiter program was deemed operational. Eventually, the ejection seats were removed from *Columbia*, but the hatches remained as an integral part of the structure of the FF.

Table 2.4-2. ***Individual Debris Objects for the Crew Module Post-Crew Module Catastrophic Event***

Release Time (GMT)	Debris Number	Description	Shape
14:00:56	55952	Toilet handle ball	sphere
14:01:01	31539	STS-107 mission patch	triangular
14:01:02	1170	Middeck floor section	rectangular
14:01:07	1155	Middeck accommodation rack (MAR)	rectangular
14:01:07	2499	Middeck floor section	rectangular
14:01:08	44199	Volume E access door	rectangular
14:01:10	23262	Locker door	rectangular
14:01:10	7717	Photo TV floodlight	rectangular
14:01:13	7662	Light emitting diode (LED) indicator panel	square
14:01:15	8820	Window shade bag	rectangular

Again, no major conclusions could be drawn from these data alone. However, like the structures ballistic analysis, these times are consistent with the times for the CMCE and the TD as determined by video.

2.4.3 Cluster analysis

For a complete discussion of the techniques, assumptions, and limitations of cluster analysis of the debris fields, see Section 2.2.3.

Catastrophic Event to Crew Module Catastrophic Event

The most significant finding from debris field analysis showed that the FF and CM appeared to remain relatively intact from the CE to the CMCE, a period of 35 seconds (see Section 2.2.3). Small amounts of forebody structure, such as some nose landing gear door tile and small structure, were released earlier. However, 87% of the FF structure and all of the CM pressure vessel structure debris clusters overlap completely (figure 2.4-4), strongly suggesting a relationship between their structural failure. The western end of the main portion of the forebody structure debris field is at a longitude of 94.5W.

The fact that the forebody maintained structural integrity immediately after the orbiter breakup may be explained by the reduction in deceleration loads occurring at the CE due to the change in ballistic number. However, thermal effects would begin to increase, and deceleration loads once again began to build up to a peak of 3.5 G at the forebody c.g. at the CMCE. Note that loads experienced at the farthest outboard edges of the forebody could be as high as 10 G to 12 G at the CMCE due to the moment arm from the c.g. as the forebody rotated.

Cabin pressurization

Cabin depressurization was an important event to identify. The initiation of the depressurization of the cabin could not be observed in the video. However, the debris field provided the opportunity to evaluate when depressurization may have started, because items that were originally stowed inside the CM and recovered west of the main forebody debris field could indicate a CM breach.

Items that were stored on board the orbiter were carefully packed, stowed, and documented prior to launch. Some items on STS-107 were stowed in the middeck and the SPACEHAB (laptop computers and LiOH canisters). Most items also had a designated entry stowage location. Common use items (such as pens and

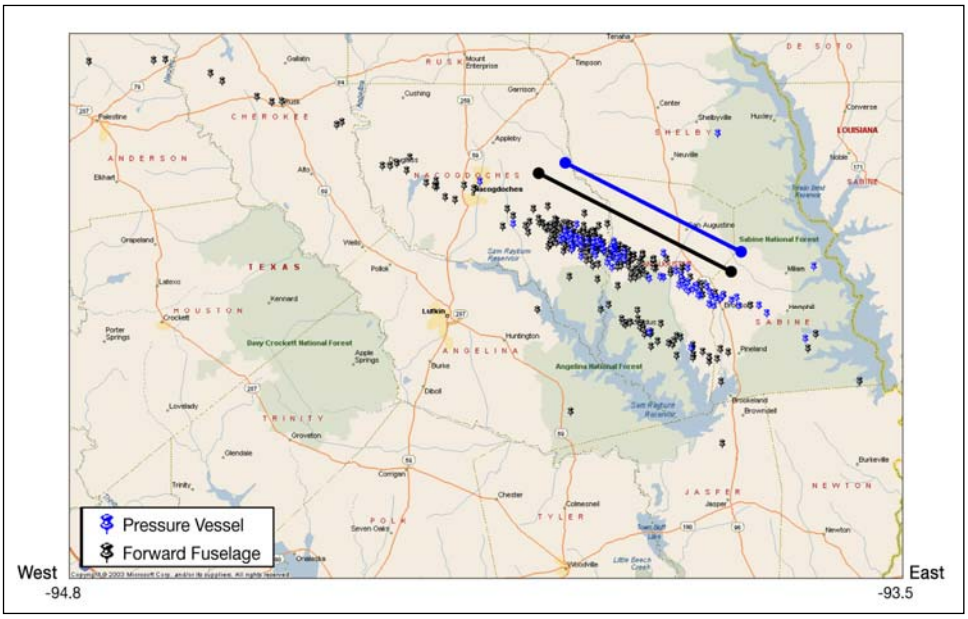


Figure 2.4-4. ***Debris field of the forward fuselage and crew module pressure vessel.***

pencils) are not carefully tracked and not necessarily re-stowed to launch configuration for entry. In particular, trash was stored in multiple areas such as the SPACEHAB, the airlock, and the middeck volumes, some of which extend into the lower equipment bay below the middeck floor. The SCSIIT chose to investigate only nonstructural items with accurately known storage locations. No CM pressure vessel structure or significant interior structure was recovered west of the main forebody debris field. Nonstructural items that were recovered west of the main forebody debris field were assumed to have been evacuated from the CM due to decreasing pressure and a small breach in the CM rather than major structural degradation. Study of these objects allowed the team to better understand when pressure inside the CM was lost, and to determine the areas where suspected breaches might have occurred.

Many "crew equipment" items that are listed in the database west of longitude 94.5W may not have originated from *Columbia*. Because they represent common use personal items, they cannot positively be identified as from the orbiter and may be discarded items that were present in the debris field. The items that can confidently be concluded came from the orbiter include empty food packets, LiOH canisters, clothing, and laptop computer debris. The bulk of these items were stored in the middeck lockers, the lower equipment bay volumes, the SPACEHAB, the airlock, and the Waste Collection System (WCS).

Seventy-one items were recovered west of longitude 94.6W and were positively identified from the middeck, flight deck, or CEE. The specific locations for these items were Volume E, middeck lockers MF43K and MF43C, port middeck and stowage volumes from the lower equipment bay, flight deck illuminator panels (i.e., acrylic sheeting), and equipment from the flight deck. These items were all small (< 8 in. and, in most cases, much smaller).

In general, small and light items did not travel downrange significant distances due to the ballistic properties of the object. Some paper/lightweight items were offset slightly northeast of the main debris footprints, probably due to the prevailing winds at the time of the accident. Figure 2.4-5 shows the debris field for nonstructural internal CM debris.

Many of the westernmost debris items came from internal stowage volumes. This implies that internal structural damage occurred at the CE. A full description of cabin depressurization is contained in Section 2.3. The integrated assessment concluded that depressurization occurred NET GMT 14:00:18 (CE) and NLT GMT 14:00:35. It ended NLT GMT 14:00:59, and most likely earlier.

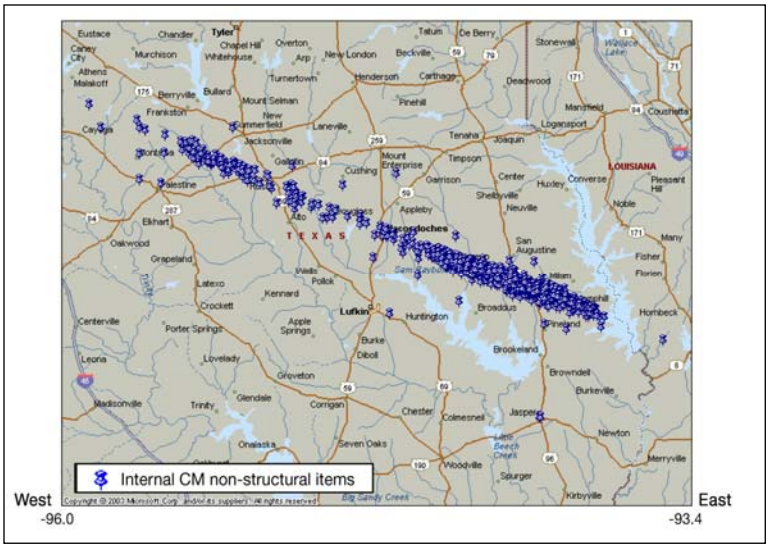


Figure 2.4-5. ***Debris field of the nonstructural items from inside the crew module pressure vessel.***

Window panes

A perceived vulnerable area of the forebody was the windows. Most of the windows in the forebody are triple-paned. Only the two aft windows, which are protected by the PLBDs during entry, are double-paned. For the three-paned windows, the exterior panes are "thermal" panes that provide thermal protection. The inner panes are "pressure" panes that provide structural support for the pressure inside the CM. The middle pane is a "redundant pane" as it is intended to be redundant for both the thermal and the pressure pane. Figure 2.4-6 shows the schematic of the three-paned window of an orbiter.

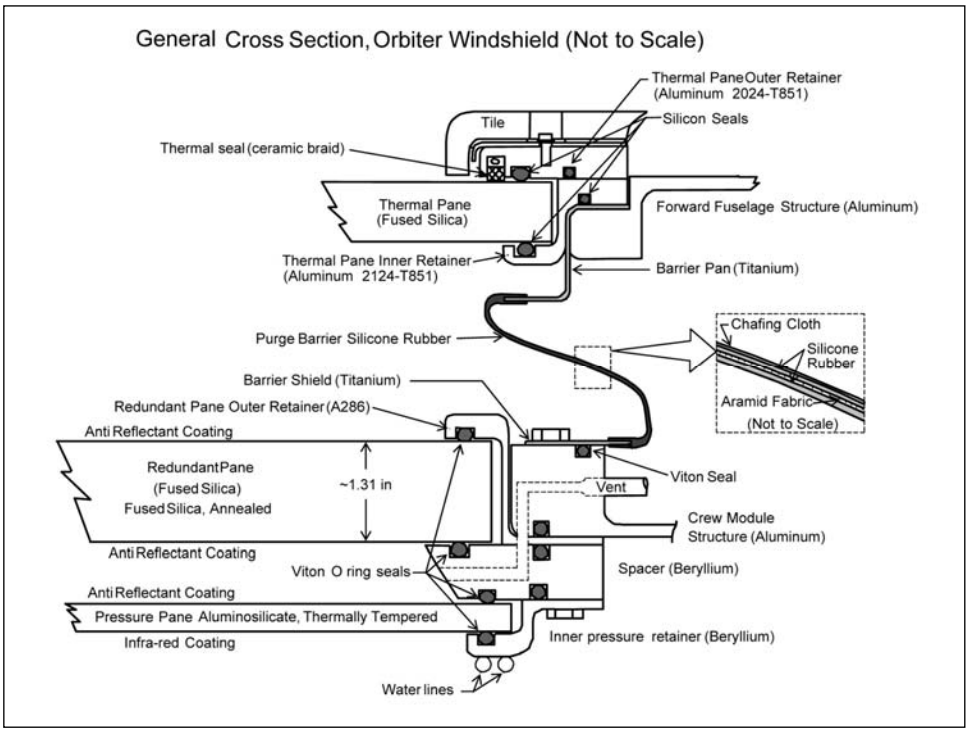


Figure 2.4-6. ***Schematic of the orbiter three-paned window.***

Figure 2.4-7 shows the numbering system for the windows; the forward windows are numbered 1 through 6, the overhead windows are 7 and 8, and the aft-facing windows are 9 and 10. Window 11 (not shown) is the side hatch window. The thermal, redundant, and pressure panes vary in thickness depending on location on the vehicle. Table 2.4-3 shows the specified thicknesses for the various panes from the 11 different windows.

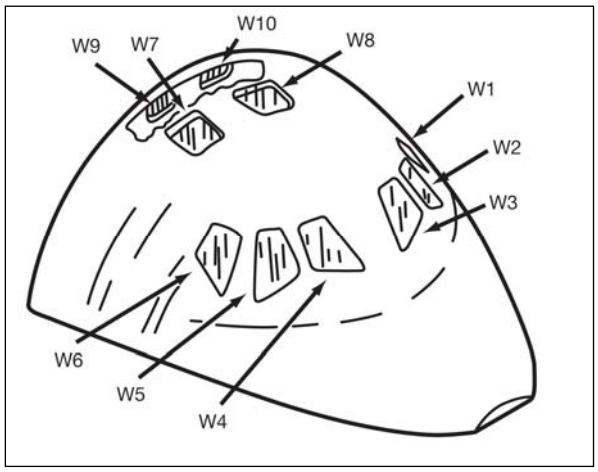


Figure 2.4-7. ***Window numbering system*****.**

Table 2.4-3. ***Specified Orbiter Window Design Thicknesses***

Size (in.)	Window pane
0.29	Side hatch thermal pane
0.33	Windows 9, 10 redundant pane (tempered) Windows 9, 10 pressure pane (tempered)
0.47	Windows 7, 8 redundant pane (tempered) Windows 7, 8 pressure pane (tempered)
0.49	Windows 7, 8 thermal pane Side hatch redundant pane
0.56	Windows 1, 6 thermal pane
0.61	Windows 2, 5 thermal pane
0.62	Side hatch pressure pane
0.63	Windows 3, 4 pressure pane (tempered)
0.65	Windows 1, 2, 5, 6 pressure pane (tempered)
0.69	Windows 3, 4 thermal pane
1.30	Windows 3, 4 redundant pane
1.32	Windows 1, 2, 5, 6 redundant pane

The orbiter windows are made of a compositionally unique fused silica that is highly thermally stable and not expected to thermally degrade under entry heating. Material analysis was not a primary means of assessment due to limited resources. Recovered glass was identified by measuring the thickness and comparing it to the data shown in Table 2.4-3. The highest confidence was in the approximately 1.3-in.-thick redundant panes of glass since this is not a common commercially available thickness. Due to the shattered nature of the tempered-glass fragments, further identification was difficult for loose fragments (those not retrieved from within the frames).

Initially, the windows were considered as a potential site of cabin breach due to thermal failure. The thermal panes, while designed to withstand thermal conditions, are not normally exposed to the highest entry heating conditions because of the geometry of the vehicle and its nominal attitude during the heating phase on entry. However, a preliminary thermal analysis showed that the structure around the windows would fail thermally before the window panes.

A different concern was whether the flexing of the structure under the varying loads at the LOC and the CE would cause the glass to shatter. The debris field analysis refuted that theory as well. Figure 2.4-8 shows the debris field coordinates for recovered glass.

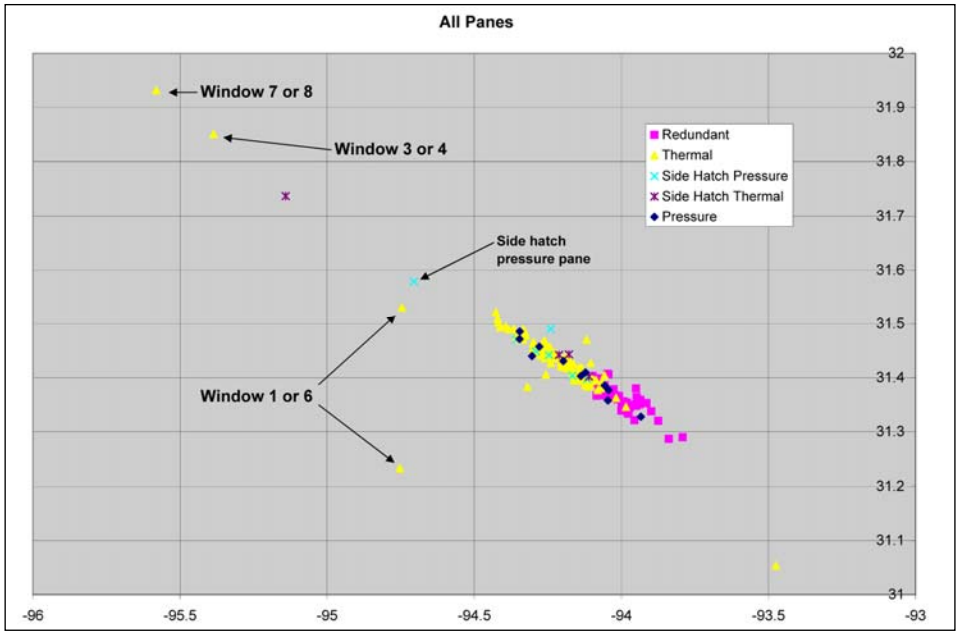


Figure 2.4-8. ***Recovered debris field for window glass*.**

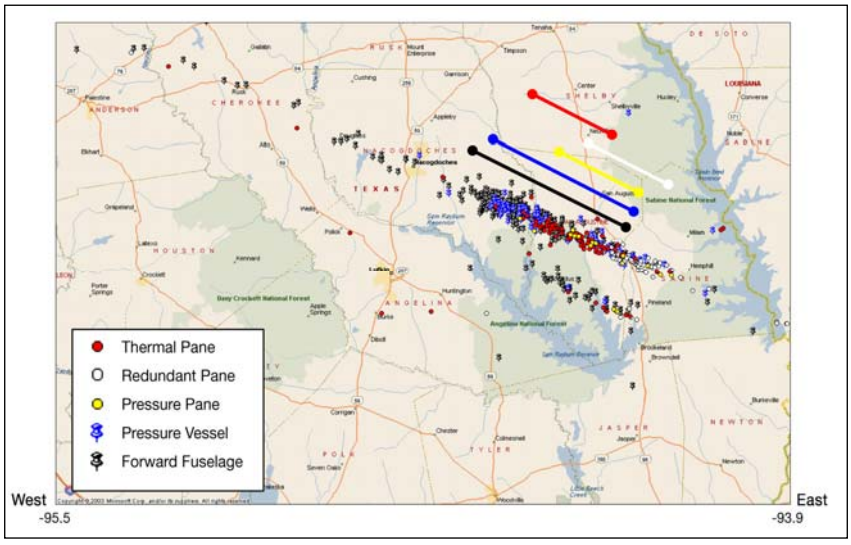


Figure 2.4-9. ***Debris field of the window panes, forward fuselage, and crew module pressure vessel*.**

Of the 201 recovered glass fragments, only six pieces of loose glass that are confidently believed to be from the orbiter were found west of the main forebody debris field. All were thermal (outer) panes with the exception of one piece of probable side hatch pressure pane. However, a piece of pressure pane that was recovered in the western end of the debris field does not alone indicate a source of depressurization. Pressure panes may be damaged due to objects moving around inside the CM as it rotated, impacting the windows. If the cabin is depressurized from a different source, these pieces can evacuate through other breaches.

If the source of the cabin depressurization had been a window, both the pressure pane and the redundant pane would have to have failed. However, redundant pane is not seen until the far eastern end in the forebody debris field. Also, in such a case, the pressure from inside the CM would likely blow the entire pane out and result in significant amounts of glass that were not seen west of the main forebody debris field.

Figure 2.4-9 shows the window debris field overlaid on the FF and pressure vessel debris fields.

Figure 2.4-9 shows that thermal pane debris was recovered east of the beginning of the main forebody debris, suggesting that the windows broke at or after the CMCE. Pressure panes were recovered west of the bulk of the redundant panes, possibly because the redundant panes were extraordinarily thick and heavy and may have traveled farther east because their ballistic number was higher. Determining the failure sequence of the windows based on ground plots alone was somewhat suspect because some of the window frames remained attached to other window frames as a unit, and the shape of the windows can potentially generate lift. Regardless, because the debris field does not contain significant pane debris west of the main CM debris field, it is concluded that the windows were not the source of a CM breach or a part of the initiating event of the forebody breakup.

Finding. The *Columbia* windows remained largely intact up until the CMCE and were not a cause of cabin depressurization.

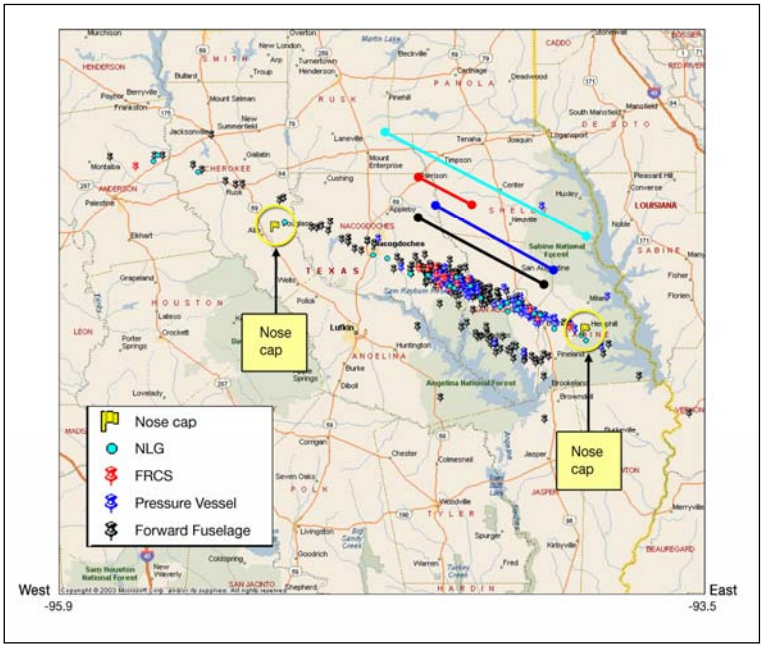


Figure 2.4-10. ***Debris field of the forebody components*.**

Post-Crew Module Catastrophic Event forebody breakup

This section discusses the debris field clusters related to the CMCE and the breakup of the forebody. Figure 2.4-10 shows the major elements of the forebody including the CM pressure vessel, FF, forward RCS, nose, and nose landing gear debris. Figure 2.4-11 shows greater detail regarding specific items and highlights the scarcity of items recovered prior to the main FF/CM debris field.

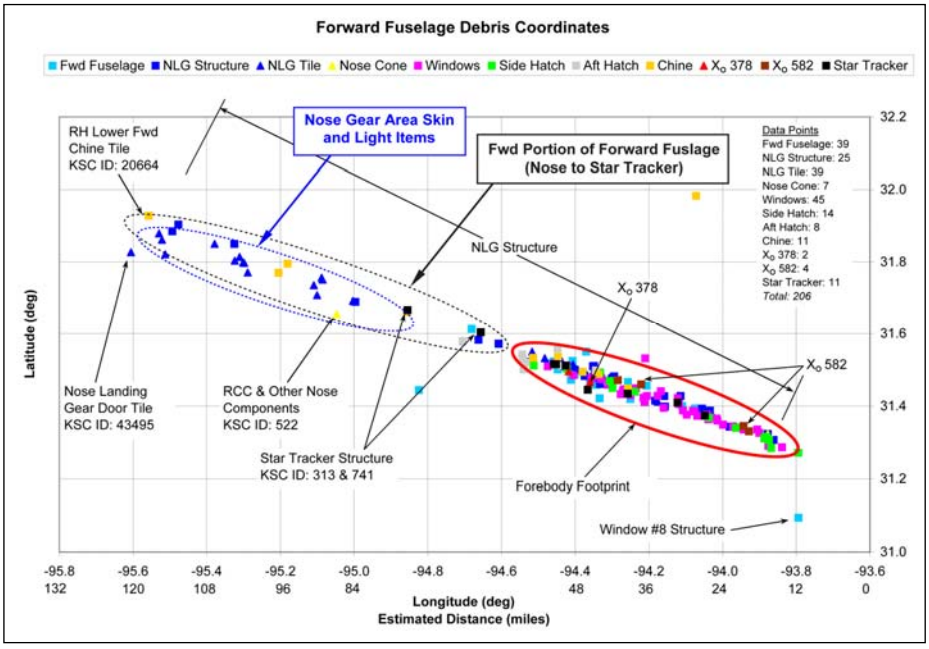


Figure 2.4-11. ***Forward fuselage debris field*.**

The debris field analysis suggests that the order of separation was as follows: nose landing gear structure and nose cap, FF, and forward RCS, followed by the CM pressure vessel. This is consistent with the video that appeared to show FF separating, followed very quickly by failure of the CM.

Crew module interior structure

The CM interior structure discussion includes the flight deck, middeck, and airlock. *Columbia* was the only orbiter with an internal airlock, meaning that the airlock structure was located inside the middeck of the CM rather than in the payload bay like the other orbiters.

Middeck and flight deck

Figure 2.4-12 shows the distribution of middeck and flight deck structure. The middeck structure appears farther west in the debris field, while the flight deck debris cluster is concentrated at the eastern end. This suggests that the flight deck remained intact longer and traveled farther downrange than the middeck.

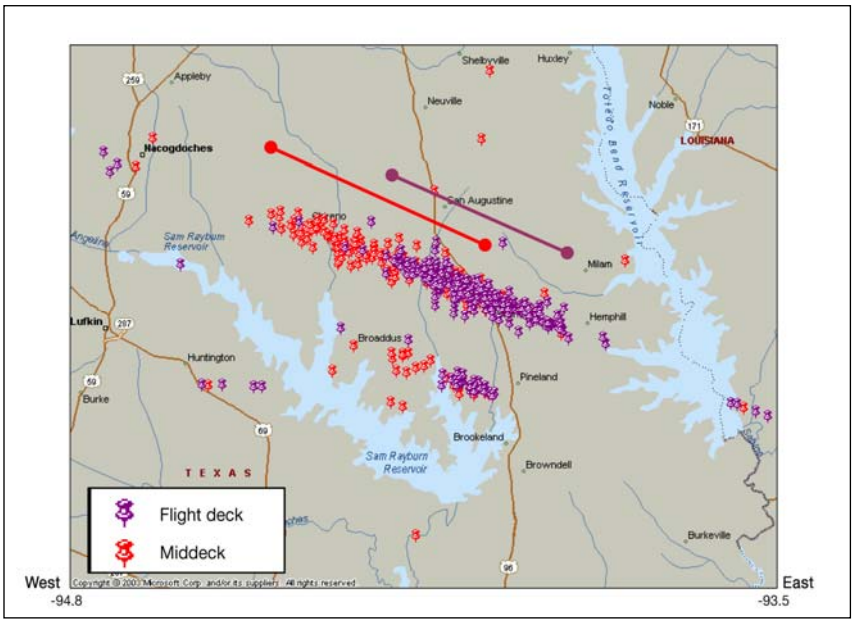


Figure 2.4-12. ***Debris field of the middeck and the flight deck.***

This conclusion is supported by the fact that almost all of the middeck floor debris was in fair to very good condition. However, very little of the structurally stronger flight deck floor was recovered. Furthermore, the flight deck items that were recovered were heavily thermally eroded. This would be the case if the flight deck stayed together and received more thermal damage as a result of having a higher ballistic number than the individual middeck components.

Ground debris footprints for the crew worn equipment and seats are not addressed in this section. The analyses and conclusion from the seat and suit assessments (see Sections 3.1 and 3.2) also concluded that the middeck elements separated before the flight deck elements. This supports the conclusions from the structural debris fields.

The debris field was also evaluated based upon whether debris items came from the port or starboard side of the orbiter. Figure 2.4-13 shows the middeck structure debris field. Figure 2.4-14 shows the flight deck panel debris field.

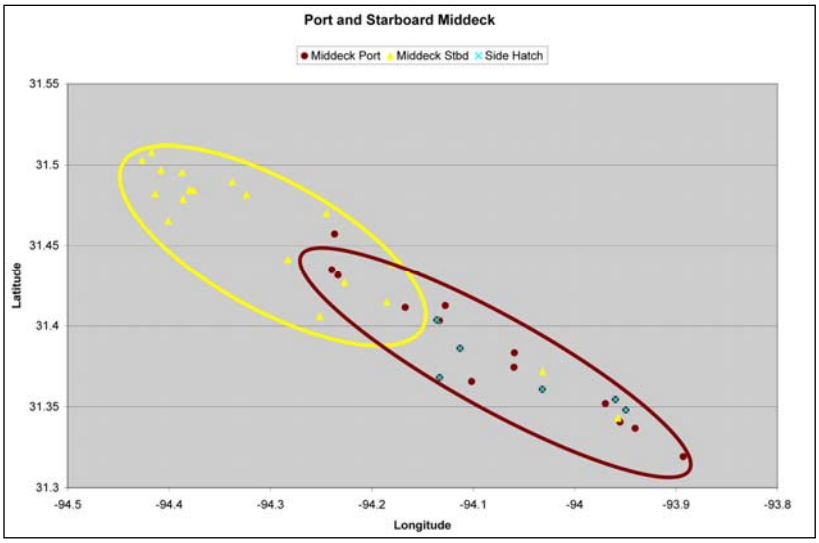


Figure 2.4-13. ***Middeck structural debris port vs. starboard*****.**

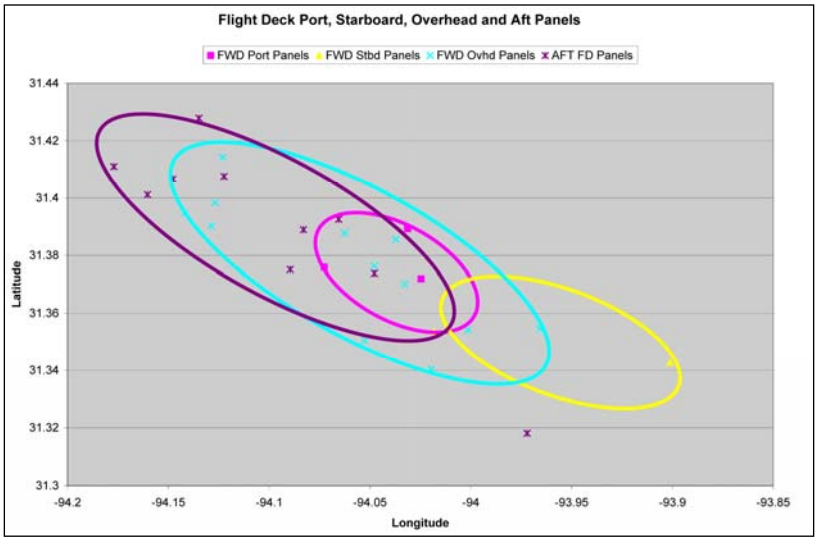


Figure 2.4-14. ***Flight deck panels by overhead, port, and starboard*****.**

There was a substantial amount of debris for the middeck analysis. Middeck starboard debris was recovered west of the middeck port debris, suggesting that the starboard side failed before the portside. Many of the items from the starboard side were lightweight items from the sleep station with low ballistic numbers, which may also affect this debris field. The debris field is consistent with a yaw to the portside, exposing the starboard side to greater thermal and aerodynamic loads. However, the flight deck panels appear inconclusive relative to a specific failure sequence of forward to aft or port vs. starboard. Not many panels were recovered from port or starboard, which makes interpretations risky.

Airlock

The airlock was located on the middeck. The aft opening of the airlock had no hatch and was open to the tunnel to the SPACEHAB in the payload bay. The forward hatch, leading to the CM, was closed and locked. Therefore, the airlock could lose pressure without impacting the internal conditions of the CM. See Section 2.2.4 regarding the failure of the TAA and subsequent depressurization of the airlock at or closely after the CE.

A few "loose" items, as well as a few items that were adhesively bonded to structure and a few pieces of secondary structure, were recovered in the western end of the debris field, indicating that these items were evacuated from the airlock when the TAA departed (figure 2.4-15). However, almost 80% of the airlock structure was recovered in the eastern portion of the forebody debris field (figure 2.4-16).

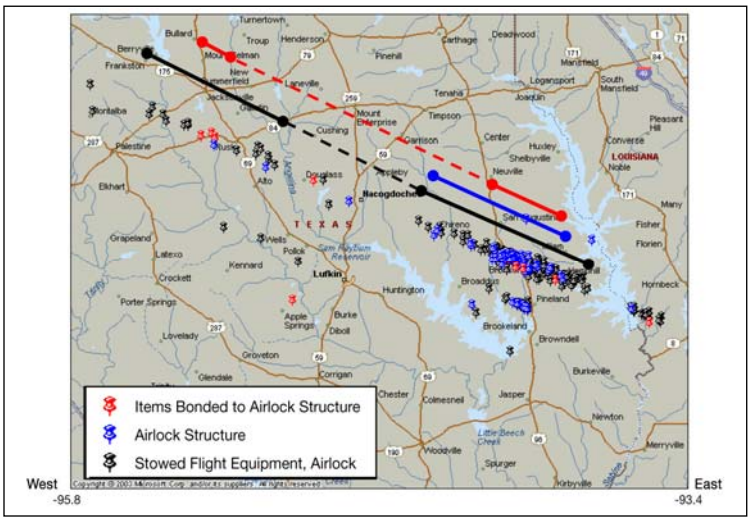


Figure 2.4-15. ***Debris field of the airlock structure, flight crew equipment stowed in the airlock, and items bonded to airlock structure.***

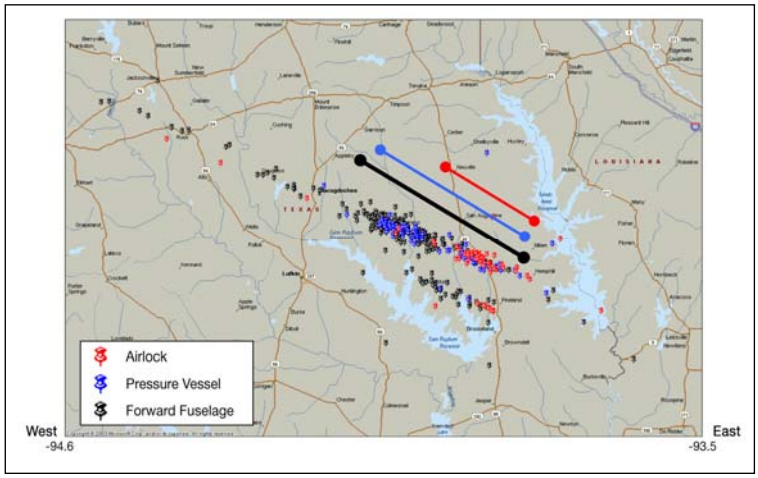
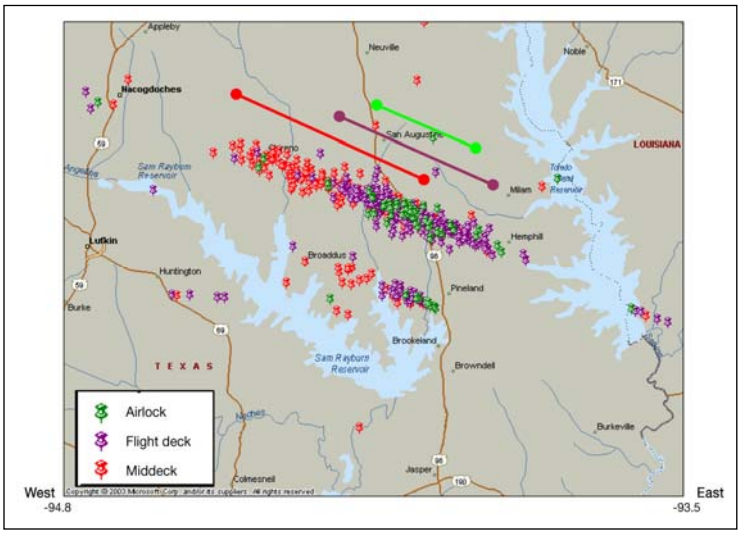


Figure 2.4-16. ***Debris field of the forward fuselage, pressure vessel, and airlock structure.***



The internal airlock is structurally attached to the CM aft of the X_o 576 bulkhead. It likely stayed with the aft bulkhead after the middeck departed. Figure 2.4-17 indicates that the airlock structural debris is also squarely in the middle of the flight deck structural debris. This suggests that the flight deck and airlock remained connected to each other by the aft bulkhead.

Figure 2.4-17. ***Debris field of the middeck, flight deck, and airlock.***

Forebody structural bulkheads

The forebody contained four major bulkheads. The bulkhead that was aft of the forward RCS compartment and immediately in front of the CM forward bulkhead was the X_o 378 bulkhead. The forward bulkhead of the CM was the X_{cm} 200 bulkhead.[2] The aft bulkhead of the CM was the X_o 576 bulkhead, and the bulkhead immediately aft of it was the X_o 582 ring frame bulkhead (figure 2.4-18).

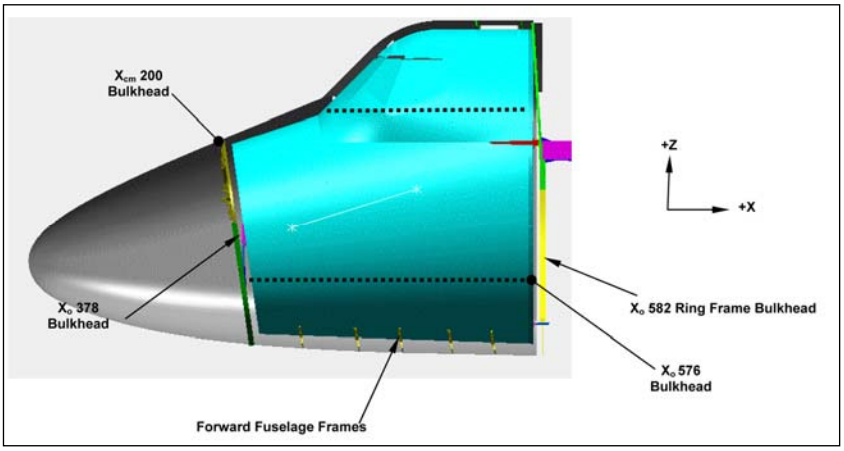


Figure 2.4-18. ***Forebody bulkheads*****.**

Figure 2.4-19 shows the bulkhead debris field clusters relative to each other. Figure 2.4-20 shows the CM (X_{cm} 200 and X_o 576) bulkhead debris fields relative to the middeck and flight deck debris fields.

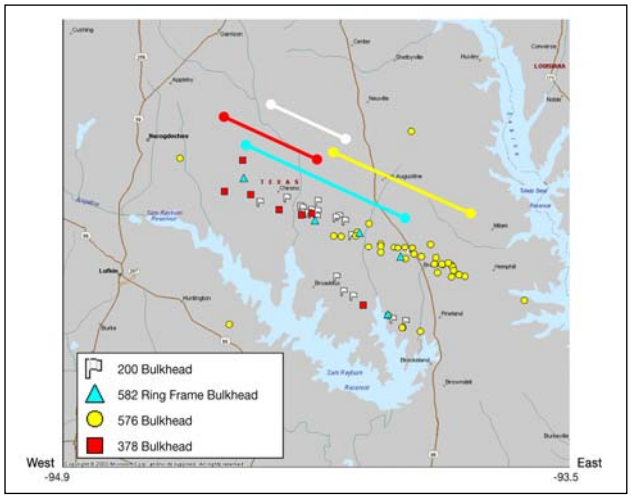


Figure 2.4-19. ***Debris field of the key forebody bulkheads*****.**

[2]The X_{cm} reference frame is the CM reference frame. The X_{cm} reference frame axes are aligned with the X_o reference frame, but the X-axis origin is different.

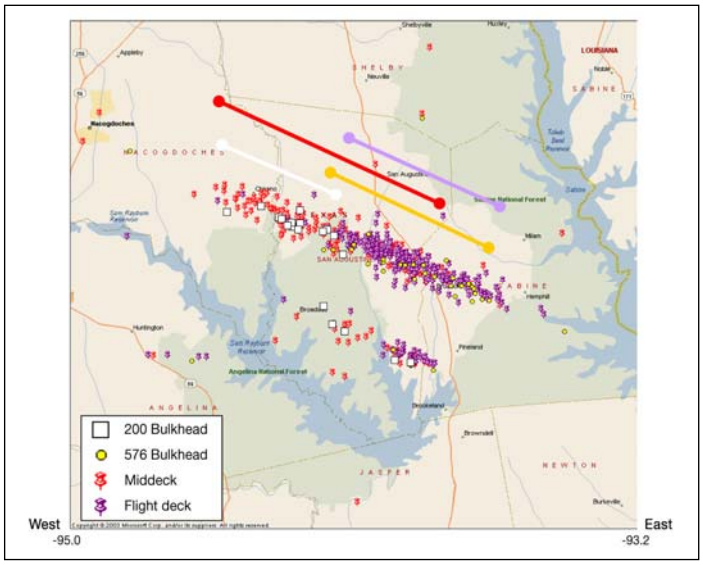


Figure 2.4-20. ***Debris field of the crew module bulkheads, middeck, and flight deck.***

The debris field clusters suggest that the two forward bulkheads appeared to fail earlier than the aft bulkheads. Also, the outer (fuselage) bulkheads failed prior to their corresponding inner (CM pressure vessel) bulkheads. The debris footprint for the CM forward bulkhead (X_{cm} 200) coincides with the center of the middeck debris field, suggesting that the X_{cm} 200 departed with the middeck. The CM aft bulkhead (X_o 576) appears to have remained with the flight deck until it disintegrated, likely also connected to the airlock since that debris field coincides as well.

This agrees with the general conclusion that the FF failed followed by the CM pressure vessel failure.

Avionics bays

The CM avionics bays debris plots were also evaluated to see whether they might provide insight into the CM failure sequence. These bays are in immediate proximity to the forward and aft bulkheads of the CM.

Debris plots represent debris from four avionics bays that are located on the middeck. Avionics Bays 1 and 2 are in the forward-most portion of the CM, immediately aft of the forward X_{cm} 200 bulkhead. Avionics Bays 3A and 3B are in the aft of the middeck on starboard and port sides of the airlock, respectively. These bays are immediately forward of the aft X_o 576 bulkhead. Although Bays 3A and 3B are smaller than Bays 1 and 2, a greater number of items were recovered from Bays 3A and 3B than from the forward bays. The recovered debris (mostly avionics boxes) were remarkably consistent in shape and subsequently estimated ballistic number, so the clusters were assumed to be adequate for a relative assessment (figure 2.4-21).

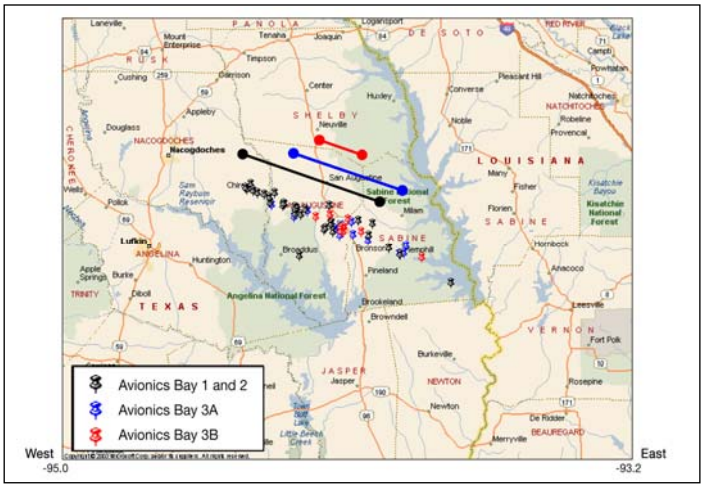


Figure 2.4-21. ***Debris field of the avionics bays.***

Debris from Bays 1 and 2 was recovered west of Bays 3A and 3B. This order matches the CM bulkhead order, supporting that the front (X_{cm} 200) bulkhead departed before the aft (X_o 576) bulkhead.

Sequencing based on cluster, video, and ballistic analysis
Figure 2.4-22 shows the relationships of some of the addressed forebody structures. The CM bulkheads are included in CM structure, and the X_o 378 bulkhead is included in the FF structure. The percentages listed in the figure refer to percentage of recovered debris, not percentage of the original intact area.

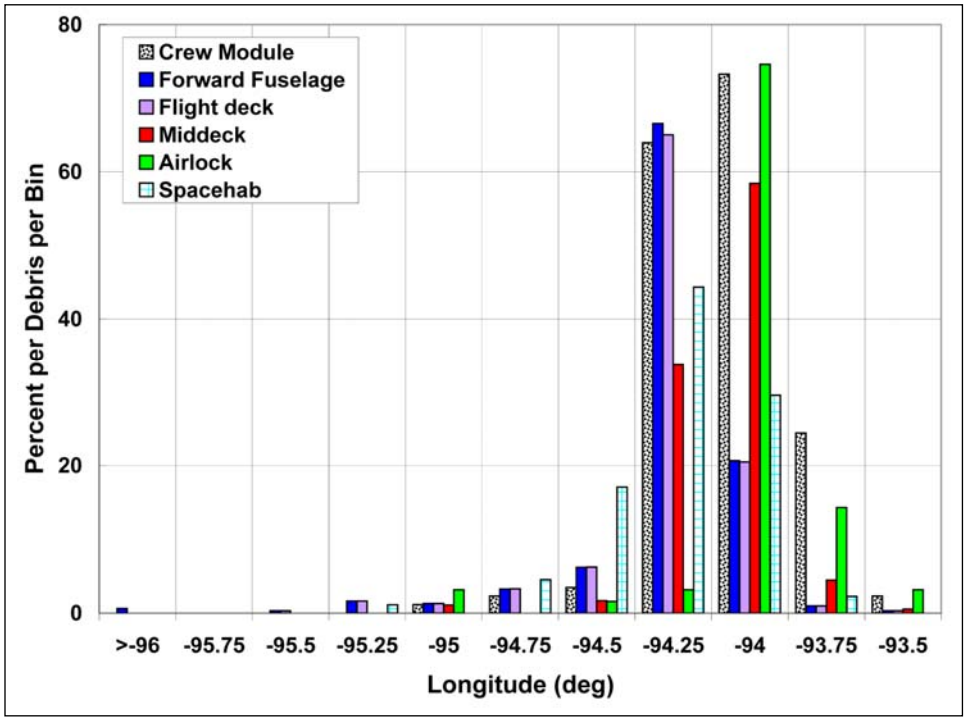


Figure 2.4-22. ***Histogram of the forebody structural components, plus SPACEHAB.***

To summarize the video and debris field findings, at the CE the CM was contained within the FF structure. The CM began to depressurize through a series of small breaches, with small amounts of FF debris being shed. Eventually, the FF structure failed, and the CM itself failed within a few seconds afterwards. The middeck and forward bulkhead (X_{cm} 200) of the CM departed, while the airlock, flight deck, and aft bulkhead (X_o 576) remained together for a short period longer until all elements separated and the TD occurred.

2.4.4 Structural analysis

This section discusses the breakup of the forebody based on structural analysis, using supporting evidence from the video, ballistic, and debris cluster analyses. See Section 2.2 for a more detailed discussion of the separation of the forebody from the orbiter. This analysis specifically discusses events that are related to the forebody at the CE and ending at the TD.

2.4.4.1 *Events at the Catastrophic Event*

General condition of forebody recovered debris
The recovered FF components are predominantly skin/stringer segments. Most components exhibited mechanical overload as the primary failure mechanism. Roughly 40% of the FF was recovered with no difference in damage levels comparing left to right or upper to lower. Two recovered RCC components, the nose cap and the chin panel (figure 2.4-23), show evidence of mechanical breakup with low thermal damage. It should be noted that the nose cap, based on field reports, apparently hit a tree before hitting the ground. The presence of intact Koropon primer on many FF components indicates that significant heating did not occur. Temperatures above 400°F (204°C) degrade the primer's appearance, and high temperatures

(>900°F) (486°C) will completely ablate it. The presence of Koropon indicates that the breakup of these elements was caused by mechanical overload rather than thermal effects.

Figure 2.4-23. ***Nose cap*** **Columbia** ***Reconstruction Database debris item no. 1114 vs. the original nose cap with chin panel*****.**

Some large pieces of the forward RCS were recovered (figure 2.4-24). These exhibit evidence of mechanical overload as the primary failure mechanism. Heating did not appear to play a significant role in the component degradation and appears to have occurred during or subsequent to the mechanical breakup.

Figure 2.4-24. ***Two large forward Reaction Control System debris pieces,*** **Columbia** ***Reconstruction Database debris item nos. 792*** **(left)** ***and 82061*** **(right)*****.***

Very few CM skin pieces were recovered. Most skin components were identified as portions of the thicker sections of the aft and forward bulkheads; the pieces all exhibit heavy thermal erosion. Several skin pieces that were identified as part of the CM center/bottom strip ("keel") were recovered. These pieces exhibit some mechanical breakup along with heat erosion within this thicker strip.

More than 65% of the middeck floor panels were recovered with paint and Koropon primer still intact, indicating that they were exposed to low thermal erosion.

Small numbers of the flight deck floor structural pieces were recovered; all recovered pieces exhibit heavy thermal erosion.

Crew module attach fittings (x-links, y-links, z-link)

All four main CM/FF attach links were recovered. In Section 2.2, it was concluded that the attach fittings, which are known as the x-links, y-links, and z-link, stayed with the forebody. These fittings attach the CM to the FF. The x-links also bridge to midbody structure. For a more detailed discussion of these attach fittings, refer to Section 2.2.4.

X-LINKS

Both the port and the starboard x-links (figure 2.4-25) were recovered nearly intact with evidence of high heating. The titanium fittings on both links experienced significant thermal exposure/melting, predominantly on the upper surfaces. Additionally, the starboard side fitting experienced significantly greater heating and erosion than the portside.

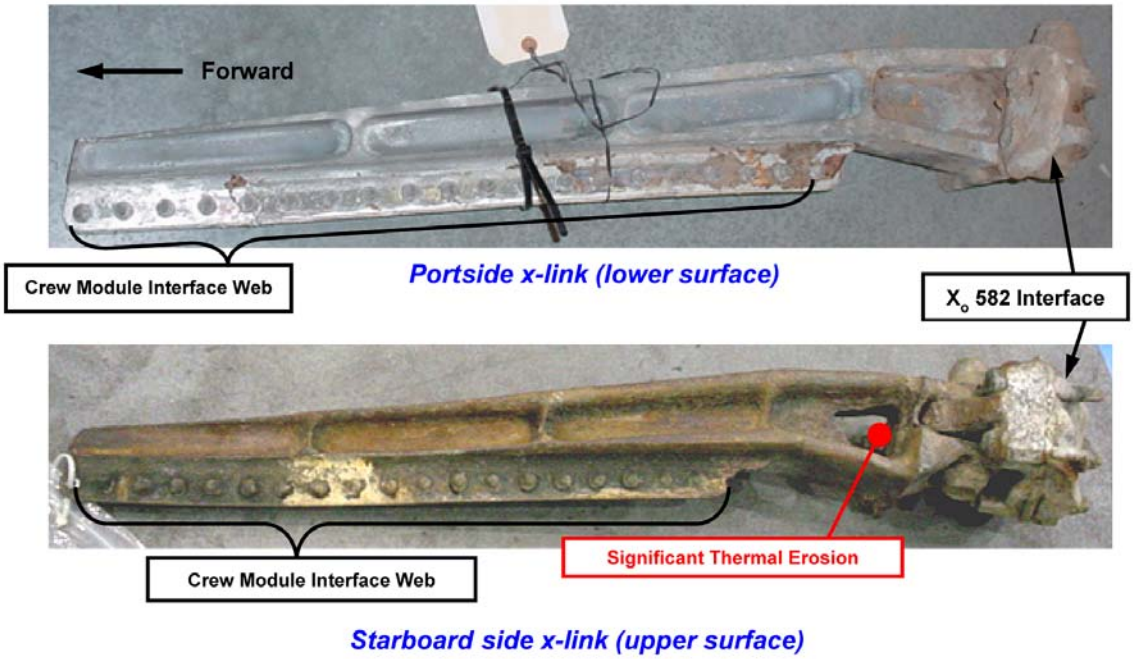


Figure 2.4-25. ***Crew module x-links,*** **Columbia** ***Reconstruction Database debris item nos. 1678*** **(top)** ***and 1765*** **(bottom).**

The titanium x-link components did not fail. The attach area on the CM side of the x-link is stronger than the FF side since it is reinforced by the flight deck floor and the X_o 576 bulkhead. The failures occurred at the weaker X_o 582 ring frame bulkhead connection points. The port x-link experienced a mechanical lug failure at the X_o 582 ring frame bulkhead interface, while the starboard side fittings pulled through the X_o 582 ring frame bulkhead at the CE. The starboard x-link retained some of the sill and X_o 582 ring frame bulkhead while the portside did not. Furthermore, it was concluded that at the CE, the forebody rotated left and pitched down, separating from the midbody (see Section 2.2.4).

Heat damage patterns on the webs of both x-links indicates that, at some point, there was hot gas flow from aft to forward above both x-links. The starboard x-link has more damage than the port x-link. Thermal analysis (see Section 2.1.6.7) of the x-links shows that entry heating alone was not capable of causing such heavy erosion. Additionally, because the surrounding structure was aluminum with a much lower melting temperature, entry heating alone would have resulted in the weakening of the surrounding structure first and release of the x-links. Section 2.1.7 discusses the other thermal mechanisms that were likely involved, shock-shock interactions and combustion. Both of these mechanisms are possible with the forebody geometry as understood from the debris.

Debris item no. 2436 is a piece of FF skin panel outside of the starboard x-link (figure 2.4-26). It was subjected to heat erosion mainly at the aft edge (along the X_o 582 ring frame bulkhead) (figure 2.4-27). The pattern of deposited molten aluminum on the inboard side indicates that it was subjected to thermal flow aft to forward over the x-link (see arrows, figure 2.4-27).

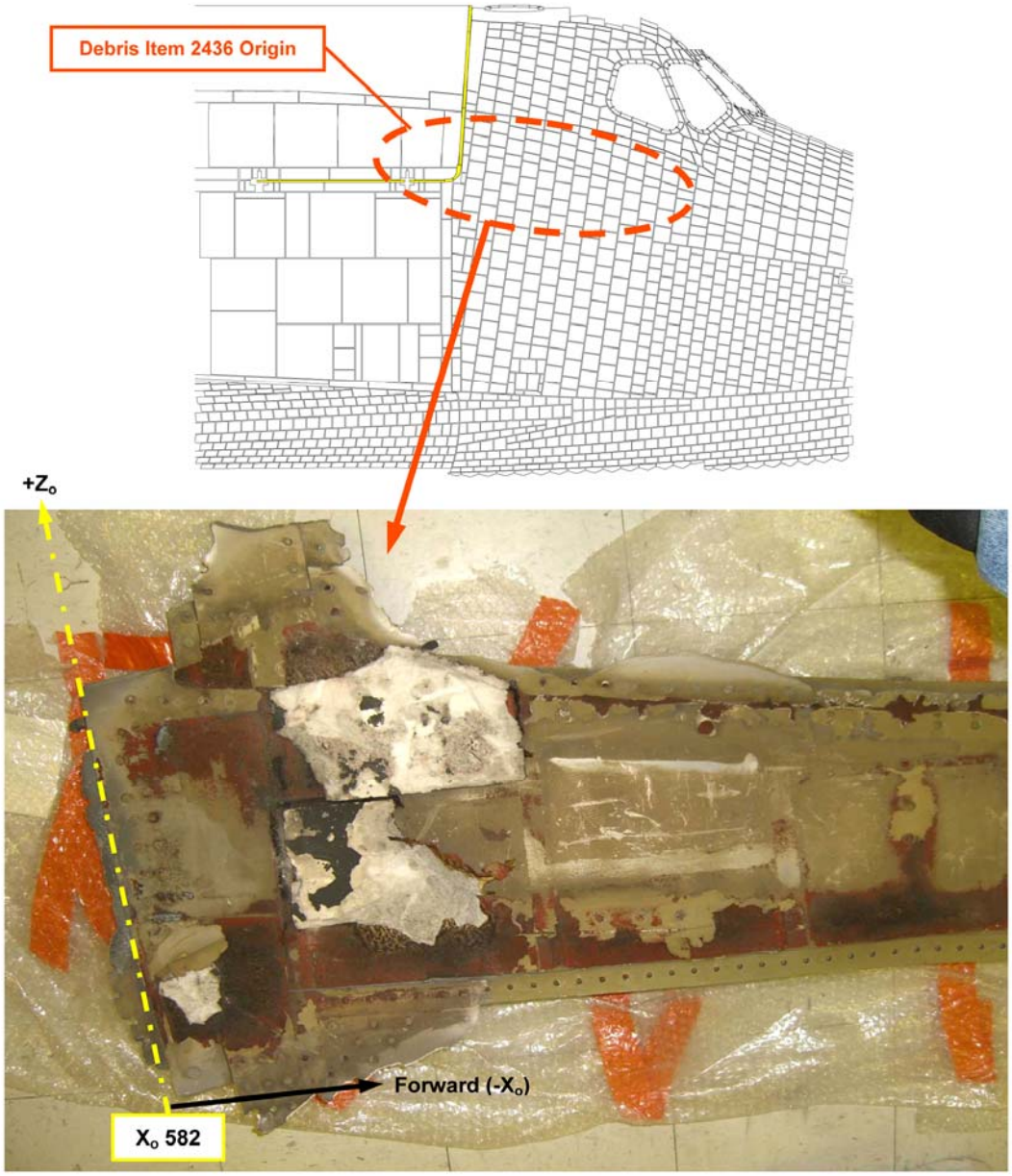


Figure 2.4-26. ***Starboard-side skin, view looking inboard,*** **Columbia** ***Reconstruction Database debris item no. 2436.***

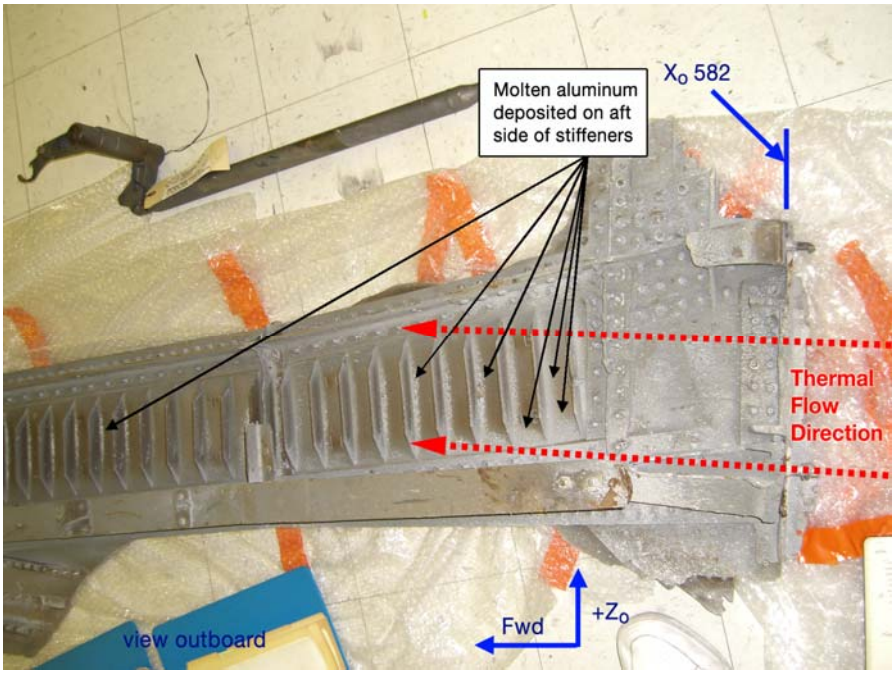


Figure 2.4-27. ***Inboard side of the starboard-side skin,*** **Columbia** ***Reconstruction Database debris item no. 2436.***

This debris shows high heat erosion at the aft end while the remaining edges failed mechanically. The pattern of molten aluminum deposition indicates that the hot gas flowed mainly from aft to forward. If the thermal event happened after this panel broke away from the FF shell, all edges should experience a similar level of thermal damage, and the deposition of molten aluminum on the inside surface of the skin should have a random pattern instead of a directional pattern (figure 2.4-27). This suggests that this piece of FF skin was in place when the thermal event occurred. It is not known when this thermal event occurred. Since the forebody was rotating, the aft portion of the forebody may have periodically been presented to thermal flow for brief periods between the CE and the CMCE. The thermal event may also have happened around the CMCE, when the FF pulled away and exposed the area to thermal flow.

Y-LINKS

The y-links attach the aft bulkhead of the CM (X_o 576) to the X_o 582 ring frame bulkhead at the lower central portion of the bulkhead. It is possible that the two y-links remained intact at the CE, since most of the X_o 582 ring frame bulkhead is believed to have stayed with the FF. After the CE, any Y-direction movement of the CM relative to the X_o 582 ring frame bulkhead would exert high tension or compression loads on these links. Failure by tension is evident on the recovered debris of these links. The portside y-link shows that it also was softened by thermal exposure along with the tension failure. This portion protrudes beyond the X_o 582 ring frame bulkhead, which would have been exposed periodically to hot gas flow following the CE as the forebody rotated (figures 2.4-28 and 2.4-29).

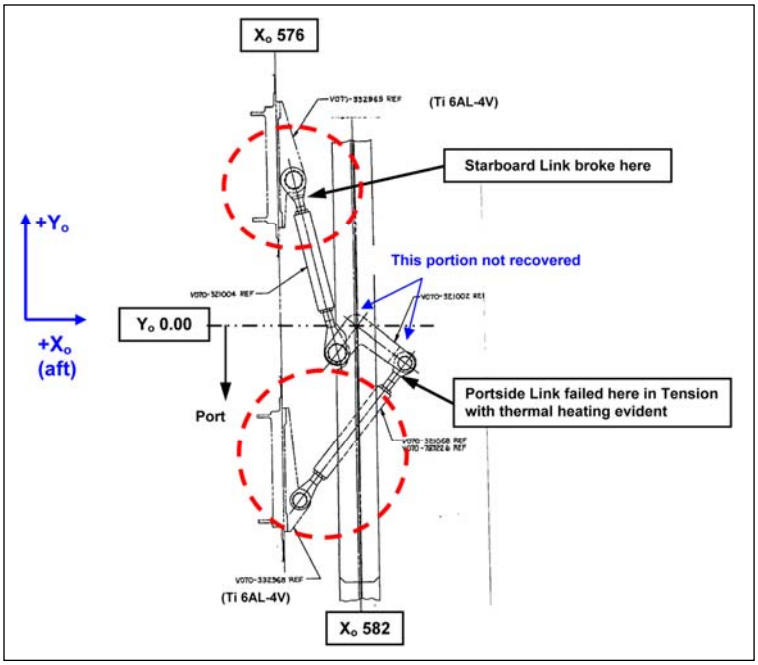


Figure 2.4-28. ***View looking from the top at the y-links*.**

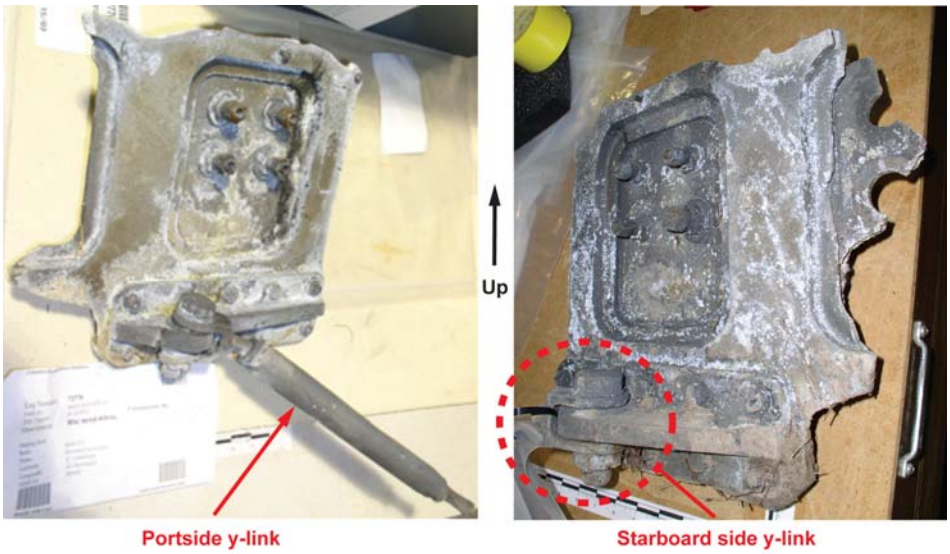


Figure 2.4-29. ***The y-link debris, still attached to the mounting on the X_o 576 bulkhead,*** **Columbia** ***Reconstruction Database debris item nos. 72770*** **(left)** ***and 9280*** **(right).**

Z-LINK

The z-link failed at the attach point to the X_{cm} 200 bulkhead (figure 2.4-30). The joint failed by a combination of fastener tensile failure and fastener insert pullout. Insert pullout requires thermal weakening of parent material because, by design, the insert is stronger than the fastener (under normal temperatures, the fastener should fail before the insert pulls out). Heating may have been occurring as the forebody rotated, allowing hot gas into the plenum between the CM and the FF. Failure resulted at the CMCE.

Alternately, after the FF separated, the X_o 378 bulkhead may have remained in place attached to the X_{cm} 200 bulkhead, connected by the z-link, and exposed to entry heating flow. The two bulkhead debris fields support this conjecture because the X_o 378 bulkhead and the X_{cm} 200 bulkhead debris fields overlap, with the X_o 378 debris field being slightly west of the X_{cm} 200 debris field. No ballistic analysis was done on either bulkhead elements to support or refute this conjecture.

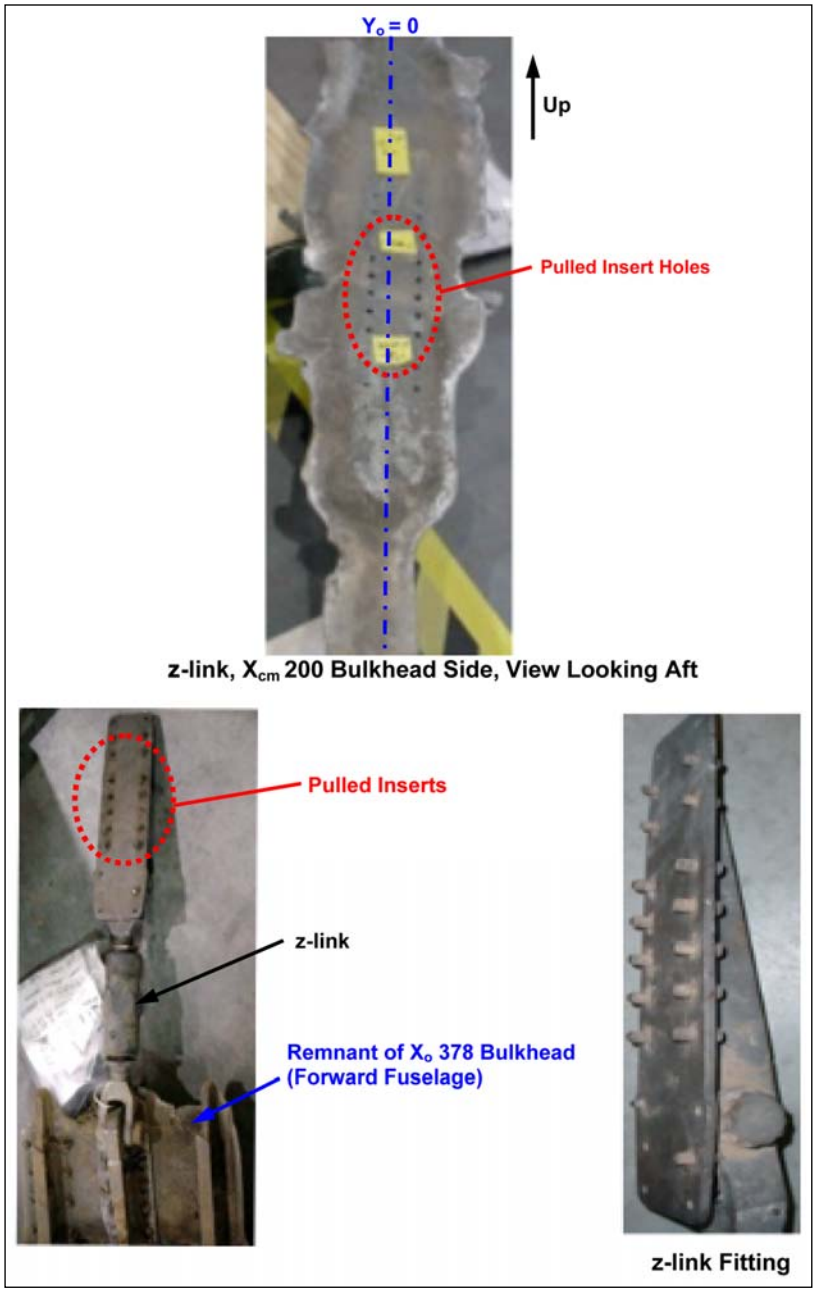
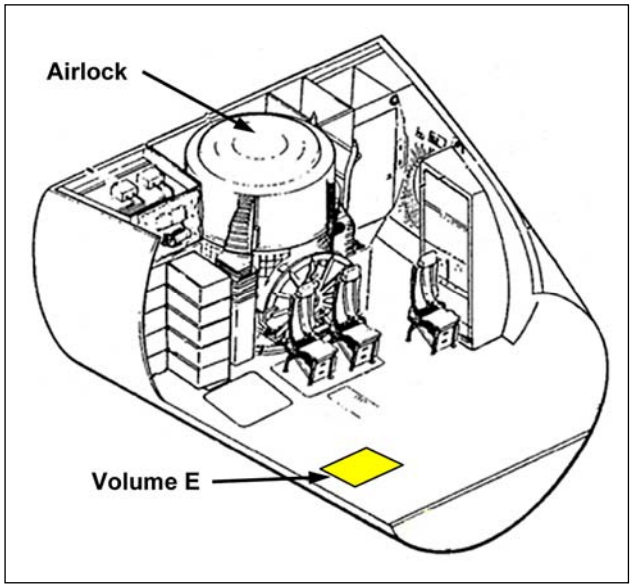


Figure 2.4-30. ***The z-link debris photographs,*** **Columbia** ***Reconstruction Database debris item no. 53828*** **(top).**

Forward fuselage and crew module interaction at the catastrophic event

The cabin depressurization analysis (see Section 2.3) led to the conclusion that the cabin began to depressurize NET the CE (GMT 14:00:18) and NLT GMT 14:00:35; and probably depressurized through several small breaches. There is limited debris evidence to pinpoint locations for these breaches.

Items that originated from inside the CM and were recovered in the western portion of the debris field were reviewed to determine a potential breach area. These items included multiple STS-107 mission patches. Most of the mission patches were stowed in a middeck sub-floor stowage volume called Volume E. In-flight access to Volume E was not possible in the *Columbia* cabin configuration, so the crew could not have accessed the patches and stowed them elsewhere. Review of the stowage configuration documents for this compartment revealed that the patches were stowed near the bottom of Volume E. Therefore, the bottom portion of the Volume E structure had to be compromised to release the patches.



Volume E is a container with an opening on the top. The top part of this volume is a machined aluminum upper frame with a door, a hinge, and latches to cover the opening. Four sides and bottom panels are made of aluminum honeycomb core and aluminum facesheets. The upper edges of the side panels are attached to the upper frame vertical flanges by rivets. As illustrated by figure 2.4-31, the door of Volume E is at the level of the middeck floor. The upper frame of the box is attached to the floor beams by 18 Milson bolts and receptacles. The bottom of Volume E is tapered to match the curvature of the lower portion of the CM pressure shell.

Figure 2.4-31. ***View of middeck floor and Volume E, looking down and aft.***

Detailed inspection of Volume E debris indicates that it was subjected to mechanical damage prior to thermal damage. The outer perimeter "picture frame" of the bottom panel was recovered and shows evidence of impact from below. Impact from the CM skin below Volume E would push the aft/outboard corner of the volume upward and then split the edge of the upper frame, as seen in the debris (figures 2.4-32 and 2.4-33).

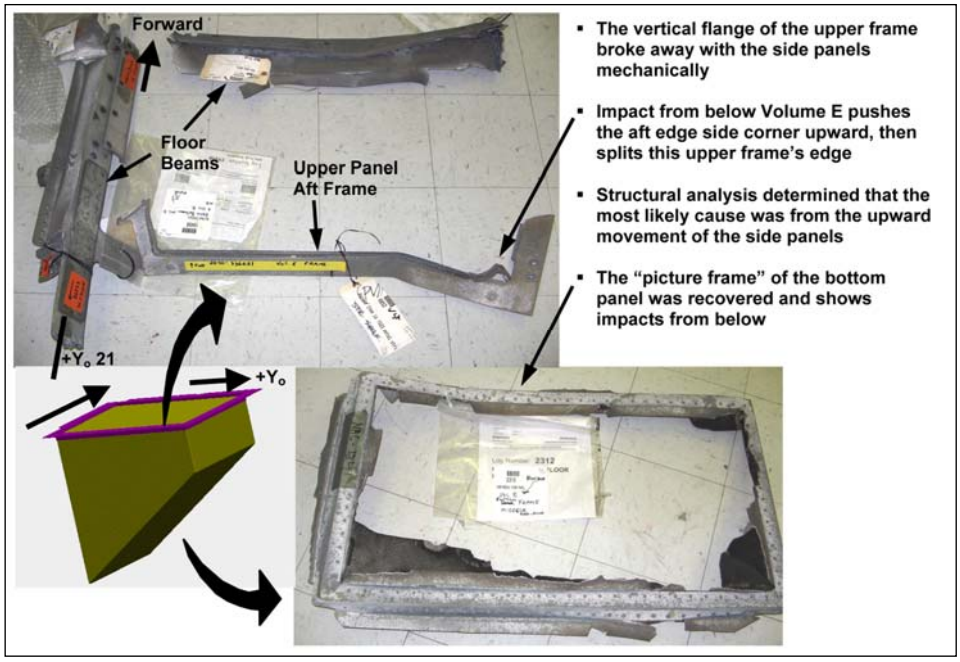


Figure 2.4-32. ***Volume E debris damage.***

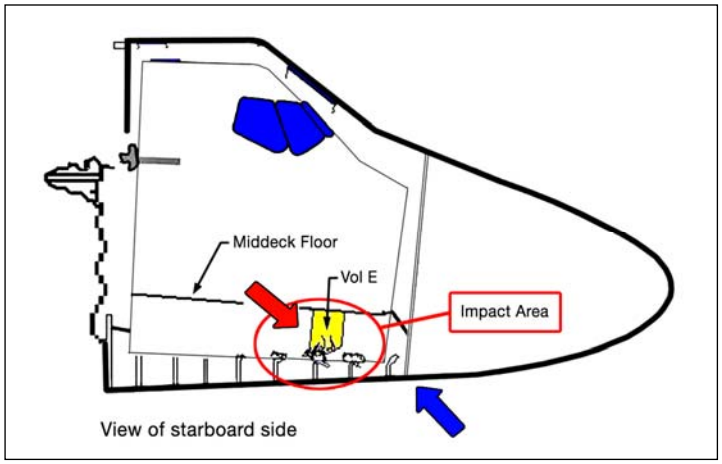


Figure 2.4-33. ***Scenario showing how the crew module pressure vessel could impact the forward fuselage, and the middeck Volume E could impact the crew module pressure vessel, with resultant damage.***

Sudden changes to the rotation of the forebody complex probably occurred at the CE because of the asymmetric release from the mid-fuselage. Due to c.g. locations of the FF and CM, centrifugal forces would tend to pull one away from the other as the forebody rotated. The large portion of the X_o 582 ring frame bulkhead was still attached to the FF. Thus, it and the other remaining attached linkages may have prevented the CM extraction from the FF.

It is likely that at or shortly after the CE, the FF structure impacted the CM skin below Volume E and, in turn, caused an impact between the CM skin and Volume E. The impact was severe enough to crack open the Volume E box, possibly near the lower portion, spilling some of its contents, including the patches. The patches then escaped the CM during CM depressurization.

The impacts under Volume E could possibly have breached the CM skin, providing a local vent path for the stowage volume contents such as the crew patch.

More impacts likely occurred as the forebody continued to rotate. From the CE to the CMCE, it is likely that the CM swung back and forth inside the FF, with both the CM and the FF experiencing impacts at multiple locations. The FF structures would have received damage, possibly creating paths for thermal inflow, as the forebody rotated. Impacts from the relative motion between the CM and the FF would damage the forward bulkheads (X_o 378 and X_{cm} 200), the fuselage frames, and the thinner areas of CM aluminum skin, star tracker well area, etc., depending on the attitude and the rotational movement. Recovered crew equipment debris also came from the forward lockers on the portside, not far from the star tracker wells, lending credence to this. Frame and bulkhead damage would destabilize the FF structures, perhaps popping open numerous items, including the star tracker panels, the TPS around the forward windows, the forward RCS module, the nose gear doors, and the side hatch outer layer. Review of forebody items found in the debris field prior to the main debris field support this conclusion, as portions of the nose gear, star tracker, and multiple tiles and portions of TPS made up this early released debris (figure 2.4-34).

Between the CE and the CMCE, it is not clear when the remaining links (z-link, four side links, and four side hatch links) failed. Obviously, sufficient integrity was maintained to hold the CM inside the FF.

Mechanical and thermal degradation of the crew module pressure vessel between the Catastrophic Event and the Crew Module Catastrophic Event

The airlock depressurized rapidly following the CE with the departure of the TAA (see Section 2.2.4). It might be speculated that rapid depressurization of the airlock with the interior of the CM still pressurized might implode the airlock structure. However, the airlock is intended for depressurization during spacewalk activities and its interfaces are reinforced. Empirical review of airlock structural capacity shows that implosion of the airlock is an unlikely scenario.

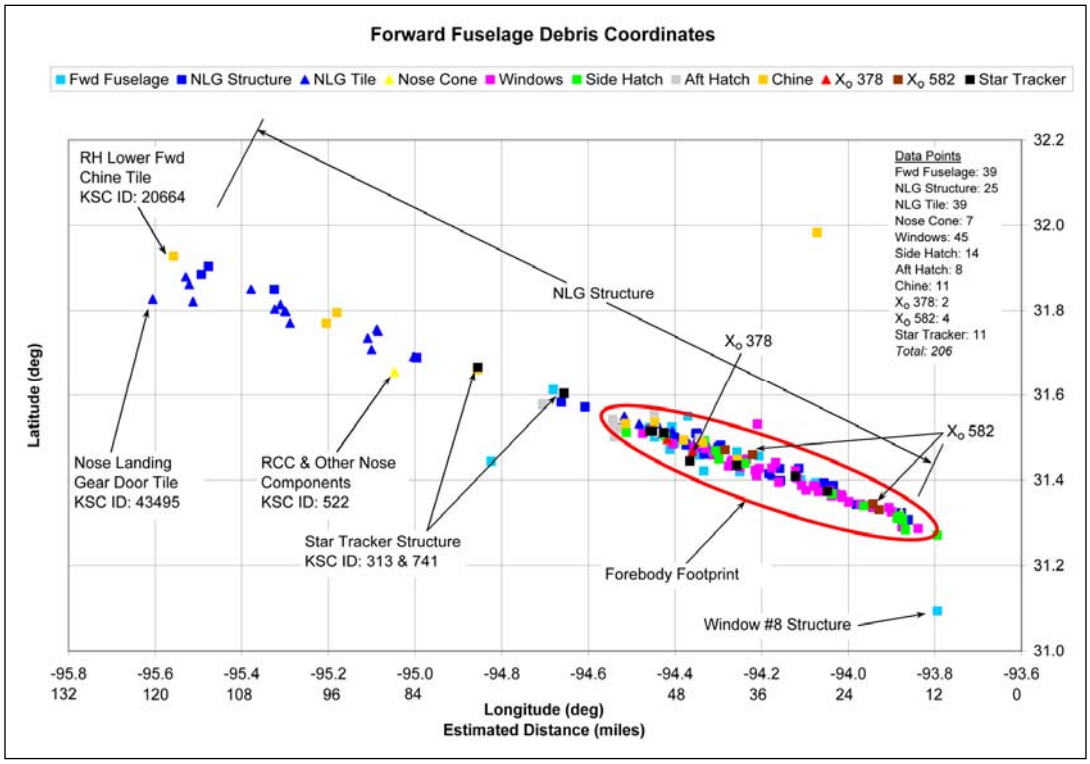


Figure 2.4-34. ***Forward fuselage debris field showing tile, nose landing gear door structure, and star tracker door structure west of the main forebody debris field.***

Other warping as a result of the stresses of the CE cannot be ruled out. However, impacts as a result of the CM moving inside the FF appear to be the best candidate for mechanical breaches because they are the best supported by the debris field.

Thermal exposure of the CM pressure vessel may have weakened certain areas resulting in mechanical failure. This would not have happened immediately at the CE but would have taken time to develop as the forebody rotated, periodically exposing unprotected segments of the forebody.

The CM aft bulkhead was a clear immediate candidate for thermal breach since, unlike the rest of the CM, it was not protected by the FF and accompanying TPS. However, the bulkhead was extremely heavy, because it was made of aluminum waffle with additional reinforcing beams, and would act as a good heat sink. Debris field evidence does not support significant thermal erosion of the CM aft bulkhead prior to CM breakup.

There are gaps between the X_o 576 bulkhead and the X_o 582 ring frame bulkhead and directly over the x-links that would have allowed hot gas entry into the plenum between the forebody and the CM when rotation resulted in the aft forebody being presented to the directional thermal flow. Mechanical breaches resulting from the impacts of the CM and the FF could also have allowed thermal flow into the plenum if the breach were presented to the velocity vector.

The CM pressure vessel skin was designed with a main cone shape that has low thickness and minimized integral stringers to save weight (figures 2.4-35 and 2.4-36). Many large areas of the cone shape have uniform low thickness—as thin as 0.039 in. Skin thickness was controlled tightly by a chemical milling process during manufacturing. Because of low and uniform thickness, a large area of skin can be heated up quickly and uniformly. This can result in rapid thermal failure of the skin panel, and heat erosion can propagate rapidly.

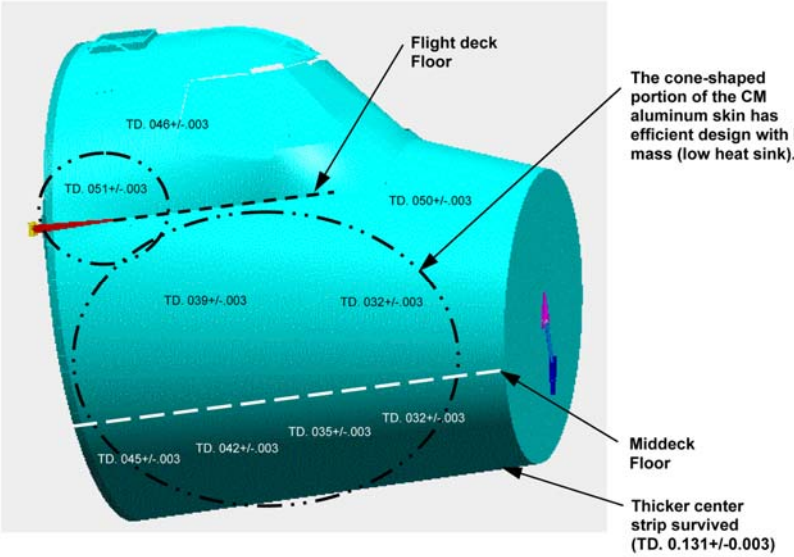


Figure 2.4-35. ***Crew module pressure vessel skin lightweight design*****. The thickness dimension (TD) is represented in inches.**

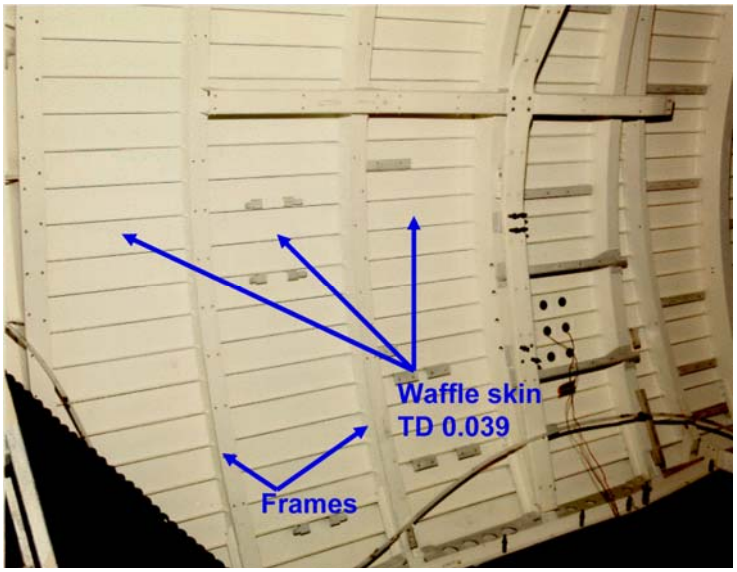


Figure 2.4-36. ***Typical crew module pressure vessel skin in middeck, looking outboard*****.**

Very little CM skin structure survived. It is very thin, and probably melted quickly after breakup when exposed to entry heating. Only a relatively thicker region of skin strip (0.131-in. thick) along the bottom at the centerline was recovered in multiple pieces (figure 2.4-37).

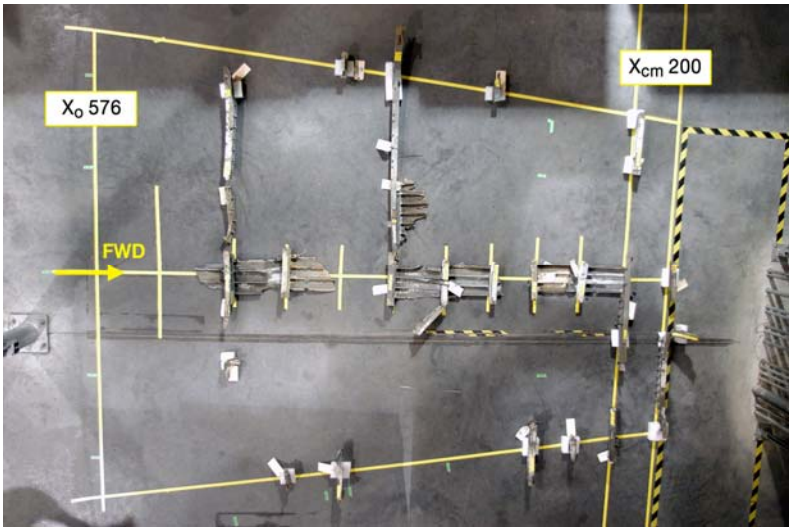


Figure 2.4-37. ***Recovered crew module pressure vessel base skin*.**

It is possible that areas of the CM skin were weakened or softened uniformly by thermal flow into the plenum between the CE and the CMCE. However, many recovered FF skin areas exhibited intact primer on the inner surfaces. This implies that most of the forebody TPS still remained to provide thermal protection on the outboard side of the FF panels before the FF breakup, and that the loss of CM pressure vessel skin was a result of exposure during the breakup after the CMCE.

2.4.4.2 *Crew Module Catastrophic Event/forebody breakup*

Forward fuselage failure

It is unclear where the initial failure of the FF occurred. Loads due to deceleration of the forebody were increasing. Although much of the forebody was protected by the TPS of the FF, periodic exposure to heating was increasing as well.

The "arrowhead" skin/TPS panel between Windows 3 and 4 (debris item no. 65049) (figures 2.4-38 and 2.4-39) on the front of the forebody was recovered near the middle of the forebody debris field. It appears to have been torn on the portside and pulled through on the starboard side by mechanical loading, probably initiated by a relative movement between the port Windows 1 through 3 and the starboard Windows 4 through 6.

Figure 2.4-38. ***Forward fuselage arrowhead panel and the forward two windows, facing in*.**

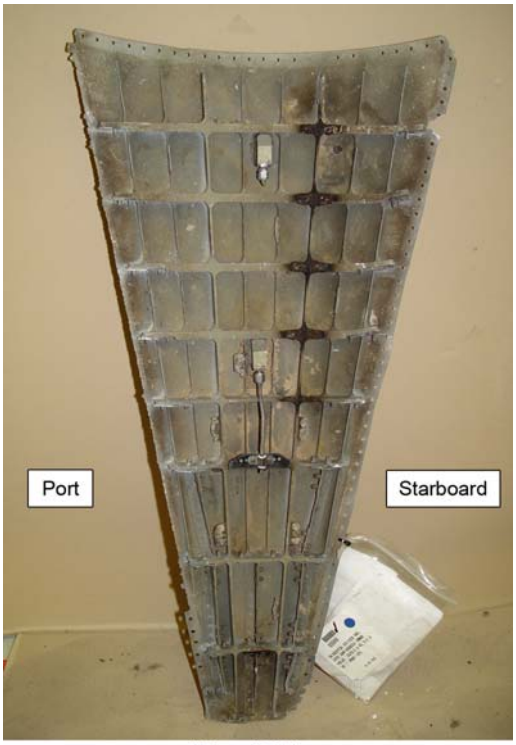


Figure 2.4-39. ***Low-heat damage on the forward fuselage arrowhead panel, facing out,*** **Columbia** ***Reconstruction Database debris item no. 65049.***

The departure of the FF Windows 1 through 6 and the FF arrowhead panel would likely trigger the departure of the rest of the FF canopy by aerodynamic forces. This may have been the initiating event of the CMCE, or it might have been simply one event during the breakup. Without the upper FF half, the lower half would immediately peel away. This is supported by the video image, which shows what appear to be two large and symmetrically sized objects separating from the forebody at the CMCE.

Figures 2.4-40 and 2.4-41 are of the FF skin debris items that lie below the CM. Many panels have little molten metal deposition on the inside surfaces. Broken edges show little thermal damage, and some still have intact primer. This indicates that the plenum behind these areas was not exposed to long durations of thermal flow. Some skin pieces even had stringers that were removed mechanically. Most of the recovered skin debris appears to have mechanically fractured edges. Many forward RCS structural items were recovered; it appears that the forward RCS was pulled away mechanically and broke up into pieces with low thermal effects. The conclusion was that it is likely that the FF skin panel peeled away and broke up quickly.

Figure 2.4-40. ***Lower forward fuselage skin and frame debris*****.**

Figure 2.4-41. ***Lower forward fuselage skin panel near external tank attachment.***

Crew module breakup

Unlike the FF, very little CM pressure vessel skin was recovered. The lack of thermal effects and interior deposition on the FF debris indicates that it was no longer with the CM when the pressure vessel skin melted. Once the FF separated, the CM would become highly susceptible to thermal heating, especially in the areas of low skin thickness. Multiple additional thermal breaches probably appeared on the CM skin within a few seconds because of the low heat sink and uniform thickness of the CM pressure vessel skin along the CM sides and bottom.

Failure modes that were assessed on the CM pressure vessel and secondary structural components suggest that fractures occurred subsequent to elevated temperature exposure (corresponding to a significant reduction of material properties). This clearly suggests that following FF separation, breakup of the CM structure occurred as a consequence of combined aerothermal heating and aerodynamic loading.

Middeck and forward bulkhead

Figure 2.4-42 shows recovered CM forward bulkhead (X_{cm} 200) debris on a grid that is the size of the original bulkhead. As described in the debris field cluster analysis, the CM forward bulkhead debris was recovered west of the aft bulkhead debris, and the middeck starboard debris was recovered west of the port debris. This suggests that the CM middeck starboard/forward was shed first and that the starboard and forward CM areas were exposed to higher heating first and/or faced the velocity vector at the time of the CM major breakup. The main CM breakup involved the departure of the middeck area, including the middeck floor and everything that was attached to that floor. Anything not firmly attached to the flight deck floor also came out with the middeck. Most equipment that was above and below the middeck floor was supported by the middeck floor; avionics bays are also supported by the side skins and bulkheads.

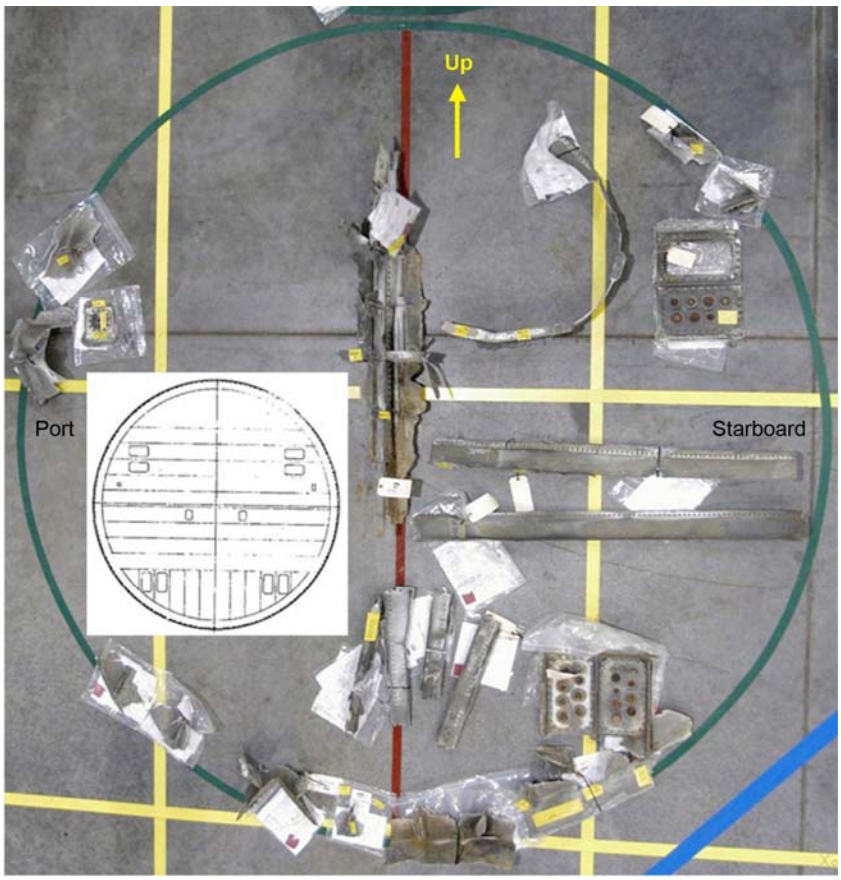


Figure 2.4-42. ***View looking forward of the recovered crew module X_{cm} 200 bulkhead debris.***

Structurally, the flight deck section is connected to the middeck section and the lower equipment bay is mainly connected by the CM side skin, avionics bay partitions, and the aft and forward bulkheads. The side skin and the forward bulkhead could be easily damaged by aerodynamic heating after the CM lost the protection from the FF shell. Therefore, as soon as the middeck side skin and forward bulkhead were compromised structurally, the lower half of the CM could have swung away from the flight deck portion by the effects of aerodynamic drag. It is likely that the whole middeck floor assembly came out together with the lower equipment bay. The hinge line appears to be at or near where the middeck floor attaches to the aft bulkhead. This motion would expand the middeck compartment (similar to opening a clam shell) and release all items attached to the middeck floor as well as other items stowed beneath that floor. Because the middeck floor disintegrated into many smaller parts without being significantly heated, it appears that the middeck floor failed as a result of structural loading. This may help to explain why more than 65% of the middeck floor was recovered without significant thermal erosion (figures 2.4-43 and 2.4-44).

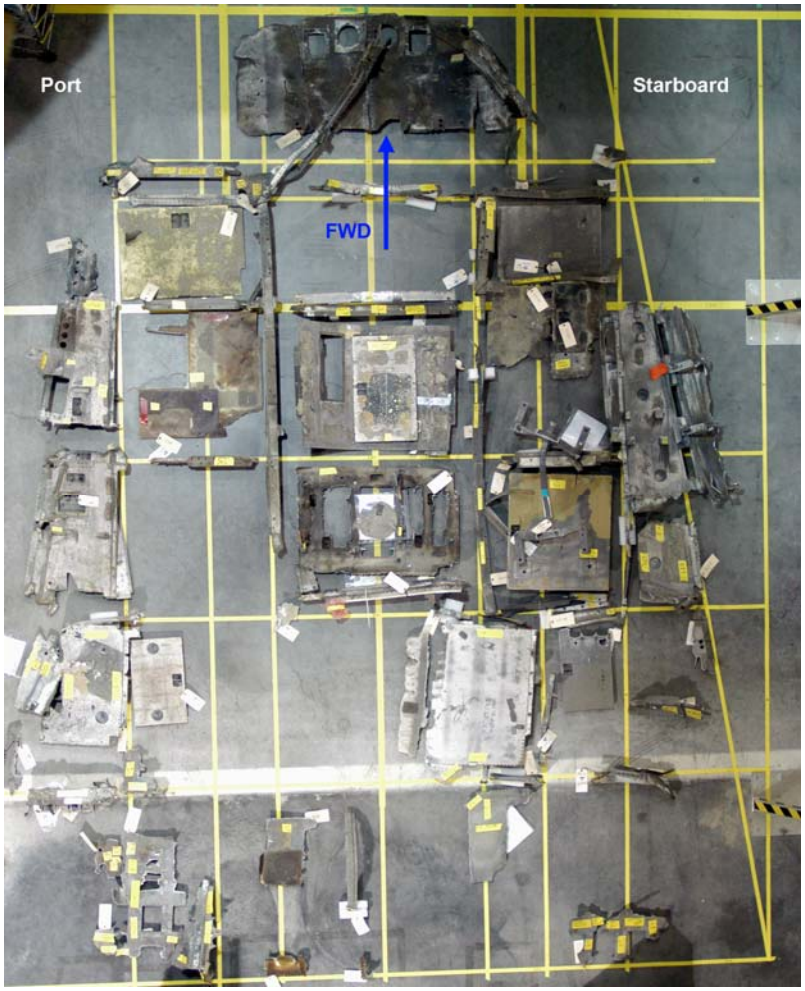


Figure 2.4-43. ***Top view of the crew module middeck floor debris*****.**

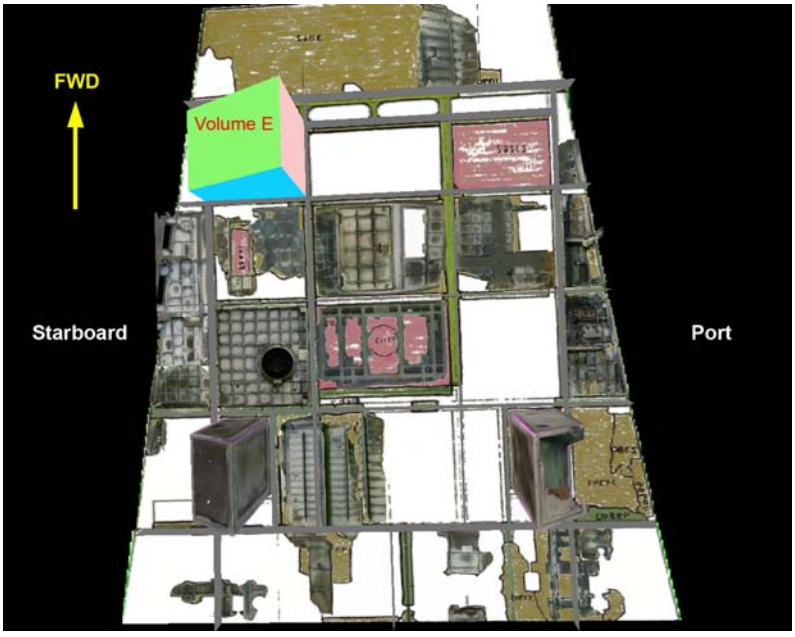


Figure 2.4-44. ***Bottom view of the recovered middeck floor items from virtual reconstruction*****.**

The middeck floor and the lower equipment bay quickly disintegrated, with middeck floor panels, crew escape pole, MAR, and other crew equipment items (window shade bag, middeck lockers, sub-floor components, MADS/OEX recorder, etc.) departing quickly from the CM.

Flight deck and aft X_o 576 bulkhead

The CM debris field suggests that after the forward bulkhead and the middeck floor departed, it was followed by the airlock, the flight deck, and the aft bulkhead. The flight deck floor debris exhibited much more thermal damage than the middeck floor debris.

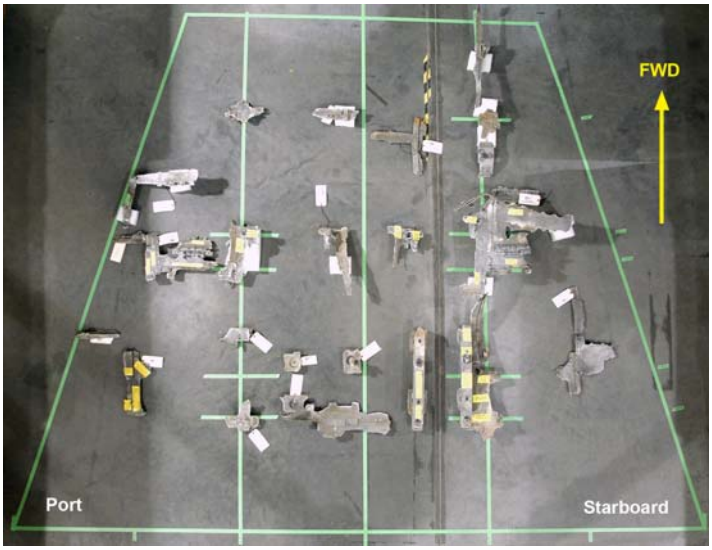


Figure 2.4-45. ***Top view of the crew module flight deck debris.***

From the thermally eroded state of the flight deck debris, it appears that the flight deck stayed nearly intact for a period of time following the departure of the middeck area. This explains why so little of the flight deck floor was recovered (figure 2.4-45). Based on debris field evidence, the internal airlock, which was supported by the MAP (which is part of the X_o 576 bulkhead), likely stayed together with the bulkhead and the flight deck.

Significant mechanical damage was noted on the starboard aft panels of the flight deck. These panels were recovered in much smaller segments than other flight deck panels. This suggests that the starboard side of the flight deck near the aft bulkhead (near the starboard x-link) experienced a more dynamic failure than the portside. This may have resulted from structural degradation that occurred when the forebody separated from the midbody at the CE in this location. It is also consistent with a starboard-to-port failure for the middeck.

In general, heavy portions of the X_o 576 bulkhead (including the MAP) survived the entry heating (figure 2.4-46). This is possibly due to the high heat absorption property of the aluminum bulkhead.

Some "T"-section stiffeners survived with little heat damage, possibly because of the early departure of these elements before the high thermal event that consumed the bulkhead.

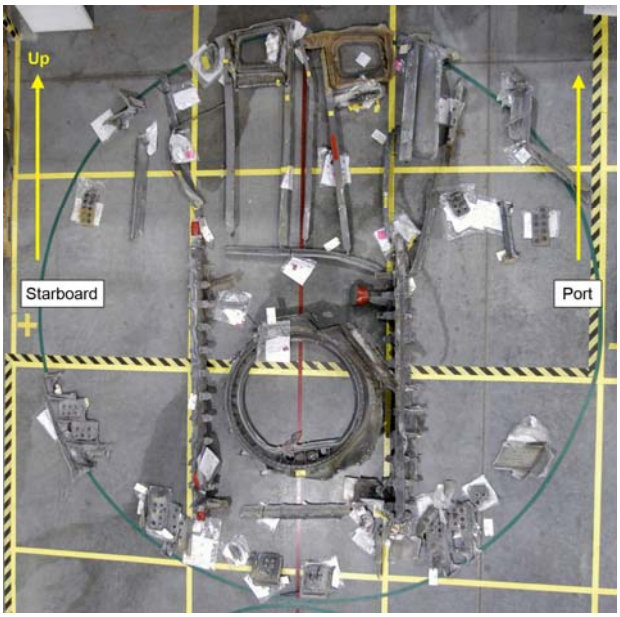


Figure 2.4-46. ***View looking aft of the recovered X_o 576 bulkhead debris.***

Crew module crew equipment

The CM contains many items that were installed to facilitate the space shuttle crew's on-orbit operations. This equipment is generally termed "crew equipment" and includes the crew seats, middeck stowage lockers, the MAR, sleep stations, the galley, the WCS, the ergometer, the Crew Escape System (CES) pole, crew worn equipment, and loose equipment that was stowed in various locations. Attention was focused on the MAR and the CES pole because they have substantial attachments to the CM structure. Generally, the debris from the sleep stations, galley, WCS, ergometer, lockers, and loose equipment was highly fragmented and did not provide significant insight into the CM breakup. Analysis on those items was limited to identification and ballistics analysis on a few select items. Because the SPACEHAB module contained loose equipment that was stowed in middeck lockers, the SPACEHAB and the CM debris footprints both contained loose equipment and locker structure debris. Therefore, loose equipment and locker structure items were excluded from analysis on the SPACEHAB and CM debris footprints.

Recovery locations of suit and seat components indicate that the middeck crew members separated from the CM before the flight deck crew members. Additionally, flight deck seats experienced higher heating than middeck seats. These findings support the debris field cluster analysis conclusion that the middeck broke up before the flight deck broke up (see Sections 3.1 and 3.2).

Middeck accommodations rack

The MAR is pinned to the port wall and the middeck floor forward of the side hatch and aft of the galley (figure 2.4-47) on the middeck. The MAR spans from middeck floor to ceiling.

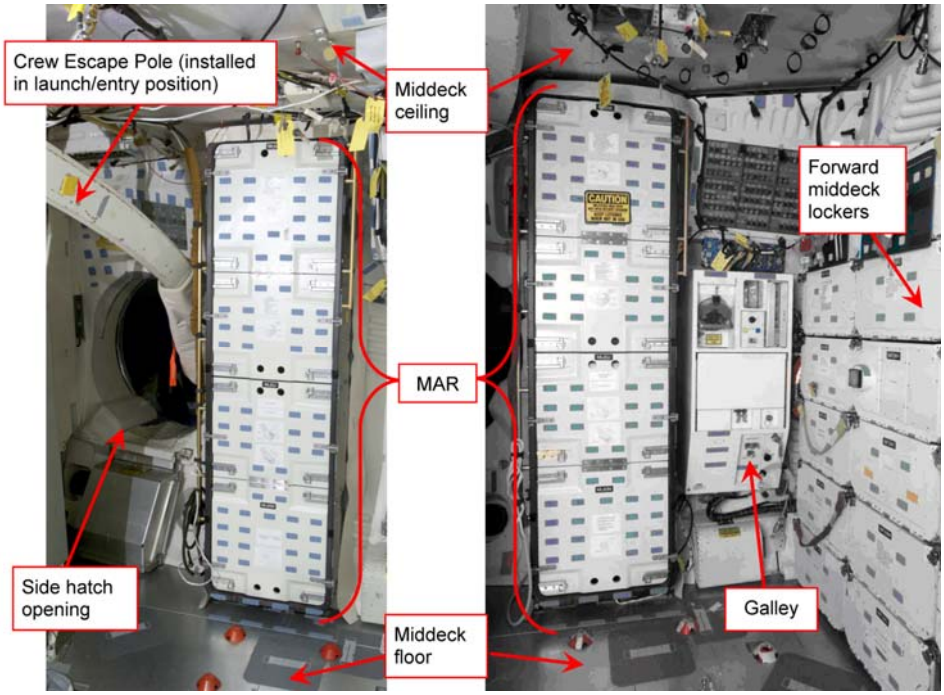


Figure 2.4-47. *Middeck accommodations rack*. [Picture from a shuttle training mockup in the JSC Space Vehicle Mockup Facility]

The MAR structure was made of a carbon fiber/epoxy composite with an aluminum honeycomb core. Two doors, which were hinged in the middle, face inboard into the CM, and were each held closed by eight locking spring latches. The MAR had two aluminum handrails that were attached to facilitate crew restraint and mobility. One handrail was mounted on the forward edge of the MAR and spanned from the top to the middle of the MAR. The other handrail was mounted on the aft edge of the MAR and spanned approximately three-quarters of the height of the MAR.

The MAR weighed 105 lbs. empty and could accommodate 12 ft^3 and over 200 lbs. of stowed items. Shelves could be bolted inside the compartment to subdivide the compartment. Stowed items were packed either in cargo transfer bags (CTBs) or foam cutouts to protect against damage.

For STS-107, the MAR contained payload general support computers (PGSCs), cables, a printer, the vacuum cleaner and attachments, medical kits, shuttle urine pretreat assembly (SUPA) hoses, and one can of LiOH. The total weight of the MAR (structure and return contents) was approximately 200 lbs.

Approximately 75% of the MAR structure was recovered, mostly intact. The blue areas in figure 2.4-48 represent the MAR structure that was recovered. The portions that were not recovered included the upper one-fourth of the MAR structure (including the upper attachment bracket), the top half of the upper door, a portion of the bottom half of the lower door, and the bottom surface of the MAR structure (including floor attachment brackets).

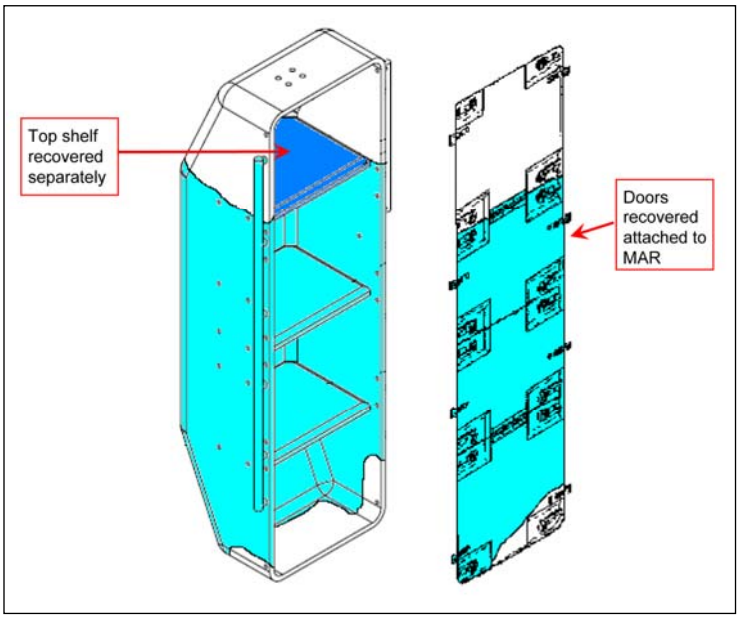


Figure 2.4-48. *Recovered middeck accommodations rack structure* (shown in blue).

The MAR was recovered with the middle and bottom shelves still attached. The top shelf was recovered separately. The lower door and the bottom half of the upper door were recovered with the MAR structure, as were the contents of the compartment between the middle and bottom shelves (CTBs with the printer and the vacuum cleaner (figure 2.4-49)). The contents of the compartment between the middle and top shelves and the contents of the compartment below the bottom shelf were recovered separately from the MAR structure and were highly fragmented.

Figure 2.4-49. *Main middeck accommodations rack structure, as found*.

The lack of impact witness marks on the MAR aft handle and the aft wall upper surface indicates that the MAR did not impact the CES pole, which is mounted just inches aft of the MAR. The lack of impact witness marks on the MAR starboard surfaces indicates that the MAR did not impact the seat 5 structure, which is mounted inches starboard of the MAR. Ballistic analysis indicates that the MAR separated from the CM shortly after the CMCE. Based on the failure of the fasteners securing the MAR attachment bracket to the middeck floor, it is concluded that the MAR separated from the middeck floor mostly to fully intact. These conclusions support the general findings that the middeck breakup was rapid and expansive, with very little interaction between major structures, and occurred at the onset of the CMCE.

Crew escape system pole

The purpose of the crew escape pole is to guide the crew member under the left wing when bailing out of the orbiter during controlled, gliding flight. For launch and landing, the escape pole is located on the middeck and is mounted to the starboard (right) ceiling and port (left) wall just forward of the side hatch (figure 2.4-50). During on-orbit operations, the escape pole is removed and stowed against the middeck ceiling.

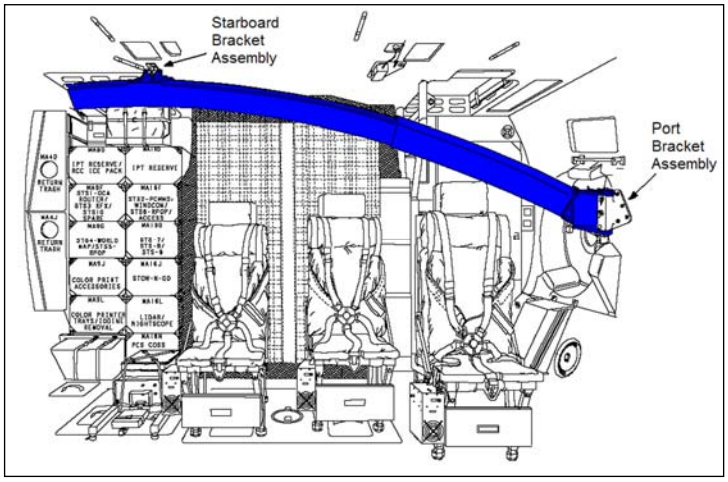


Figure 2.4-50. ***Crew escape pole in launch/landing position, middeck, looking aft*****.**

The escape pole consisted primarily of a curved, spring-loaded telescoping aluminum cylinder and steel spring. It was contained within an aluminum housing. The complete assembly weighed 267 lbs. A magazine containing eight crew lanyards was attached to the port end of the pole housing near the side hatch tunnel (figure 2.4-51). In the event of a bailout, the orbiter side hatch is jettisoned pyrotechnically, the pole is deployed, and each crew member extends a D-ring and bridle from his/her parachute pack and attaches it to a snap hook on the outermost lanyard. As the crew member egresses the orbiter, the pole directs him/her beneath and away from the vehicle. Upon bailout, the forces on the lanyard and bridle initiate automatic parachute deployment.

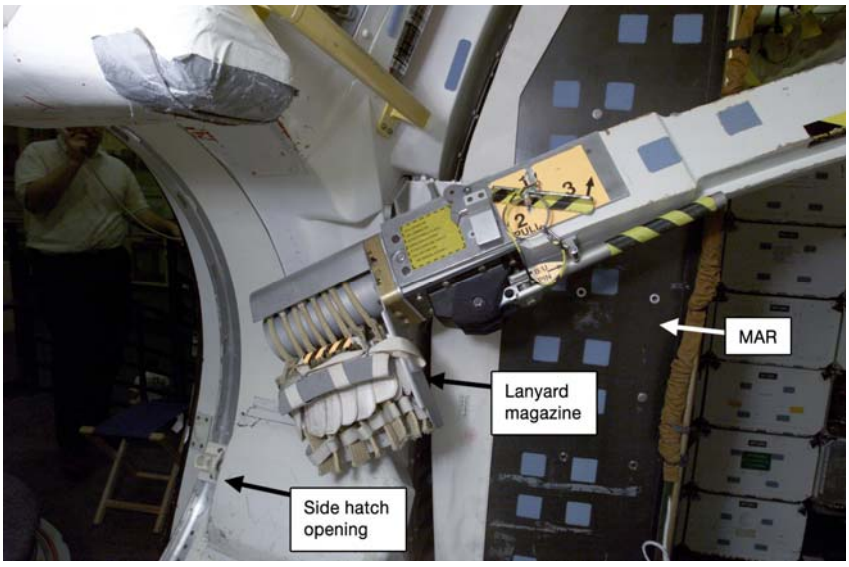


Figure 2.4-51. ***Port end of the crew escape pole, showing lanyard magazine, middeck, looking forward*****. [Picture from a shuttle training mockup in the JSC Space Vehicle Mockup Facility]**

The SCSIIT database contains detailed information regarding the analysis performed on the CES pole and the conclusions made from the analysis. The conclusions and their relevance to the CM breakup are presented here.

The debris indicates that the pole was installed in the launch/landing position. There is no evidence to suggest that the pole had been deployed (or that the side hatch had been jettisoned[3]). The pole housing, the main pole, and the extension pole (figure 2.4-52) were recovered separately. The starboard end of the housing, including the middeck ceiling mounting bracket, shows evidence of thermal damage. The main pole and extension pole (figure 2.4-53) were recovered in relatively good condition, showing little mechanical or thermal damage. The main pole deployment spring was recovered separately from the housing, with the starboard end cap of the pole housing attached to the spring. Additionally, the portion of the CM port wall to which the pole attaches was recovered and analyzed. Several lanyards were recovered separately, but the lanyard magazine was not recovered.

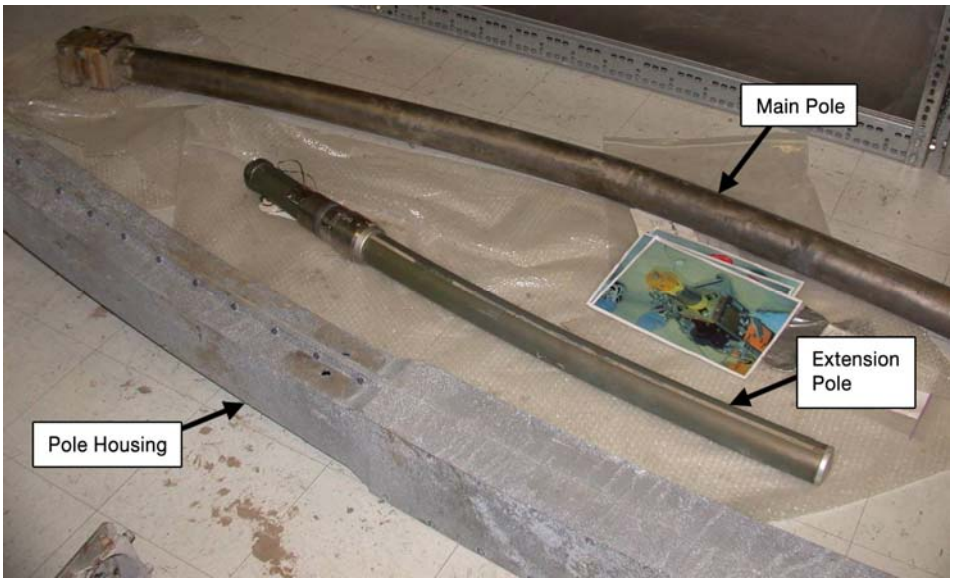


Figure 2.4-52. ***Pole housing, main and extension poles*****.**

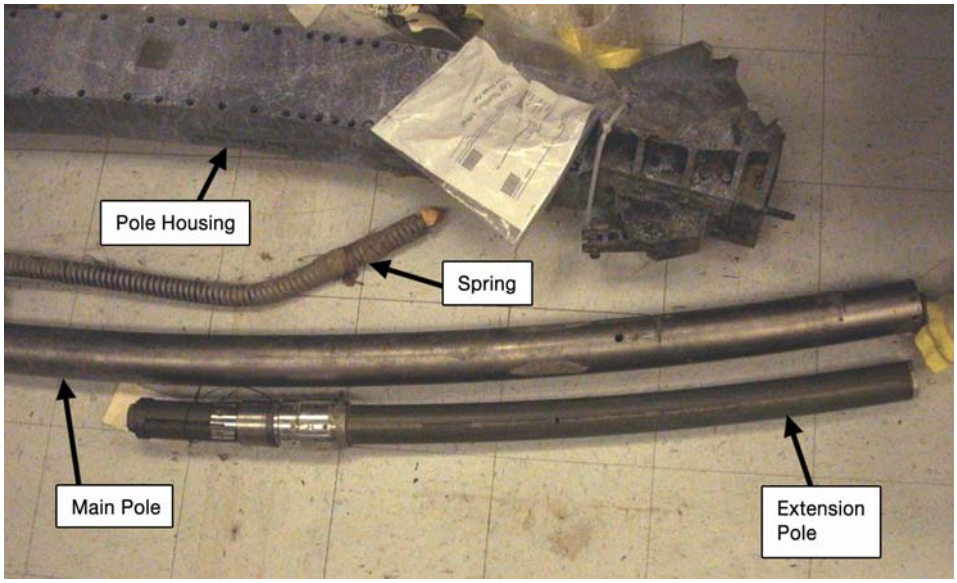


Figure 2.4-53. ***Port ends of housing, deployment spring, and main and extension poles*****.**

[3]Several pyrotechnic components were recovered. All indicated that the hatch jettison system had not been activated.

Deformation of the upper portion of the knuckle that attaches to the CM indicates that it experienced an upward cantilever load (figure 2.4-54). This could be caused by the starboard end of the pole moving upward. This suggests that the flight deck separated from the middeck roughly at the flight deck floor/ middeck ceiling level.

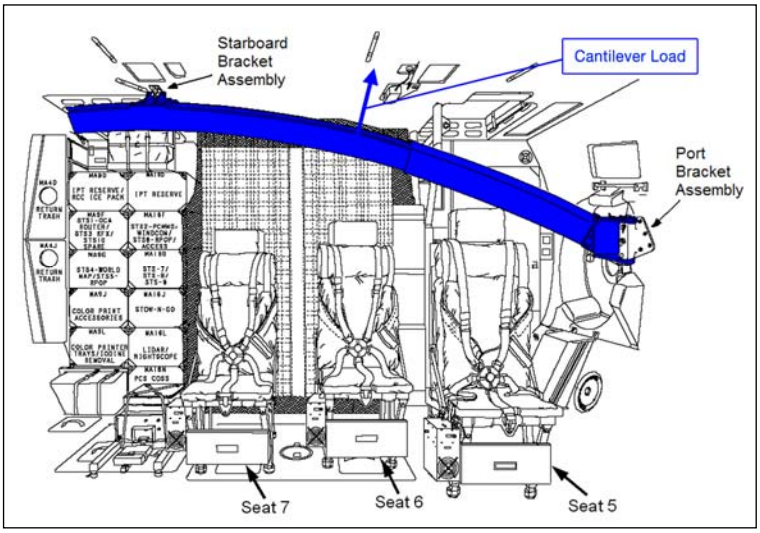


Figure 2.4-54. ***Crew escape pole in launch/landing position, middeck, looking aft, with direction of cantilever load noted from debris.***

The relative lack of significant witness marks on the pole housing indicates that it experienced very few impacts. These conclusions support the earlier conclusions that the middeck breakup was rapid and expansive, with very little interaction between major structures.

Flight deck instrument panels

Many of the flight deck panels were recovered and identified. Mechanical and thermal damage to the recovered panels was evaluated to assist in understanding the sequence of the breakup in an attempt to identify the location of the initial breach in the CM. Some of the panel parts were severely torn and deformed, yet some were mostly intact with less damage. Most of the recovered panels were photographed as orthogonally as possible under consistent lighting conditions, and the images were imported into a 3-dimensional computer model of the orbiter flight deck to create a virtual reconstruction of the flight deck (figures 2.4-55 and 2.4-56) (see discussion of virtual reconstruction in Chapter 4).

Figure 2.4-55. ***Intact orbiter flight deck from the Shuttle Mission Simulator.***

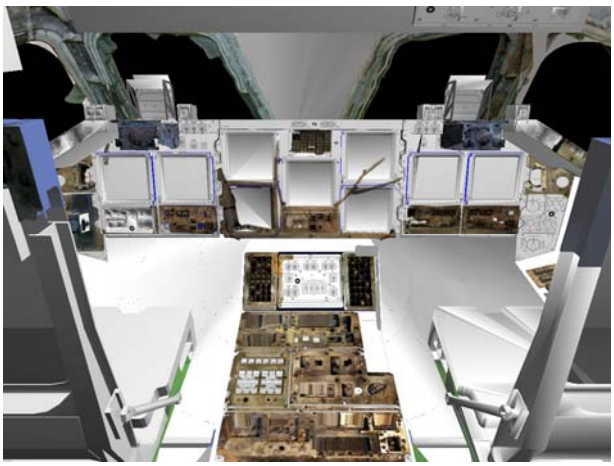


Figure 2.4-56. ***Virtual reconstruction of the recovered*** **Columbia** ***flight deck panels*****.**

This virtual reconstruction was studied to investigate whether there were clear indications of a thermal or structural breach. However, adjacent panels were seen to have received significantly different amounts of thermal damage. This indicated that some panels broke off earlier than other panels, or were temporarily shielded from the thermal flow during the breakup. The damage varied greatly from one panel to the next, indicating a chaotic breakup sequence. No clear evidence of the initial breach location could be determined.

2.4.4.3 *Orbiter window analysis*

At least one window frame (CM or FF) was recovered from every window. Two large debris assemblies were recovered (figures 2.4-57 and 2.4-58). Each large piece is a complete assembly of three CM forward window frames (Windows 1, 2, and 3 and Windows 4, 5, and 6) with some broken glass pieces still captured in their retainers. Three CM window frames of Windows 7, 9, and 10 were recovered in separate pieces since they were not connected to each other by heavy CM skin structures (figure 2.4-59). Five thermal frames (FF) were also recovered.

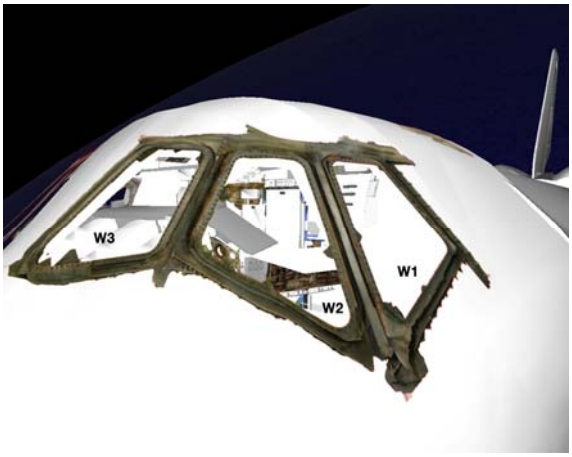


Figure 2.4-57. ***Port view looking aft, recovered*** **Columbia** ***crew module Windows 1, 2, and 3*****.**

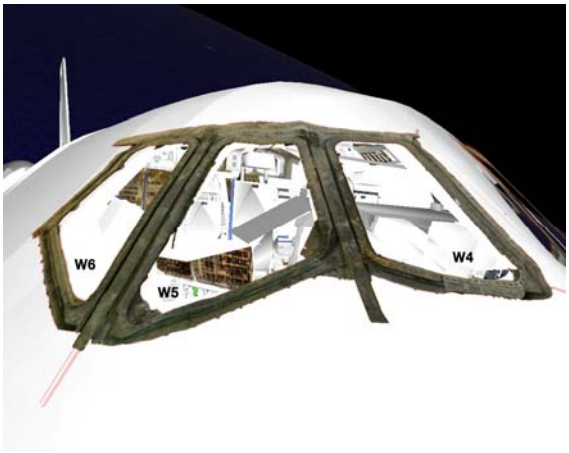


Figure 2.4-58. ***Starboard view looking aft, recovered*** **Columbia** ***crew module Windows 4, 5, and 6*****.**

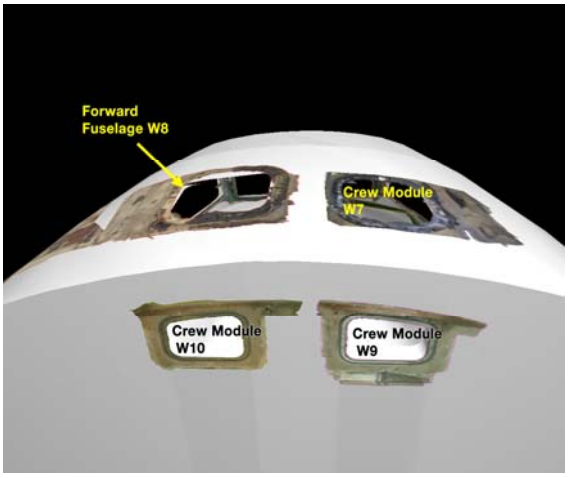


Figure 2.4-59. ***View looking forward, recovered*** **Columbia** ***crew module Windows 7, 9, and 10 and forward fuselage Window 8*****.**

All CM window frames are heavily reinforced to limit the window retainer deflection when the CM is exposed to internal pressure load and flight loads. The redundant panes are also made from the same material as the outer thermal pane (fused silica) so that if the outer thermal pane failed, the middle redundant pane still could be able to take some limited thermal flow for a short time.

Debris evidence suggests that most of the CM skin structures surrounding the CM flight deck windows had been melted away during the event. One exception is the upper edge of CM aft Windows 9 and 10, which shows mechanical fracture edges.

To summarize previous conclusions, there is no evidence to suggest that failure of the CM window panes was the cause of cabin depressurization. The debris field shows that no glass from the redundant panes, which would have had to have failed to lose pressure, was recovered west of the main forebody debris field. Most of the loose glass from all panes was recovered from the main forebody debris field, suggesting that the windows shattered during the CMCE. The departure of the FF Windows 1 through 6 and the FF arrow-head panel may have triggered the departure of the rest of the FF canopy by aerodynamic forces.

The recovered glass showed a marked discoloration. Discussions with the window subsystem manager confirmed that the window appearance did not match the appearance of thermally discolored glass. Close inspection showed that the discoloration appeared to be deposited material. Analysis of the deposition on the glass was conducted to determine the sequence of events experienced by the windows.

Evaluation was restricted to only thermal (outer) pane glass from which the location was positively identified. Based on thickness, one piece of glass was identified as either Window 3 or Window 4, but for purposes of this assessment this was considered sufficiently specific. When possible, glass samples were obtained from what remained in the various window frames.

For purposes of this report, the deposition that coated the glass was referred to generically as the “char layer,” regardless on which window it formed. Because the char layer was suspected to have formed at relatively high altitudes and temperatures, it was presumed that it remained intact from formation through eventual ground impact. Therefore, it was assumed that the char layer was sufficiently adhered to the glass surfaces such that all lightly attached particles were either deposited later in the breakup sequence or were field contamination. The harvested samples were cleaned in a laboratory setting using standard preparation techniques. Figure 2.4-60 is an example of an extracted thermal pane in its cleaned state, prior to sectioning. Black lines on the images denote approximate sectioning planes.

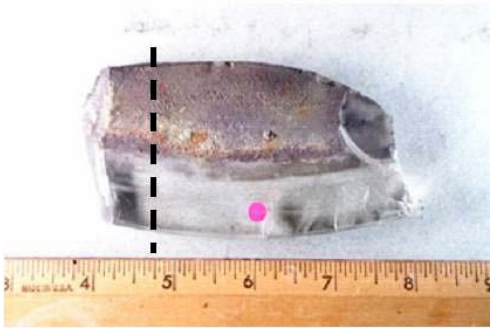


Figure 2.4-60. *Window fragment removed from the thermal pane frame from Window 8.* The black dashed line denotes the sectioning plane. The red dot was used to indicate the outward face of the pane.

Various aspects of char layer characterization were performed by the JSC Materials and Processing Office, the JSC Astromaterials Research Office, the KSC Failure Analysis and Materials Evaluation Branch, and the White Sands Test Facility. Electron and light microscopy, X-ray diffraction, powder diffraction, layer metrology, and phase characterization were all performed.[4,5]

The char layer coverage observed on Windows 3, 4, and 5 appeared relatively translucent when placed in front of a light source. The coloration of the char layer from these samples varied from a brownish tan to a dark brown hue. The translucent characteristics of these samples implied a relatively thin deposit thickness. By comparison, samples from Windows 7 and 8 were notably opaque and had more of a blackened appearance. The surface texture of the window samples appeared relatively rough, consistent with re-solidified molten deposition. Visual examination alone was not able to assess the relative thickness of the deposits on Windows 7 and 8.

Cross sectioning of the various thermal pane window remnants was performed using standard cross-sectioning metallographic techniques. The char layer deposits on all thermal panes examined were evaluated and characterized based on mean thickness and constitution of the deposits (voids, inclusions, etc.). In general, the thermal panes for Windows 3, 4, and 5, were covered by a char layer deposit that ranged from a few microns (μm) to nearly 100 μm. The deposit in the char layer for these forward-facing windows was not continuous; the areas without deposition retained their translucence. In the regions of continuous deposit, the thickness profiles were irregular, indicating sporadic deposition of the material. For regions where deposits were thick, void entrainment (porosity) was evident. In contrast to the exterior surfaces of the forward-facing windows, the exterior surface of the thermal panes for Windows 7 and 8 samples were covered by a char layer deposit that ranged from 30 μm to nearly 500 μm and appeared continuous. Although the char layers on these windows varied in thickness, the mean thickness was on the order of 50 to 100μm. The measurements taken on the interior and fracture surfaces of Windows 7 and 8 samples were consistent with those of the forward-facing windows.

In-depth materials analysis was performed on the char layers for the forward windows to compare to the char layer on the overhead windows.

While almost every window sample contained multiple metallic species, aluminum was the predominant component of the char layer with other metal species existing in either discrete features or within a very narrow region of the layer. Interspersed in the layers were globules of silicon throughout the thickness. The spectroscopic signature indicated that the majority of this layer was a heavily oxidized aluminum amalgam consistent with a 2000-series aluminum alloy. This series aluminum alloy is used in the FF and CM structure.

A porous layer of aluminum was deposited on all window pane (forward and overhead thermal) samples, including both external and fracture surfaces. The porous nature of this feature in the char layer was considered to be a result of a dynamic process of deposition when the fragments of the glass, and the structure that contained them, possessed a high relative and turbulent motion to the deposition source. The deposition source, likely a 2000-series aluminum alloy and probably from the aluminum 2024 FF structure, was dispersed in the form of molten/semi-molten particles that partially cooled and/or oxidized prior to

[4]J. D. Olivas, L. Hulse, B. Mayeaux, S. McDanels, P. Melroy, G. Morgan, Z. Rhaman, L. Schaschl, T. Wallace, and C. Zapata, *Examination of OV-102 Thermal Pane Window Debris – Final Report*, KSC-MSL-2008-0178 (in press).
[5]J. D. Olivas, M. C. Wright, R. Christoffersen, D. M. Cone, and S. J. McDanels, *Crystallographic oxide phase identification of char deposits obtained from space shuttle* Columbia *window debris*, Acta Materialia, 2008 (in press).

impacting the glass substrate. Additionally, this feature of the char layer was irregular in thickness and also had distinct particles of other oxidized metal systems.

The formation of this porous char layer can be explained by several scenarios. The deposition source could have been nearby, but the area had highly turbulent relative motion. Or, the deposition source could have been a significant distance from the window, creating high dispersion in the molten material flow. This is less likely because liquid droplets would cool quickly and be less likely to adhere. Finally, the deposition could have resulted from the glass passing through a rapidly solidifying vapor surrounding the forebody as a result of thermal erosion of materials. It is conceivable that all three processes were occurring, either simultaneously or discretely. Given the debris field of recovered glass and the presence of the deposition on the fracture surfaces and inner panes, it is concluded that this deposition event occurred between the CMCE and the TD.

In addition to this porous layer, the two overhead thermal panes showed unique layers not seen on the forward windows. On these two panes, two additional layers, which contained titanium in appreciable quantities, were identified below the porous aluminum-rich layer. The overhead thermal pane window char was loosely characterized into three layers: a titanium-rich layer closest to the glass, a titanium-aluminum-rich layer, and the porous aluminum-rich layer that was described above. Crystallographic investigation of the titanium-rich region adjacent to the glass indicates that the nodules are consistent with TiO_2, a titanium oxide.

The aluminum-rich, thinner outer char layer appeared to be deposited via a different mechanism than the lower titanium-rich layers. Based on the lack of porosity, the environment in which the lower titanium-rich layers were deposited was likely substantially less dynamic than the environment during the deposition of the porous aluminum-rich top layer. This continuity of the titanium-rich layers suggested that the source was likely in close proximity to the windows; longer distances from the source would result in a more dispersed and turbulent flow, and probably result in material cooling/solidifying prior to impact on the window, which was not seen. The transition from titanium to a mixture of aluminum/titanium seems to imply that some time after the titanium began to deposit on the windows, aluminum from a nearby source also began to deposit on the windows. Analysis of the deposition on the carrier panel tile surrounding the overhead window panes showed similar deposition layer patterns. This deposition occurred prior to the breakup of the windows and structure, which was presumed to have begun at the CMCE. The existence of this distinctive three-zone char layer only on the exterior surfaces of the thermal panes from Windows 7 and 8 also supports the previous debris field finding that the windows were largely intact through the CMCE.

The previous findings were considered interesting enough to lead to a search for the source of the titanium and aluminum. The intention was to discover whether there was any information that was suggestive of the structural condition and the orientation of the forebody prior to the CMCE. A search for forward structures containing titanium showed that the nearest source of titanium material to the windows was the forward PLBD rollers. These rollers are made of titanium and aluminum with an Inconel sleeve. The PLBDs of the orbiter, when closed, rest on these roller mechanisms, which are attached to the X_o 582 ring frame bulkhead upper arch in the forward part of the payload bay. These rollers are in close proximity to Windows 7 and 8. The structure that supported the roller components was primarily composed of 2024 aluminum alloy and also displayed evidence of significant thermal erosion. However, regions farther away from the rollers showed minimal signs of thermal erosions; green Koropon primer was still present on portions of these remote regions (figures 2.4-61, 2.4-62, and 2.4-63).

Figure 2.4-61. Endeavour, *OV-105, X_o 582 ring frame bulkhead arch with rollers*. Circled in red is one of the eight rollers.

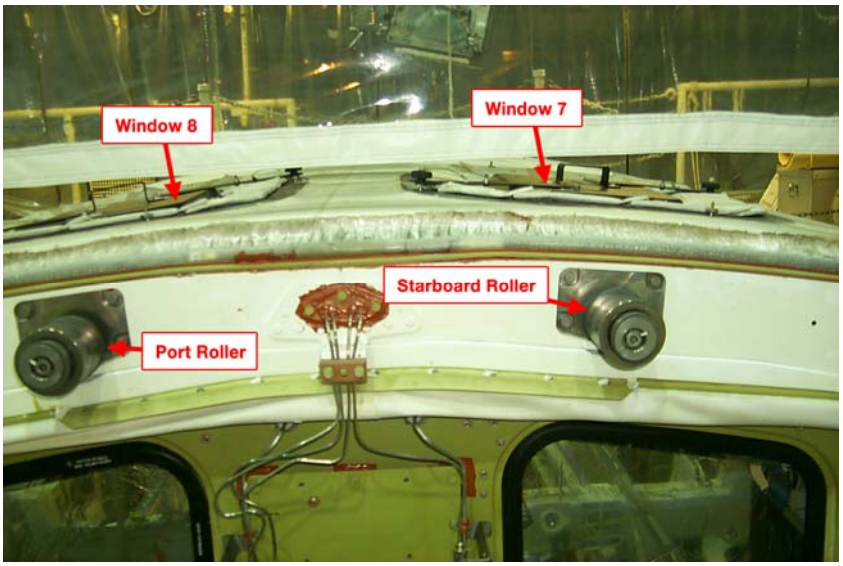


Figure 2.4-62. *Nominal configuration (*Endeavour*, OV-105) of the two inner rollers and overhead windows*.

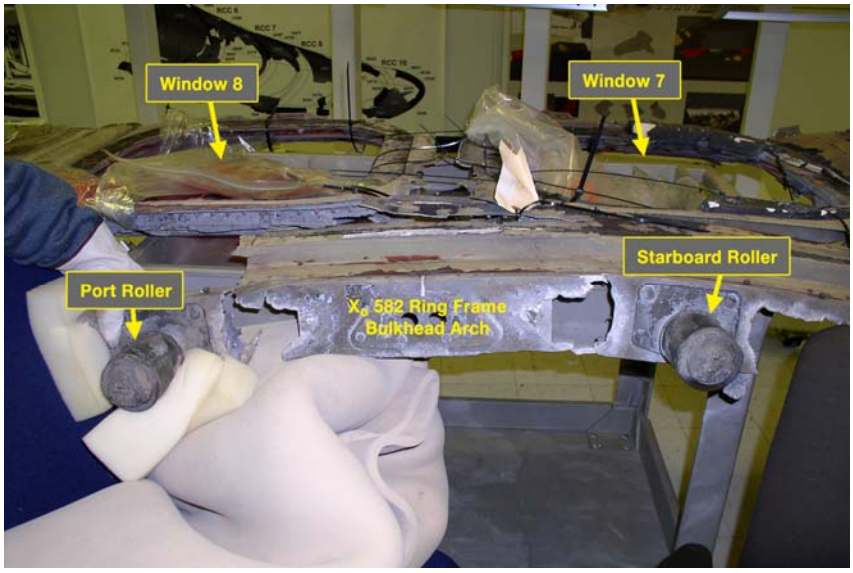


Figure 2.4-63. Columbia *debris for same location*. Note eroded rollers and eroded region of the X_o 582 ring frame bulkhead arch between the rollers.

The recovered rollers for the location closest to the overhead windows both showed significant signs of erosion. While these rollers were not the only components made of titanium, they were the only ones recovered that possessed the proper material, proximity, and thermal indications and were, therefore, concluded as the source location. Given these findings, the rollers were the most probable source of titanium causing the deposition on the windows.

Since the rollers are located on the X_o 582 ring frame bulkhead arch aft and below these two windows, and protected by the PLBDs, the PLBDs must have been compromised or fully departed while the bulkhead arch was still attached to the CM. Additionally, the forebody must have been traveling aft-end forward for some period of time to have the directional thermal flow that caused the titanium to "vaporize" flow over the glass external surface (figure 2.4-64). Since no titanium was found on the internal surface of the thermal panes or on the external surface of the Windows 7 and 8 redundant panes, it can be concluded that the thermal panes were intact at the time of this event. This is consistent with other findings showing that the FF remained with the CM until breakup of both elements.

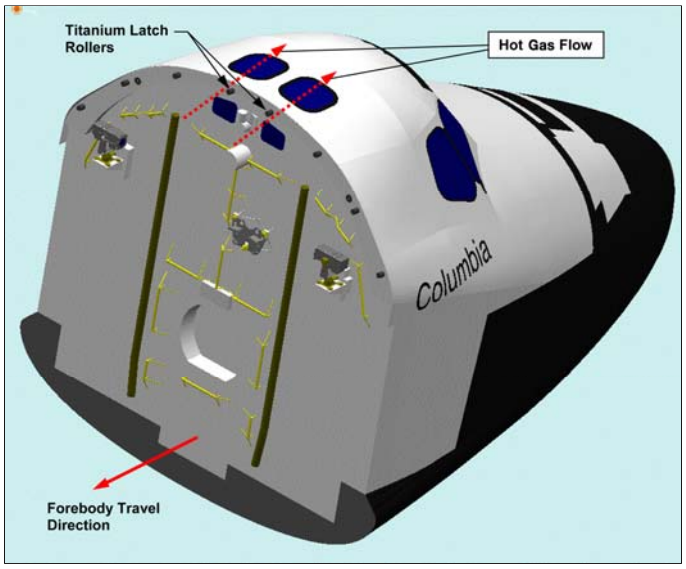


Figure 2.4-64. ***Titanium deposit on windows indicates forebody traveled backwards.***

Finding. Windows 7 and 8 experienced a titanium deposition event that occurred prior to window breakup.

Finding. The most probably source for the titanium deposition on Windows 7 and 8 was the PLBD rollers. These rollers were not exposed to heat flow until after the PLBDs were compromised.

Finding. All the windows had an aluminum-rich deposition, which was consistent with a turbulent process.

A complete discussion of the thermal mechanisms that may have led to the titanium deposition is covered in Section 2.1.6.8.

2.4.5 Synopsis of forebody breakup sequence

The orbiter breakup (CE) or subsequent impacts between the CM and the FF caused small mechanical breaches on the CM skin. The FF shell stayed with the CM until the CMCE at GMT 14:00:53. At the CMCE, the FF most likely separated in two large segments, upper and lower. The departure of the FF arrowhead panel and FF Windows 1 through 6 may have triggered the departure of the rest of the FF canopy by aerodynamic forces.

Once the protective FF structure departed, the CM side skin was consumed by thermal exposure. Breakup of the CM structure occurred as a consequence of aerothermal heating and aerodynamic loading. The CM broke up with the middeck and forward bulkhead departing, most likely from starboard to port. The middeck breakup was rapid and expansive, with very little interaction between major structures.

The flight deck remained intact for some period after the middeck separated. The flight deck likely remained with the airlock and the aft bulkhead. As a relatively intact "pod" with a high ballistic number, the flight deck experienced more thermal exposure until its final breakup completed the CMCE.

The forebody was broken down to subcomponents that were too small and dispersed to see on video at GMT 14:01:10. This was described as the TD. Cascading failures and thermal damage were still occurring, but the CM no longer had any structural integrity at this time.

> ***Recommendation L3-1.*** Future vehicles should incorporate a design analysis for breakup to help guide design toward the most graceful degradation of the integrated vehicle systems and structure to maximize crew survival.

Chapter 3 – Occupant Protection

3.1 *Crew Seats*
3.2 *Crew Worn Equipment*
3.3 *Crew Training*
3.4 *Crew Analysis*

3.1 Crew Seats

The seats, which are the interface between the vehicle structure and the crew members, provide a source of data for the accelerations and thermal environments that the crew members experienced. This section provides background information on the design and construction of shuttle crew seats, and describes the detailed analyses performed on the *Columbia* crew seats. These analyses included review of the recovered videos recorded on orbit, review of the materials analyses performed by the *Columbia* Accident Investigation Board (CAIB)/Crew Survival Working Group (CSWG), and inspection of debris items including microscopic inspections of the inertial reels mechanisms and straps.

The following is a summary of the findings, conclusions, and recommendations for this section:

Finding. Evidence from the inertial reel straps indicates that the seats 1, 2, and 3 straps were mostly extended at the time of strap failure. The seats 4, 6, and 7 straps were extended during a material deposition period (seat 4 at least 8 in., or ~36% extended; seat 6 at least 21.25 in., or ~96% extended; and seat 7 at least 21.5 in., or ~98% extended). Medical evidence (see Section 3.4) indicates that some of the crew members received injuries consistent with insufficient upper body restraint.

Conclusion L2-2. The seat inertial reels did not lock.

Conclusion L2-3. Lethal injuries resulted from inadequate upper body restraint and protection during rotational motion.

Recommendation L1-3/L5-1. Future spacecraft crew survival systems should not rely on manual activation to protect the crew.

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments.

Recommendation L2-8. The current shuttle inertial reels should be manually locked at the first sign of an off-nominal situation.

Finding. The seat 2 inertial reel strap exhibits “strap dumping” failure features. The strap failed progressively, possibly due to damage to the lateral edge of the strap from contact with the sharp edge of the strap pass-through slot.

Recommendation L2-5. Incorporate features into the pass-through slots on the seats such that the slot will not damage the strap.

Finding. All inertial reel straps are tested with static loads at room temperature. Load testing has not been conducted to determine the loads required to fail the straps at elevated temperatures or under dynamic loads. Testing has not been conducted to determine the material properties (combustion vs. chemical degradation vs. melting) in a high-temperature/low-oxygen (O_2)/low-pressure environment.

Recommendation L2-6. Perform dynamic testing of straps and testing of straps at elevated temperatures to determine load-carrying capabilities under these conditions. Perform testing of strap materials in high-temperature/low-oxygen/low-pressure environments to determine materials properties under these conditions.

Finding. While all seat piece-parts include serial numbers, only the serial numbers of the inertial reels were recorded and tracked to a specific seat assembly. The lack of configuration management documentation hindered the process of ascribing the seat debris items to specific seat locations.

Recommendation A5. Develop equipment failure investigation marking ("fingerprinting") requirements and policies for space flight programs. Equipment fingerprinting requires three aspects to be effective: component serialization, marking, and tracking to the lowest assembly level practical.

3.1.1 Seat design and construction

Two types of seats are used on the space shuttle. The Pilot seats are used by the mission Commander (CDR) and Pilot (PLT), and Mission Specialist seats are used by all other crew members. Both types of seats provide for crew member positioning and restraint during launch, entry, and some on-orbit operations. Both types of seats are also designed to accommodate a fully suited crew member.

Seat positions are numbered 1 through 7, beginning with the CDR's position on the flight deck (seat 1) and ending with the starboard-most Mission Specialist seat on the middeck (seat 7). Seats 1 through 4 are on the flight deck (figure 3.1-1), and seats 5 through 7 are on the middeck (figure 3.1-2). Seats 1 through 5 are flown on all missions. Seats 6 and 7 are flown as required.

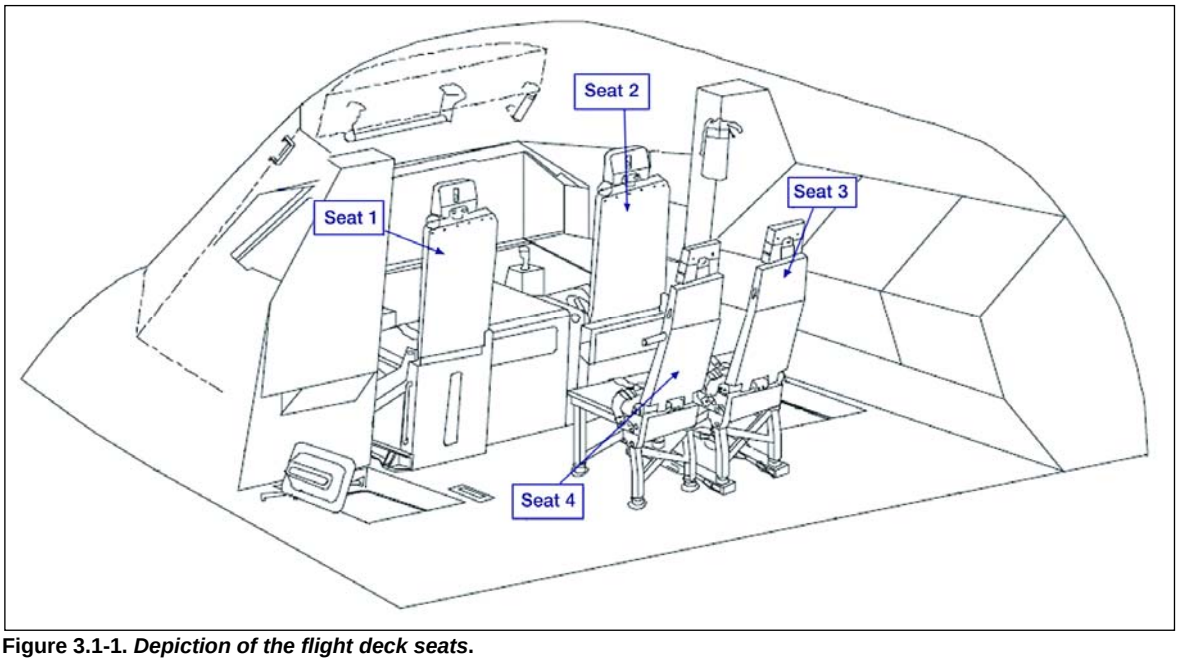


Figure 3.1-1. ***Depiction of the flight deck seats.***

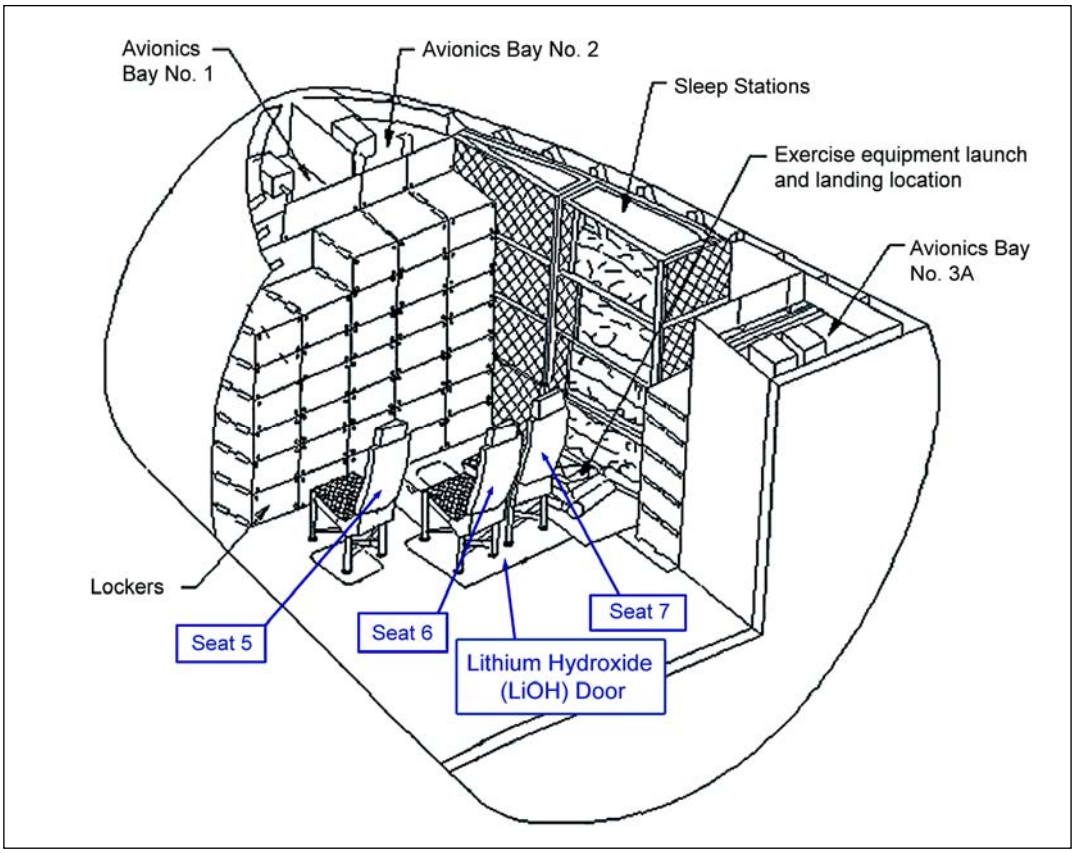


Figure 3.1-2. *Depiction of the middeck seats*. [Adapted from the Shuttle Crew Operations Manual]

The Pilot seats (figure 3.1-3) and Mission Specialist seats (figure 3.1-4) have several common design features. Both seat types have identical seatbacks, five-point restraints, headrests, and MA-8 inertial reels (a part of the restraint system).[1] The five-point harness restrains the upper torso with shoulder belts, and the lower body with lap and crotch belts. All belts connect to a rotary buckle that is permanently mounted to the crotch belt. The shoulder belts join to a single strap, the inertial reel lead-in strap, which attaches to the inertial reel mechanism mounted inside the seatback. The inertial reel will lock automatically due to accelerations pulling the strap out at 1.78 G to 2 G. A lever to manually lock and unlock the inertial reel is located on the left side of the seat pan. Both seat types accommodate the attachment of O_2 hoses and communications cables, and both have attachment brackets for cooling units used in conjunction with the crew member suits for crew member comfort.

[1]The MA-8 inertial reel is an off-the-shelf design used in military helicopter seats and was not designed specifically for the orbiter.

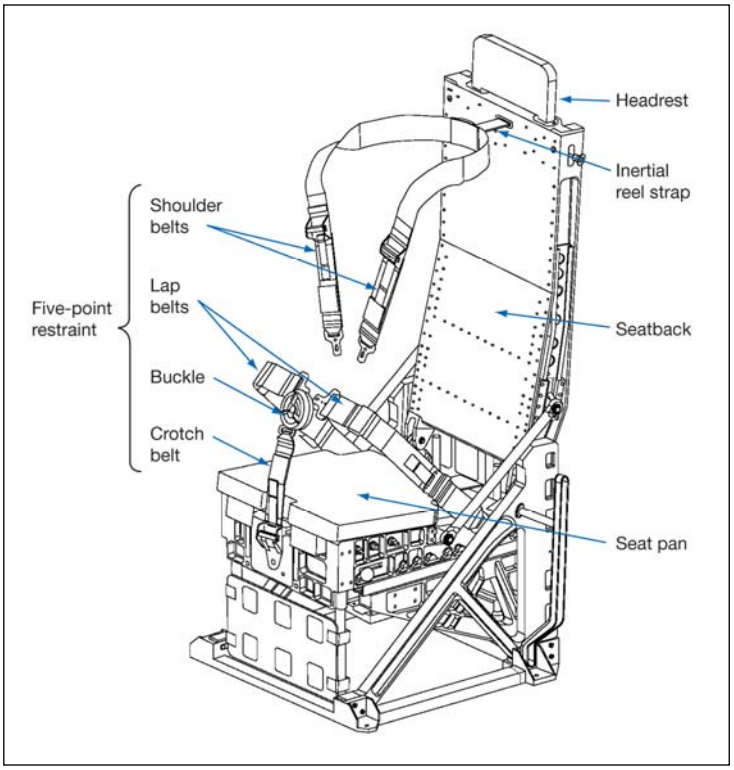


Figure 3.1-3. *Pilot seat*. [Adapted from the Space Shuttle Systems Handbook]

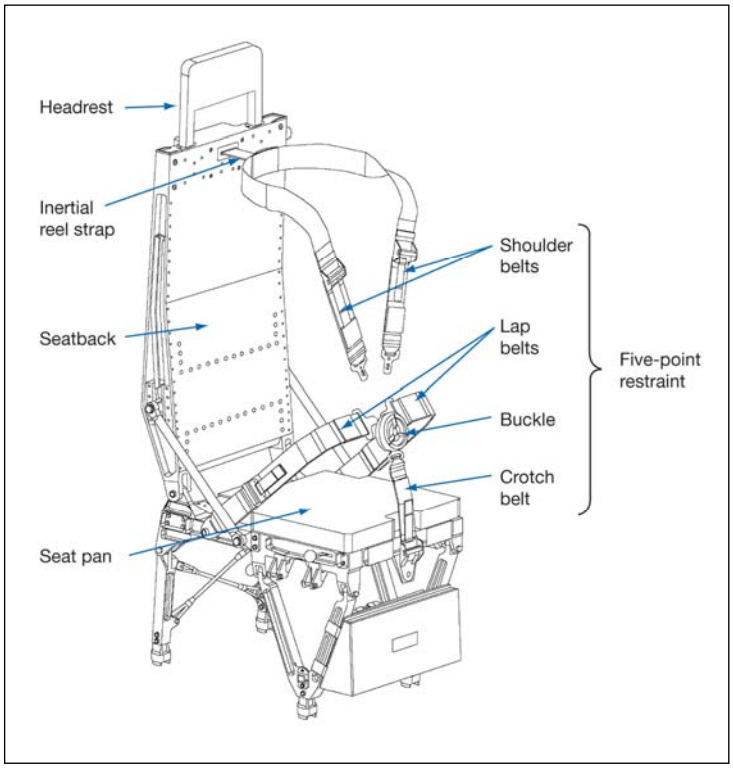


Figure 3.1-4. *Mission Specialist seat*. [Adapted from the Space Shuttle Systems Handbook]

The seat pan and base of the Pilot seats (seats 1 and 2) differ from the Mission Specialist seats. The Pilot seat base is permanently mounted to the flight deck floor and incorporates mechanisms providing up/down and forward/aft adjustability of the seat position. The Pilot seats also provide a mounting base for the rotational hand controllers (RHCs) (figure 3.1-5). RHCs are control sticks that are used by the CDR and PLT to control vehicle rotation about the roll, yaw, and pitch axes during ascent, orbit, and entry. The RHCs provide input to computers to actuate various vehicle control effectors (aerosurfaces and/or Reaction Control System (RCS) jets)).

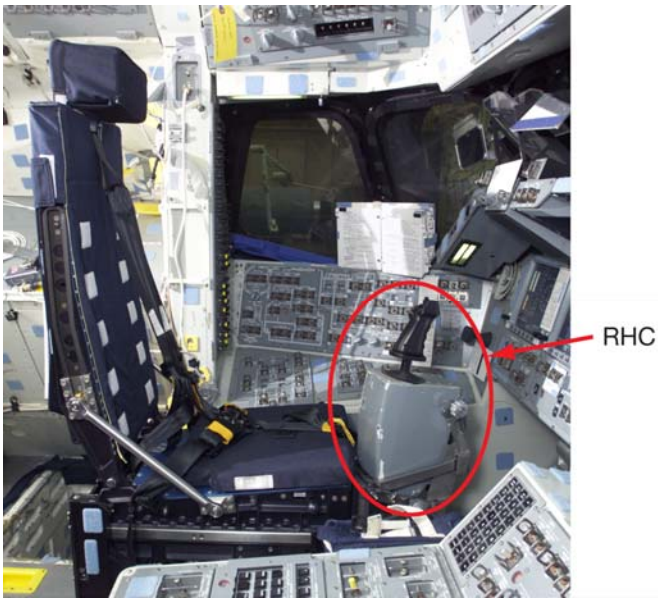


Figure 3.1-5. *Commander's seat and one of the rotational hand controllers, flight deck*. [Picture from a shuttle training mockup in the JSC Space Vehicle Mockup Facility, looking from starboard to port]

Crew members assigned seats 3 through 7 use the Mission Specialist seats for launch and entry. Five Mission Specialist seats were flown on STS-107. Seat 3 was mounted on a special sled assembly that was unique to *Columbia*.

The Mission Specialist seats have no base, but have foldable legs attached to the seat pan. The legs are equipped with quick-disconnect fittings to allow for seat removal and stowage. Once the shuttle is on orbit, all Mission Specialist seats are detached from the floor, folded, and stowed in various locations depending on crew preference. During deorbit preparation, the crew reinstalls the Mission Specialist seats.

3.1.2 STS-107 seat configuration

The recovered middeck and flight deck videos revealed information that was related to the configuration of the crew seats and other flight crew equipment. The recovered middeck video recorded deorbit preparation (D/O PREP) activities on the middeck. Although there is no credible timestamp[2] on the video, the STS-107 crew plan[3] offers some insight into when the recording probably occurred.

The video shows that all middeck seats were installed. The seat 5 crew member is suited, and the seat 1 and seat 6 crew members are donning their suits. Based on the crew D/O PREP plan, this 30-minute video probably recorded events from approximately Greenwich Mean Time (GMT) 11:40:00 to approximately GMT 12:10:00.

The recovered flight deck video is a 13-minute, 11-second video with sound that was recorded more than 1 hour after the middeck video described above. This video recorded entry events on the flight deck from approximately GMT 13:35:34 to GMT 13:48:45. It shows that all of the flight deck seats were installed and the flight deck crew members were properly restrained. This video provides a good view of the crew members in seats 1 and 2, and fair views of the crew members in seats 3 and 4. It also shows that all of the flight deck crew members were properly secured with seatbelts to prevent floating from the seats (tight lap belts). No slack in the belts is visible in the shoulder harnesses, and the crew members are able to move their upper bodies, indicating that the shoulder harness inertial reels are not locked, which is normal at this point during the mission.

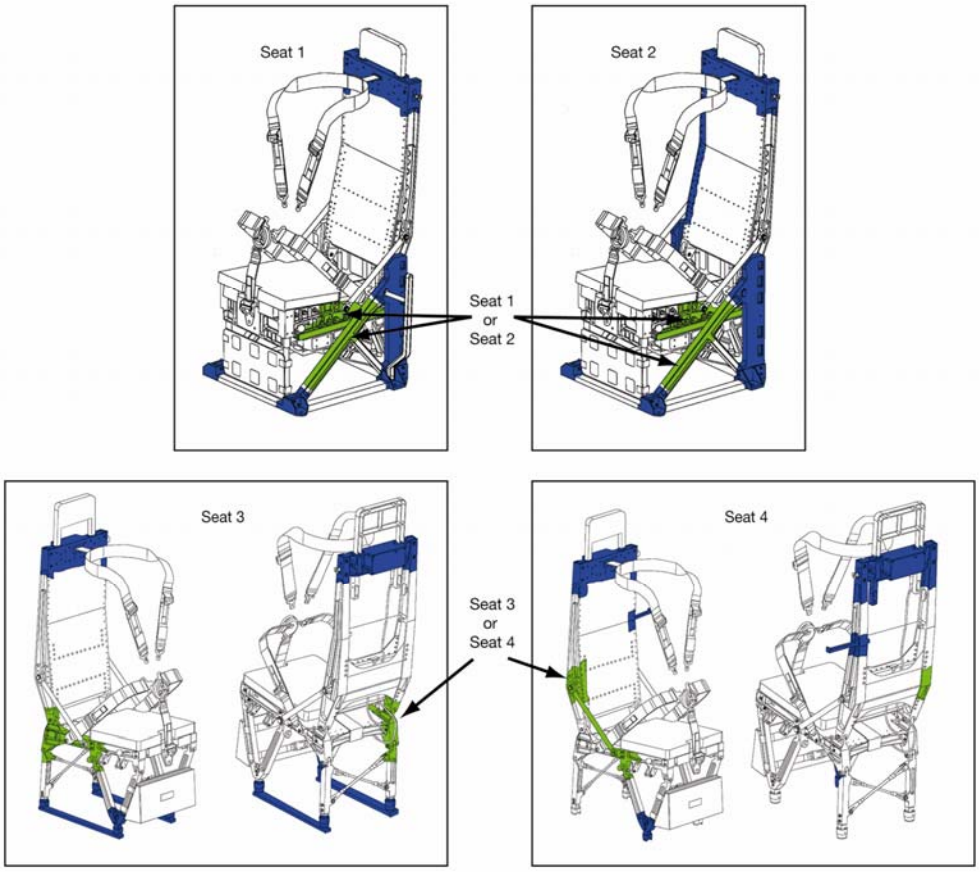
[2]The video has no air-to-ground audio, nor does it record any actions that can be time-verified through telemetry or onboard data recorders, so a precise time-stamp cannot be determined.
[3]Pre-mission, each shuttle crew develops a detailed deorbit preparation plan that is tailored from the formal Flight Data File procedures.

From video evidence, investigators concluded that all of the seats were installed and the flight deck crew members were properly strapped into their seats. Although no recovered video shows that the middeck crew members were strapped into their seats, medical findings and evidence in the seat debris described below confirms that two middeck crew members were fully strapped in and that one middeck crew member was at least partially restrained in the seat.

3.1.3 Seat structure

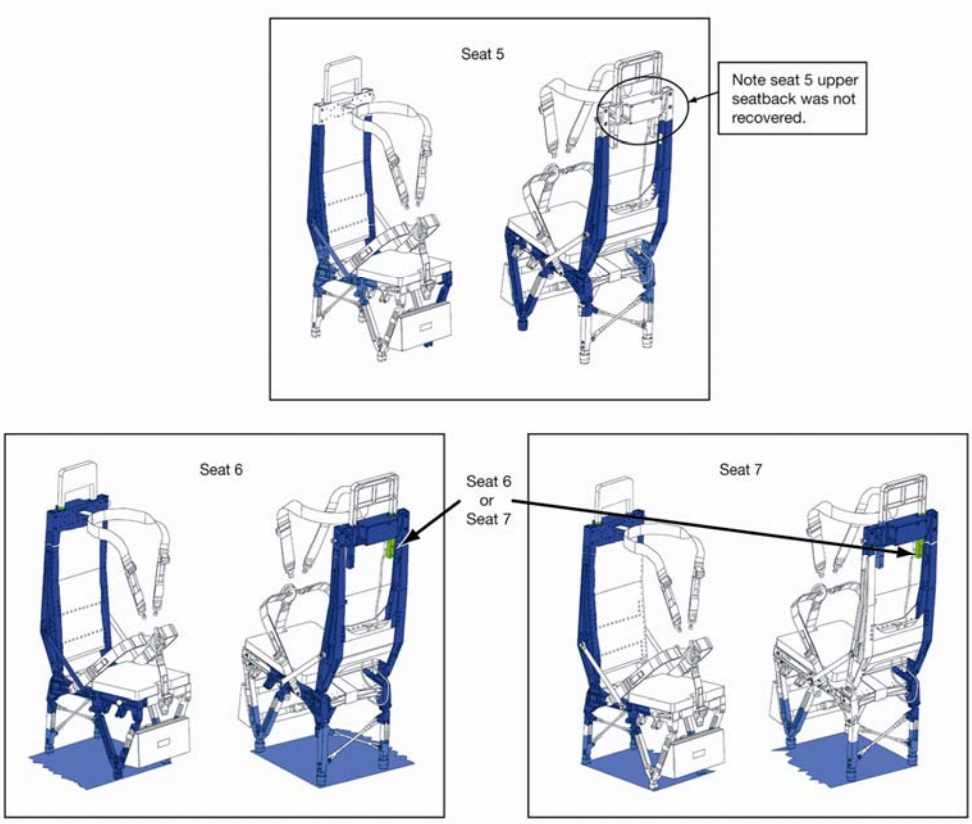
More than 68 pieces of seat structure debris were recovered. Positive assignment of the recovered seat debris to specific seat locations was difficult and required considerable analysis of subtle differences between the seats and the mounting locations because a significant portion of the lightweight seat design is common to all seven seats. Several of the recovered components were ascribable as coming from the flight deck seats (seats 1, 2, 3, and 4) and the middeck seats (seats 5, 6, and 7).

Figures 3.1-6 and 3.1-7 show seat structure pieces that were identified to each seat location. The blue items in these figures represent items that were positively identified to a seat location. The green items are those items that could be from one of two seats. Figure 3.1-8 shows the major seat structure debris pieces that could not be identified to a specific seat location (the colors in this figure distinguish the different pieces and are otherwise inconsequential). Although represented in figure 3.1-8 on just two seats, the seatback and seat restraint items could be from any of the seven seats. However, the seat pan items could only be from the Mission Specialist seats.



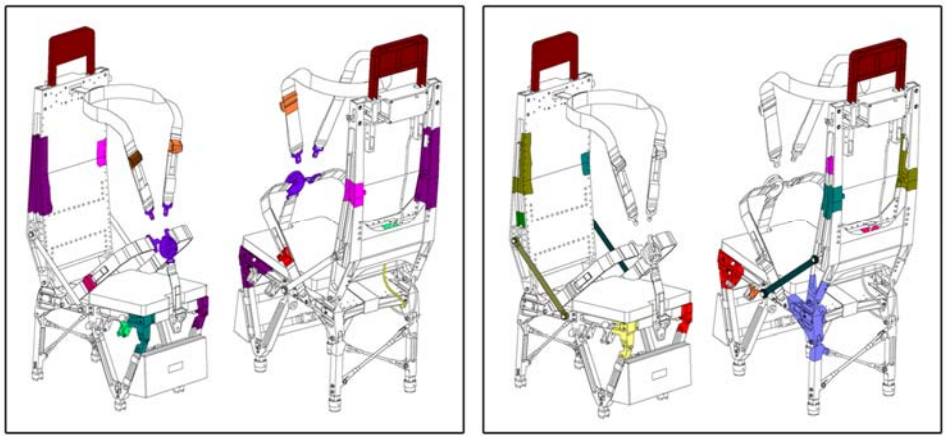
Blue items are positively identified to a specific seat;
green items could be from either one of the two indicated seats.

Figure 3.1-6. *Identified debris from the flight deck seats*. [Adapted from the Space Shuttle Systems Handbook]



Blue items are positively identified to a specific seat;
green items could be from either one of the two indicated seats.

Figure 3.1-7. *Identified debris from the middeck seats*. [Adapted from the Space Shuttle Systems Handbook]



Colors are used only to distinguish different debris items.

Figure 3.1-8. *Debris from unknown seat locations*. [Adapted from the Space Shuttle Systems Handbook]

The only consistent piece of seat debris that was positively identified to six seat locations (except seat 5) was a portion of the upper seatback that included the seat restraint inertial reel mechanism and strap.

Major differences in the magnitude of thermal exposure were identified on the flight deck vs. the middeck seat locations. The flight deck seat components, as well as the flight deck floor structure, were highly melted and/or deposited with splattered aluminum on all surfaces. Close inspection of the fracture surfaces on the flight deck seat attach points revealed deformation towards the vehicle starboard direction.

Materials analysis revealed that the flight deck seats collected deposits of melted aluminum from locations throughout the cabin area. Materials that were consistent with the bulkheads and outer pressure shell (2219 aluminum), the surrounding primary and secondary structure (2219, 2024, 2124, and 7075 aluminum), and the seats (2024 and 7075 aluminum) were discovered on the upper and lower surfaces of the recovered seat debris. The CAIB Report concluded,[4] and this report concurs, that the flight deck seats remained attached to the flight deck floor panels and adjacent to the surrounding structure during a period of significant thermal exposure and material deposition.

The magnitude and distribution of heating of flight deck seat components indicates a prolonged attachment to other crew module (CM) structure during exposure to heating. This indicates that the flight deck seat components were released from the CM later in the breakup sequence than the middeck seat components. Ground plots of debris recovery locations also support this conclusion (figure 3.1-9). Figure 3.1-10 compares a portion of the seat leg from a flight deck Mission Specialist seat component with a similar component from a middeck seat.

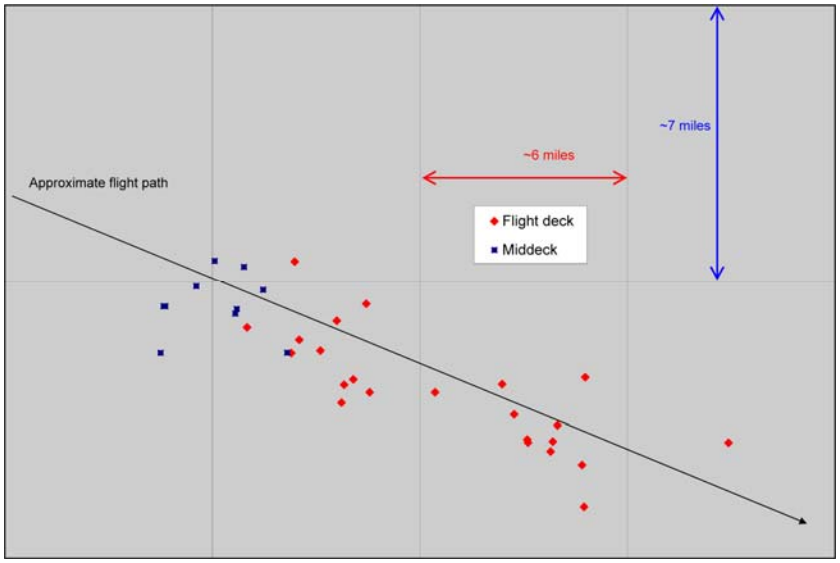


Figure 3.1-9. ***Recovery locations of seat structure debris.***

[4]*Columbia* Accident Investigation Board Report, Volume V, Appendix G.12, Crew Survivability Report, October 2003, p. 362.

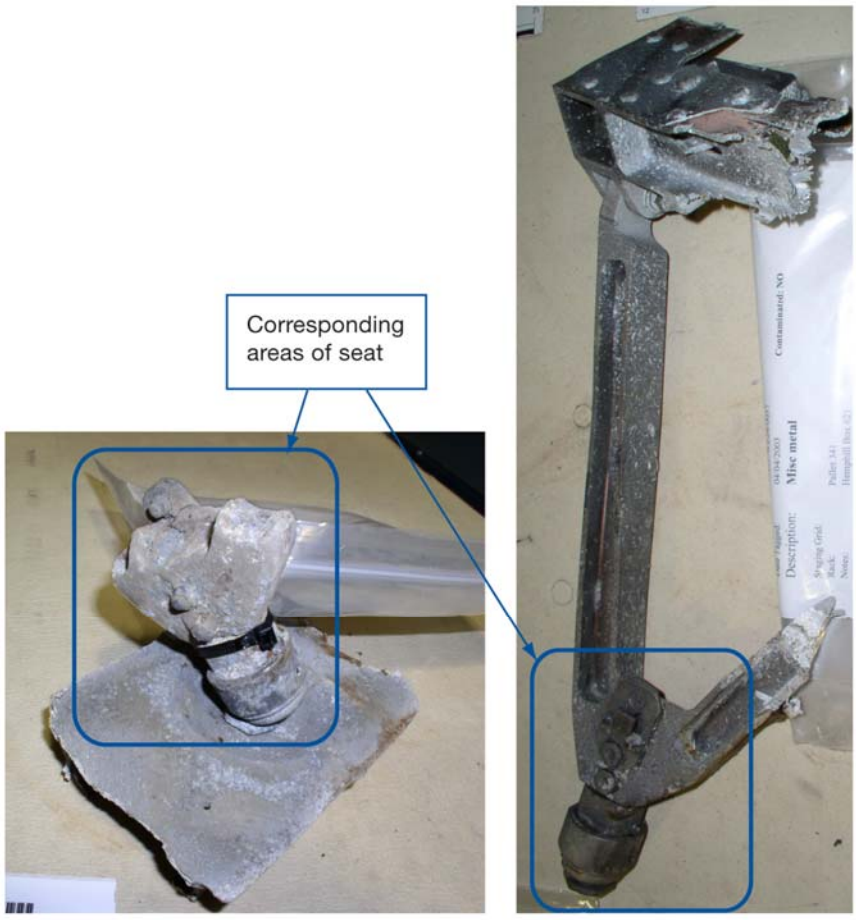


Figure 3.1-10. ***Comparison of seat debris from a flight deck Mission Specialist seat*** **(left)** ***and a middeck Mission Specialist seat*** **(right).**

Although all seat components experienced significant heating, the middeck seats were less eroded and fragmented when compared to the flight deck seats. Material analysis revealed that the seat 6 and 7 components collected significant deposits of melted aluminum from the lithium hydroxide (LiOH) door to which they were attached. Unlike the flight deck seat components, analysis revealed no materials consistent with the pressure shell deposited on the middeck seats. The CAIB Report concluded,[5] and this report concurs, that seats 6 and 7 remained attached only to the LiOH door during a period of significant thermal exposure and material deposition. The highly directional nature of the deposition indicates that the LiOH door/seat 6/seat 7 combination attained a stable attitude (figure 3.1-11) during thermal exposure. The close proximity of debris recovery locations indicates that the seat 6 and 7 components separated from the LiOH door shortly before ground impact.

[5]*Columbia* Accident Investigation Board Report, Volume V, Appendix G.12, Crew Survivability Report, October 2003, p. 362.

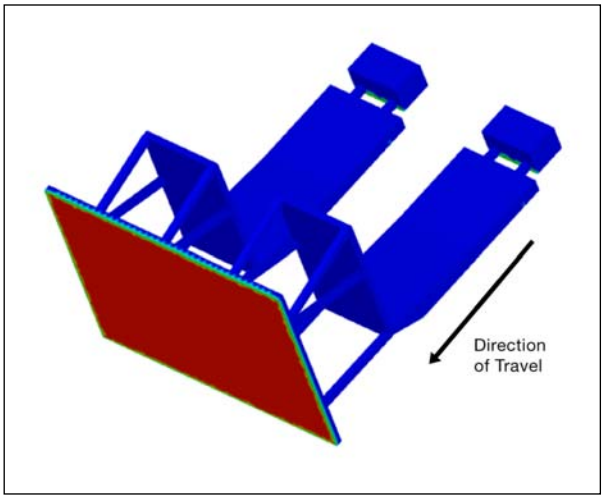


Figure 3.1-11. ***Stable attitude of the lithium hydroxide door/seat 6/seat 7 combination, as indicated by deposition.***

Of the seats, seat 6 had the highest percentage of structure recovered and identified; seats 5 and 7 had slightly fewer structural components that were identified. The flight deck seats had significantly lower percentages of identified structural components.

Nearly all seat fractures occurred at minimum thermal cross-sectional areas (minimum thermal mass), away from any large heat sink locations.[6] Common seat fracture locations are shown as red lines in figure 3.1-12. It is also noteworthy that nearly all thin-sheet aluminum materials (closeout panels on the seat pan and seatback) are missing (i.e., overloaded/melted away). Additionally, with the exception of the inertial reel straps, very little belt material and seat cushion material was recovered.

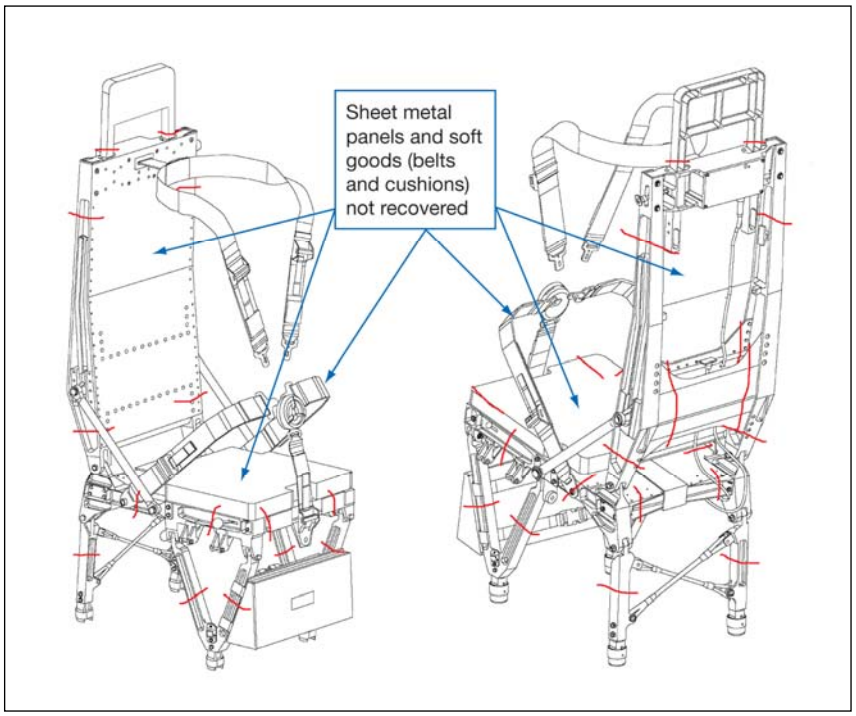


Figure 3.1-12. ***Common seat failure locations*** **(shown in red). [Adapted from the Space Shuttle Systems Handbook]**

Note: Mission Specialist seat shown; lower seat components on seats 1 and 2 are different.

[6]A heat sink is an area of the structure that has more material and takes longer to heat when exposed to elevated temperatures. Areas of minimal thermal cross-sectional area have less material and take less time to heat when exposed to elevated temperatures.

Examples of the fracture occurring at minimal thermal cross-sectional areas are evident on the seat leg failures for seats 6 and 7. Seats 6 and 7 are attached to a middeck floor panel that is the lid to the LiOH sub-floor storage compartment (figure 3.1-13). This panel, known as the "LiOH door," and the attached legs pieces are shown in figure 3.1-14.

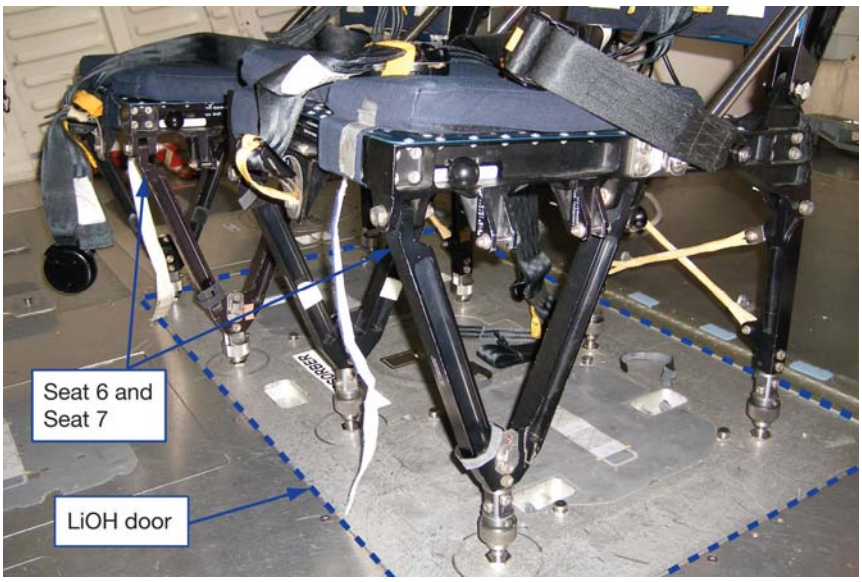


Figure 3.1-13. ***Example of an intact lithium hydroxide door with seats 6 and 7 attached*****. [Picture from the Crew Compartment Trainer]**

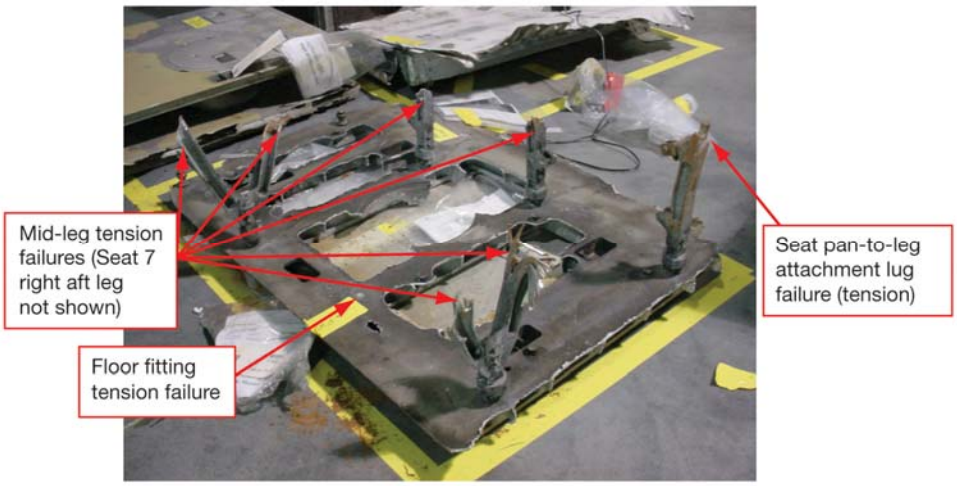


Figure 3.1-14. ***Recovered lithium hydroxide door with seats 6 and 7 legs attached*****.**

Seats 6 and 7 leg failures occurred with the legs in tension. The lowest strength margin (and, therefore, the expected failure point) at room temperature is at the leg attachment lug at the top of the leg. Analysis indicates that the leg attachment lug should fail at around 12,000 lbs. at room temperature, but only the left aft leg of seat 6 failed at the attachment lug (figure 3.1-14). Another expected leg failure point is the seat attachment floor fitting. At room temperature, the floor fitting should fail at around 21,400 lbs., but only the right forward leg from seat 6 failed at the floor fitting.

Five of the eight seat legs failed at mid-leg locations. The room temperature failure load for mid-leg fractures of the forward leg is 45,000 lbs. and the aft leg failure load is 24,000 lbs.

The seat 7 right forward leg floor fitting is present, but the corresponding seat 7 leg was not recovered; therefore, the failure location was not the floor fitting. Because this leg is very close to the edge of the LiOH door, it is possible that the failure occurred because the locking collar[7] was thermally damaged as the LiOH door/seat 6/seat 7 complex experienced entry heating.

Structural assessments were performed on the legs (which are made of 7075 aluminum, with a melting point between 890°F (477°C) and 1,175°F (635°C)) and the floor fitting (which is made of Inconel 718, with a melting point between 2,300°F (1,260°C) and 2,440°F (1,338°C)) to evaluate materials strengths with respect to temperature. As mentioned above, a force of 45,000 lbs. is required to cause a mid-leg tension fracture of the forward leg at room temperature. The required fracturing force decreases to 33,000 lbs. (~75% of room temperature strength) at 300°F (149°C), 21,000 lbs. (~45%) at 400°F (204°C), 9,000 lbs. (~20%) at 500°F (260°C), and 4,950 lbs. (~11%) at 600°F (316°C).

A force of 21,400 lbs. is required to fracture the floor fitting at room temperature. This force decreases to 20,300 lbs. (~95% of room temperature strength) at 300°F (149°C), 20,100 lbs. (~94%) at 400°F (204°C), and 19,900 lbs. (~93%) at 600°F (316°C). These values are plotted in figure 3.1-15.

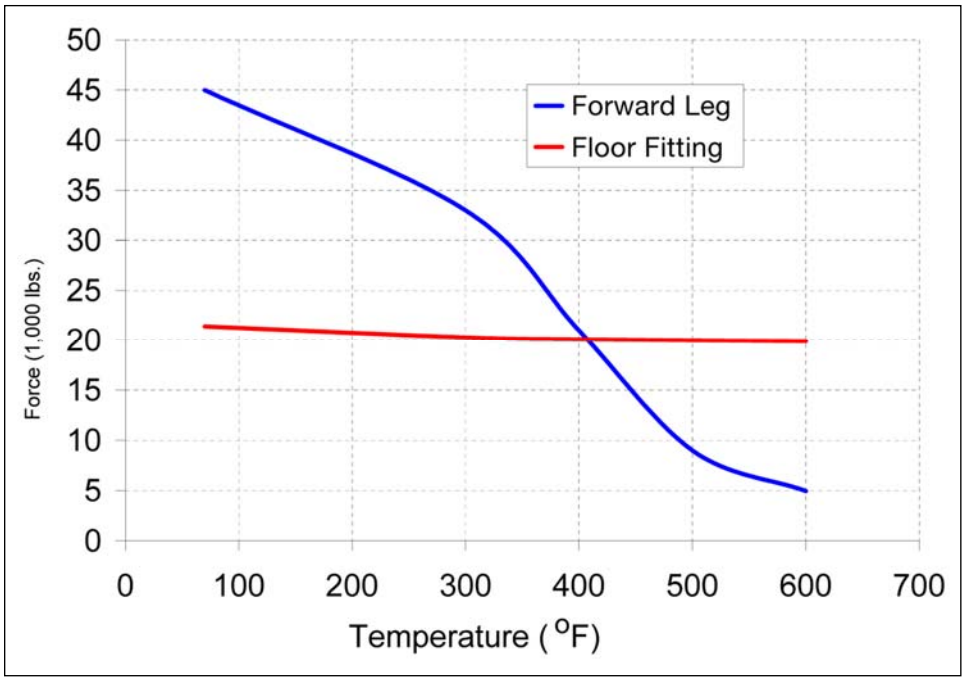


Figure 3.1-15. ***Failure force of seat forward leg and floor fitting vs. temperature.***

This plot reveals that heating greater than 400°F (204°C) is needed to weaken the materials such that the leg failure would occur at the mid-leg location. Otherwise, the forward legs should have failed at the floor fitting, which did not occur.

The nylon material used for the seatbelts will lose strength as the temperature increases above 250°F (121°C), and will melt at approximately 400°F (204°C). Therefore, the same heating event that caused material properties changes in the metallic components of the seat structure also resulted in a complete loss of the nylon seat restraints.

[7]The locking collar is used to lock the seat leg to the floor fitting.

Inspection of the seat structure fracture surfaces revealed a "delamination" fracture pattern that is consistent throughout the seat debris items. The most drastic of the fractures makes it appear almost as if the 7075 aluminum material is constructed of a laminate material (figure 3.1-16). This phenomenon is termed a "broom-straw" fracture.

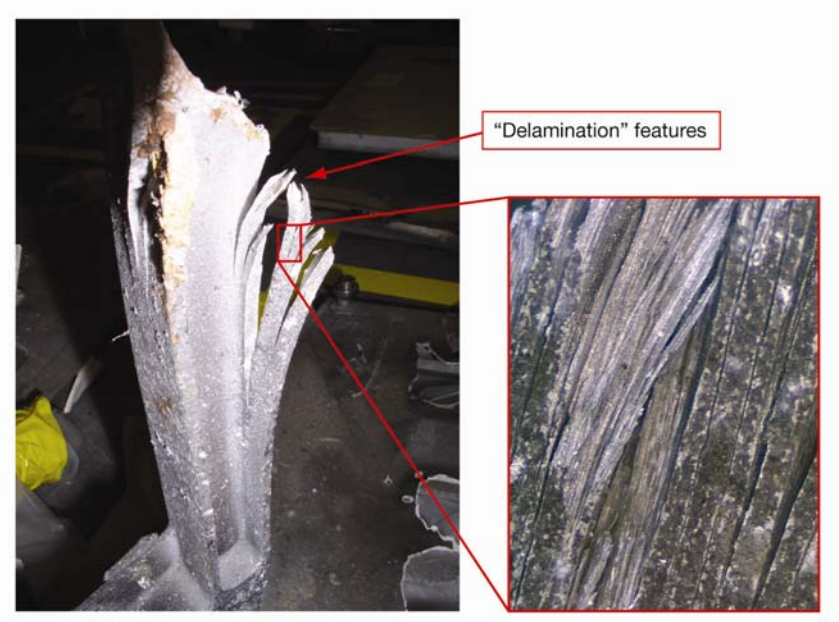


Figure 3.1-16. ***Example of a "broom-straw" fracture on a*** **Columbia** ***seat leg.***

Scanning electron microscope analysis of the seat broom-straw fracture surface cross sections revealed fully intergranular fractures and equiaxed grain[8] microstructure. This finding is consistent with exposure to elevated temperatures and high strain rates. Equiaxed grains were discovered both along and away from the crack surfaces (figure 3.1-17).

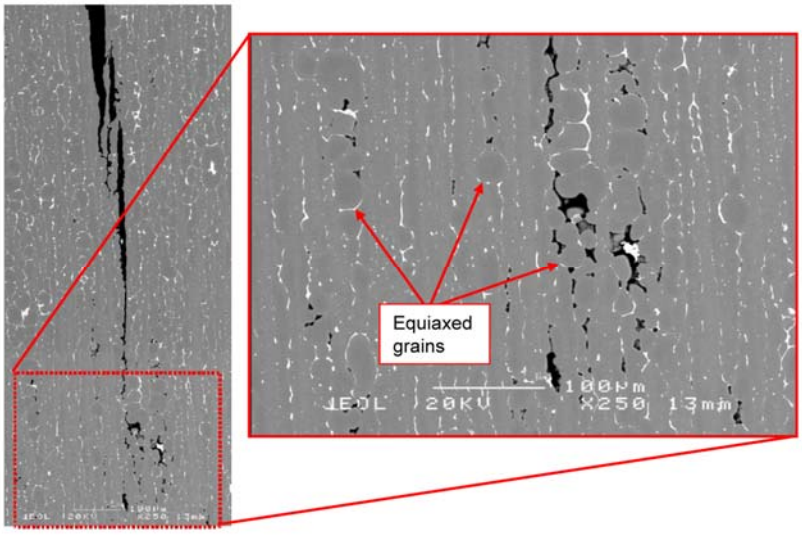


Figure 3.1-17. ***Scanning electron microscope image of broom-straw fracture surface cross sections.***

[8]A grain of approximately the same size in all three dimensions; characteristic of a recrystallized microstructure.

Metallurgical evaluation was completed in proximity to, and away from, the crack surfaces using energy dispersive X-ray spectroscopy. This revealed heavy grain boundary precipitation that is consistent with the aluminum alloy experiencing temperatures greater than 900°F (482°C).

All of these features are consistent with material that is exposed to high temperatures. Significant lack of ductility surrounding the fracture areas indicates that the failures occurred at relatively high strain rates.

In all cases, seat failure occurred as a result of thermal exposure (resulting in material property degradation) followed by mechanical overload. The additional thermal degradation on the flight deck seat components is accounted for by a longer period of heating as a result of the higher ballistic number associated with an intact, free-flying flight deck.

Analysis of the debris led the CAIB to conclusions[9] regarding the method of seat failure (i.e., thermal exposure followed by mechanical loading), but left one question unanswered: "Why did the seats fragment so much?" As was the case with much of the *Columbia* debris, many tiny debris impact craters and/or material deposits were found on the seat debris. However, very few witness marks (dents, scrapes, divots, etc.) were found, indicating little or no impacts with debris items larger than roughly 0.25 in. Therefore, debris-debris collisions are probably not a factor in the fragmentation of the seat structure. Because the failure mechanism involved heating of the structure to temperatures exceeding 400°F (204°C) (and probably exceeding 900°F (482°C)), the nylon seat restraint material was not present as the seat structure was breaking. This led the Spacecraft Crew Survival Integrated Investigation Team (SCSIIT) to conclude that the seat was unoccupied at the time it was breaking up. No other conclusion can be made because the thermal and aerodynamic mechanisms in the high-altitude, hypersonic flight regime are not well understood.

3.1.4 Upper seatbacks/inertial reels

Six out of the seven upper seatback items (all but seat 5) were recovered. Each item was positively identified to a seat position. Each recovered upper seatback debris item contained the inertial reel/recoil mechanism and some amount of strap material, which recoiled into the inertial reel housing following strap failure. This strap material amounted to the only significant nylon webbing recovered from each of the seatbelt restraint systems. Figure 3.1-18 shows the location of the upper seatback/inertial reel.

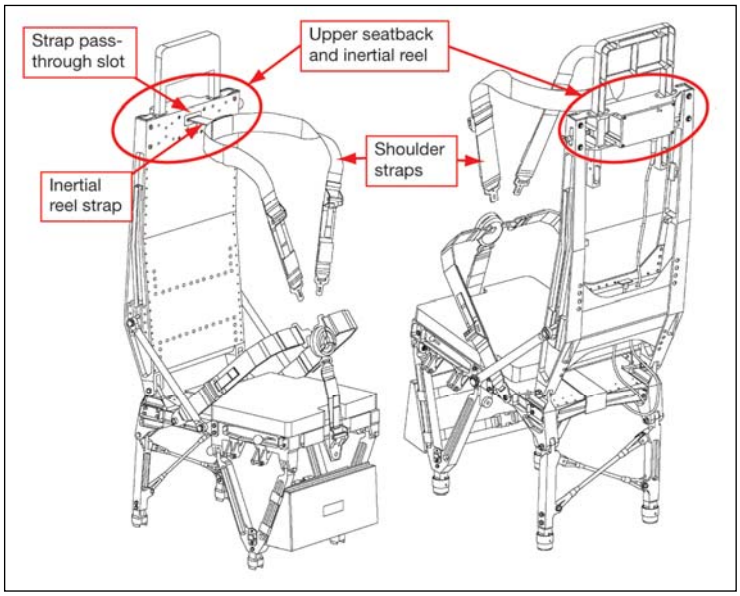


Figure 3.1-18. *Upper seatback debris item location*. [Adapted from the Space Shuttle Systems Handbook]

[9] *Columbia* Accident Investigation Board Report, Volume V, Appendix G.12, Crew Survivability Report, October 2003, p. 364.

As with the general seat failures that were described previously, all of the fractures at the upper seatback location occurred at the minimum thermal cross section, away from heat sinks on either side. Fracture surfaces did not exhibit melting or materials deposition, indicating that the upper seatback fractures occurred near the end of the period of heating and material deposition. Figure 3.1-19 shows all six recovered upper seatback items. Table 3.1-1 summarizes the upper seatback findings.

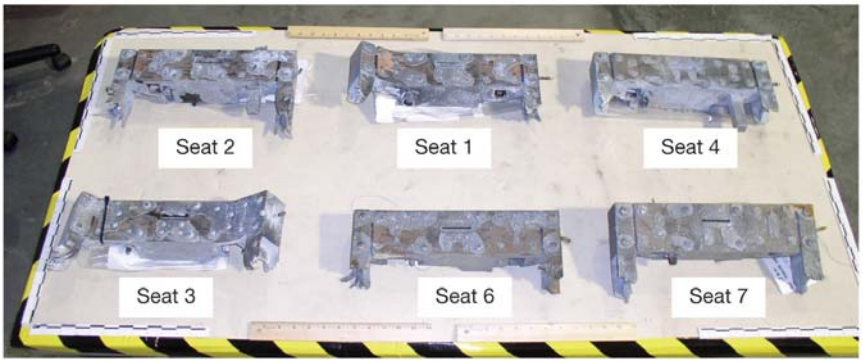


Figure 3.1-19. *Upper seatbacks* (front view).

Table 3.1-1. *Upper Seatback Comparisons*

Seat	Deformation	Fractures	Thermal Effects
Seat 1	Right frame member is bent forward, about the headrest bushing.	Tensile/bending fractures (with "broom-straw" features) are present on both the left and the right seatback frame members, just below the lower surface of the upper seatback/inertial reel assembly.	A moderate amount of splattered aluminum is present throughout, primarily deposited on the upper and lower surfaces (the upper surfaces more than the lower). No melting of fracture surfaces is noted.
Seat 2	Deformation is limited to the areas local to the fractures.	Tensile/bending fractures (with "broom-straw" features) are present on both the left and the right seatback frame members, just below the lower surface of the upper seatback/inertial reel assembly.	A moderate amount of splattered aluminum is present throughout, primarily deposited on the upper and lower surfaces (the upper surfaces more than the lower). No melting of fracture surfaces is noted.
Seat 3	Left and right frame members bent forward. Left side is bent about the headrest bushing; right side is bent just outboard of the bushing.	Tensile/bending fractures (with "broom-straw" features) are present on both the left and the right seatback frame members, just below the lower surface of the upper seatback/inertial reel assembly.	A large amount of splattered aluminum is present throughout, primarily deposited on the upper and lower surfaces. Noted the absence of deposited material on the portside bushing. No melting of fracture surfaces is noted.
Seat 4	Deformation is limited to the areas local to the fractures.	Tensile/bending fractures (with "broom-straw" features) are present on both the left and the right seatback frame members, just below the lower surface of the upper seatback/inertial reel assembly.	A large amount of splattered aluminum is present throughout, primarily deposited on the upper and lower surfaces. No melting of fracture surfaces is noted.
Seat 6	Slight deformation in the right frame member, bent about the headrest bushing.	Tensile/bending fractures (with "broom-straw" features) are present on both the left and the right seatback frame members, just below the lower surface of the upper seatback/inertial reel assembly.	A large amount of splattered aluminum material is deposited primarily on the lower surface. Noted a near complete absence of deposition on the upper surface. No melting of fracture surfaces is noted.

Table 3.1-1. *Upper Seatback Comparisons* (Continued)

Seat	Deformation	Fractures	Thermal Effects
Seat 7	Deformation is limited to the areas local to the fractures.	Tensile/bending fractures (with "broom-straw" features) are present on both the left and the right seatback frame members, just below the lower surface of the upper seatback/inertial reel assembly.	A large amount of splattered aluminum material is deposited on the lower surface. Noted a complete absence of deposition on the upper surface. No melting of fracture surfaces is noted.
Interpreta-tions		Seat fractures occur at minimum thermal cross section, away from heat sinks on either side.	Middeck items appear to have experienced significant splatter initiating from below. Flight deck items appear to have experienced significant splatter initiating from above and below. Lack of melting on the fracture surfaces indicates that the fractures occurred after the period of heating.
Conclusions	Observed deformations are affiliated with seat breakup fractures.	Fractures are consistent with general seat failure mechanism (thermal heating followed by mechanical overload).	Flight deck items experienced significant splattering initiating from above and below. Middeck items experienced significant splattering initiating from below *only*. Upper seatback fractures occurred after the period of heating.

For each seat, the inertial reel housing was removed from the upper seatback inertial reel cavity (figure 3.1-20). The cavity, housing, and mounting hardware were inspected for deformation, witness marks, debris impacts, and material deposition. Results for each seat position are generalized below.

Upper seatback (rear view). Inertial rear cavity (rear view of intact seat).

Figure 3.1-20. *Inertial reel mounting location*.

Only seat 3 showed deformation of the inertial reel housing mounting hardware. All four mounting bolts were bent slightly. One inertial reel housing mounting lug was broken with melted metal deposits on both fracture surfaces; i.e., the fracture surface of the lug and the fracture surface of the inertial reel housing (figure 3.1-21). The inertial reel mounting hardware for the other five recovered upper seatbacks exhibited no deformations.

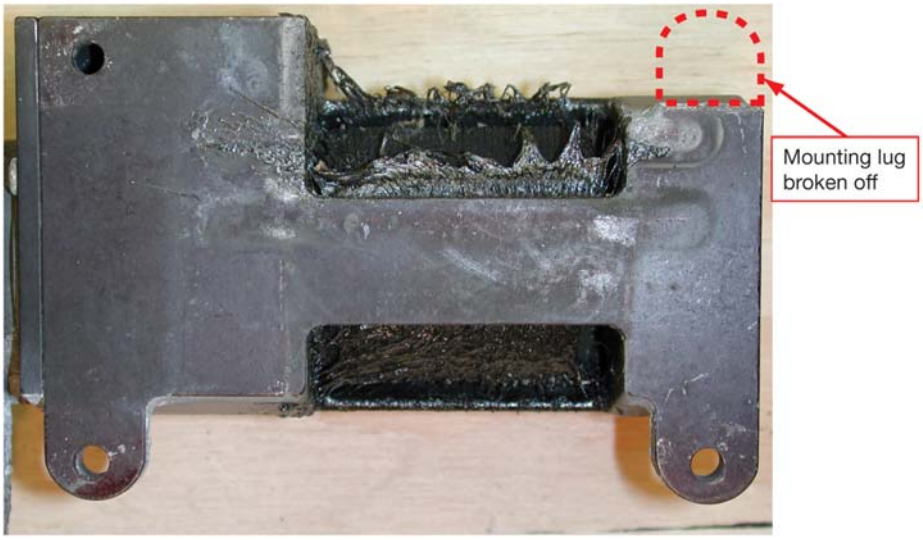


Figure 3.1-21. ***Seat 3 inertial reel, looking aft*****.**

In all cases, melted strap material (nylon) was discovered inside the upper seatback inertial reel cavity, including on the strap rollers (figure 3.1-22). Only seat 3 had melted strap material on the external surface of the upper seatback. In this instance, the melted strap material flowed out of and away from the inertial reel strap pass-through slot, indicating that the melted material originated from inside the upper seatback inertial reel cavity. The lack of melted strap material on the outside of the seatbacks indicates that the torso restraint system failure occurred at the inertial reel strap and the remaining inertial reel strap retracted completely into the seatback, leaving no material to be melted and deposited on the external surfaces of the seats.

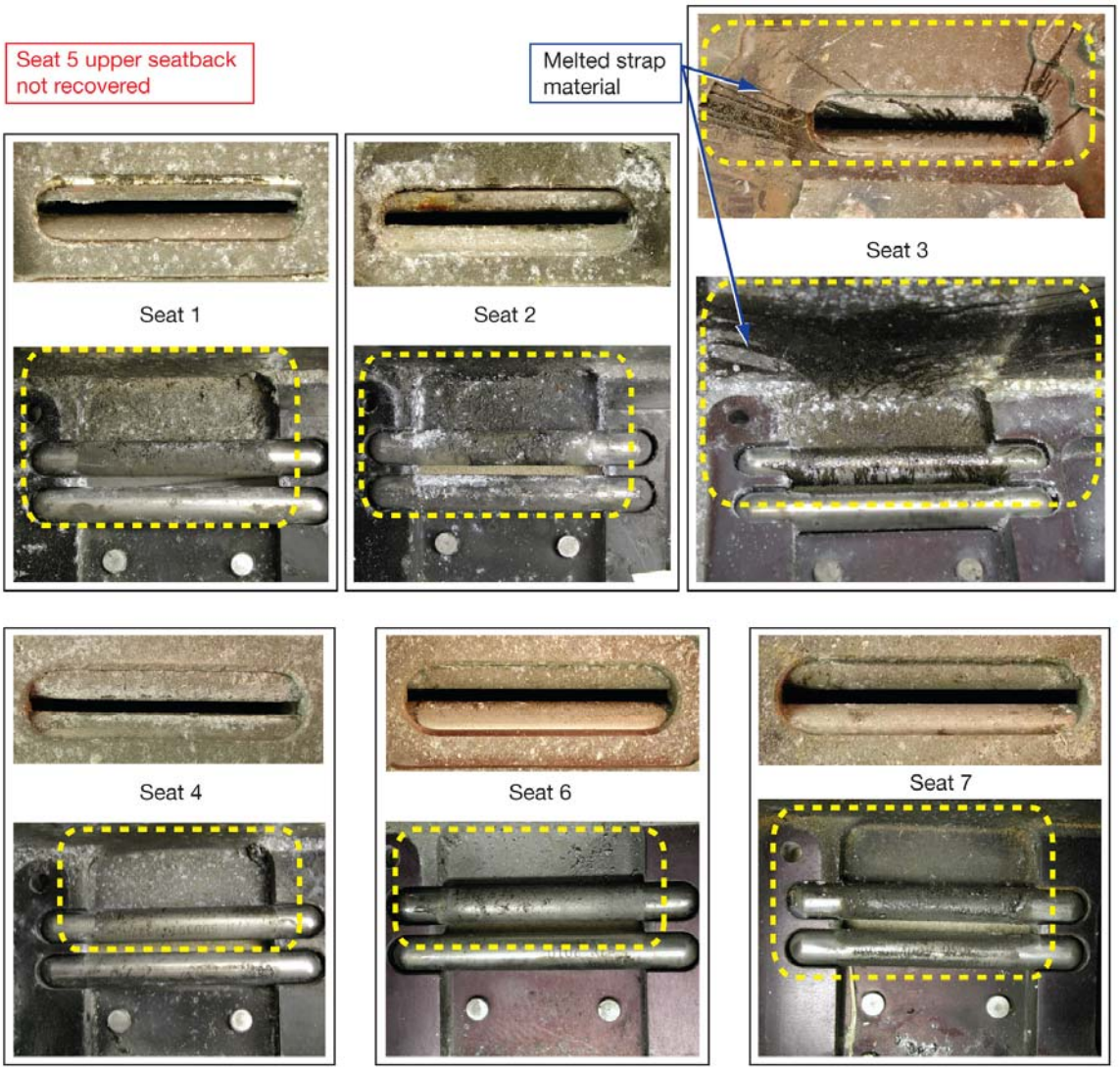
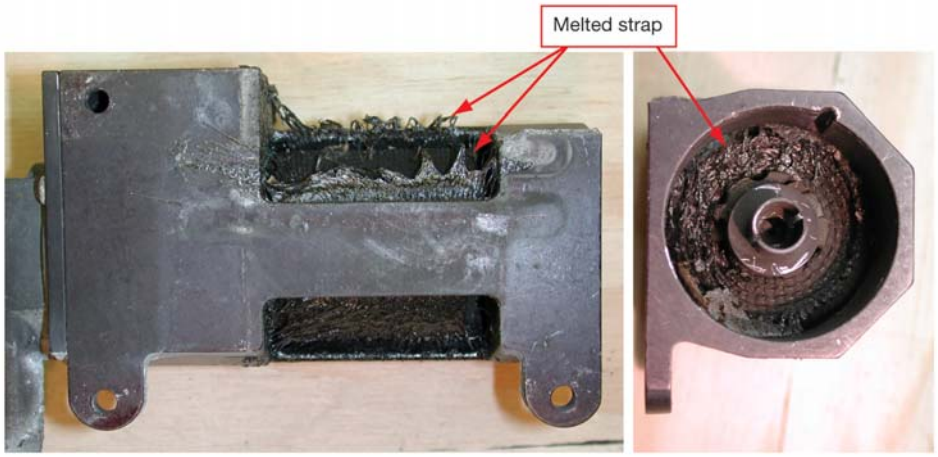


Figure 3.1-22. *Upper seatback inertial reel strap pass-through slots* – external (top)/internal (bottom); melted strap material areas outlined in yellow.

For all cases except seat 3, the melting and flow patterns of melted strap material are consistent with airflow entering the inertial reel lead-in strap pass-through slot in the upper seatback (forward-to-aft flow with respect to the seat). The melting and interior flow patterns of seat 3 are consistent with airflow entering the pass-through slot for the majority of the time that the seat was exposed to heating. Melted strap material on the exterior of the upper seatback indicates that airflow forced melted material out of the pass-through slot (aft-to-forward flow) for some period(s) of time. However, the radial flow pattern on the exterior surfaces indicates forward-to-aft airflow on the upper seatback once the melted material exited the slot. These melting and flow patterns could indicate that the upper seatback item was tumbling as it experienced heating.

In all cases, the inertial reel straps were melted only in areas that were immediately adjacent to the strap pass-through openings and along the lateral edges of the strap (figure 3.1-23). The strap material in other areas was not melted and appears normal.



(a) Front view. (b) Side view (end cap removed).

Figure 3.1-23. *Inertial reel melted strap material*.

3.1.5 Inertial reel straps

Each recovered inertial reel mechanism was disassembled to inspect the inertial reel strap and the inertial reel locking mechanism. This inspection revealed that strap failures occurred at various locations along the strap length.[10]

Five of the six recovered strap ends terminated along a straight line. The exception was the strap for seat 2, which terminated in a jagged line. Away from, but in close proximity to, the melted areas, the residual strap material remains flexible. This is consistent with melting that occurred after the strap recoiled into the housing (the straight lines correspond to where the straps were exposed at the pass-through slot). If the whole strap was exposed to significant heating/melting, a definite thermal gradient (varying degrees of melting) and some melting and "pulling" at the broken end of the strap would be seen. However, none of the straps exhibited a thermal gradient along the strap or melting and pulling at the broken end. The demarcation between melted and non-melted areas is very distinct (figure 3.1-24), indicating that the straps were protected inside the inertial reel housings during the period of high heating.

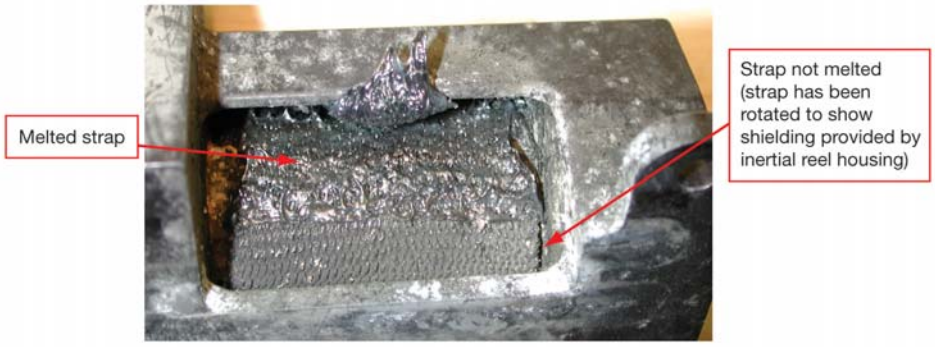


Figure 3.1-24. *Close-up of strap melt pattern*.

[10]The seats normally have 22 in. of inertial reel strap when measured from the reel to the shoulder harness y-split attachment.

The inertial reel straps failed primarily due to mechanical overload. Limited melting of the straps (and only in distinct areas that were unprotected by the inertial reel housing) indicates that melting was not a significant factor in the inertial reel strap failure. The mechanical overload may have been affected by elevated temperatures that weakened the straps, but these temperatures were not sufficient to cause obvious thermal damage to the strap. Significant strap melting occurred after the inertial reel straps failed and recoiled into the housing. Therefore, the inertial reel straps failed (and the *Columbia* crew members were, at most, partially restrained in their seats) prior to the end of the period of thermal exposure.

Metallic material was discovered on upper and lower strap surfaces when the straps were extended for inspection. Melted aluminum material was deposited on the straps for four of the six recovered inertial reels. The two exceptions were seats 1 and 2, both of which had strap failures at or near the inertial reel end of the straps. The deposition differed in character from the top-level depositions seen on the windows, which were more diffuse and uniform. Seat strap deposition consisted of globules or spheroids of metallic material (> 1/32 in.) that are widely scattered across the strap. Generally, there were no more than three or four deposited globules of metal per strap. Deposits on seats 4, 6, and 7 straps were close to the inertial reel end of the straps – areas of the straps that would not be exposed when the crew members were sitting upright in their seats. For material deposition to occur in these areas, the crew members had to be leaning to the side or leaning forward (or a combination of both), thereby extending the straps out of the inertial reel housing. The deposition of these material globules on the straps occurred before crew separation from the seats, yet after the CM pressure shell had been breached and the CM had been depressurized.

Evidence from the inertial reel straps indicates that the seats 1, 2, and 3 straps were mostly extended at the time of strap failure. The seats 4, 6, and 7 straps were extended during a material deposition period (seat 4 at least 8 in., or ~36% extended; seat 6 at least 21.25 in., or ~96% extended; and seat 7 at least 21.5 in., or ~98% extended). This indicates that the crew members were in seats 1 through 4 and seats 6 and 7 with at least the shoulder belts attached to the seatbelt buckle (the seat 5 upper seatback was not recovered). It is concluded that the inertial reels did not lock.[11]

Finding. Evidence from the inertial reel straps indicates that the seats 1, 2, and 3 straps were mostly extended at the time of strap failure. The seats 4, 6, and 7 straps were extended during a material deposition period (seat 4 at least 8 in., or ~36% extended; seat 6 at least 21.25 in., or ~96% extended; and seat 7 at least 21.5 in., or ~98% extended). Medical evidence (see Section 3.4) indicates that some of the crew members received injuries consistent with insufficient upper body restraint.

Conclusion L2-2. The seat inertial reels did not lock.

Conclusion L2-3. Lethal injuries resulted from inadequate upper body restraint and protection during rotational motion.

Recommendation L1-3/L5-1. Future spacecraft crew survival systems should not rely on manual activation to protect the crew.

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments.

[11]This conclusion is consistent with the entry simulation X-axis loads (discussed in Section 2.1.3 and shown in figure 2.1-16) remaining below the inertial reel auto-lock threshold of 1.78 G. Additionally, this conclusion is consistent with findings described in the Department of Defense Joint Service Specification Guide JSSG-2010-7, Crash Protection Handbook. The handbook describes a failure mode of MA-6 type inertial reels in which the reels failed to lock in crashes involving X-axis loads below the auto-lock threshold and subsequent Z-axis loads forcing the seat occupant down and forward, resulting in the occupant's torso being unrestrained during the crash dynamics. The handbook states that the "MA-6/MA-8 units were shown to be deficient in design and proven to be unreliable in survivable crash conditions." This precipitated a revision to MIL-R-8236, which is the military specification governing performance criteria for inertial reels used in military aircraft. The update to the military specification and the publication of the crash protection handbook occurred after the shuttle seats were designed.

Recommendation L2-8. The current shuttle inertial reels should be manually locked at the first sign of an off-nominal situation.

Indentations matching the linear ridges on the inertial reel spool were observed 0 to 4 in. from the strap attach location (figure 3.1-25). These linear indentations near the inertial reel, were present on all the straps (except the seat 1 strap, which failed next to the inertial reel attachment). Inertial reel straps for 11 other shuttle flight seats in inventory (not *Columbia* debris) and training seats were inspected for similar linear features. The inertial reel straps for all seats exhibited the same waves that were found on the *Columbia* inertial reel straps. Therefore, the waves are a result of strap stowage, not an indication of loading on the strap.

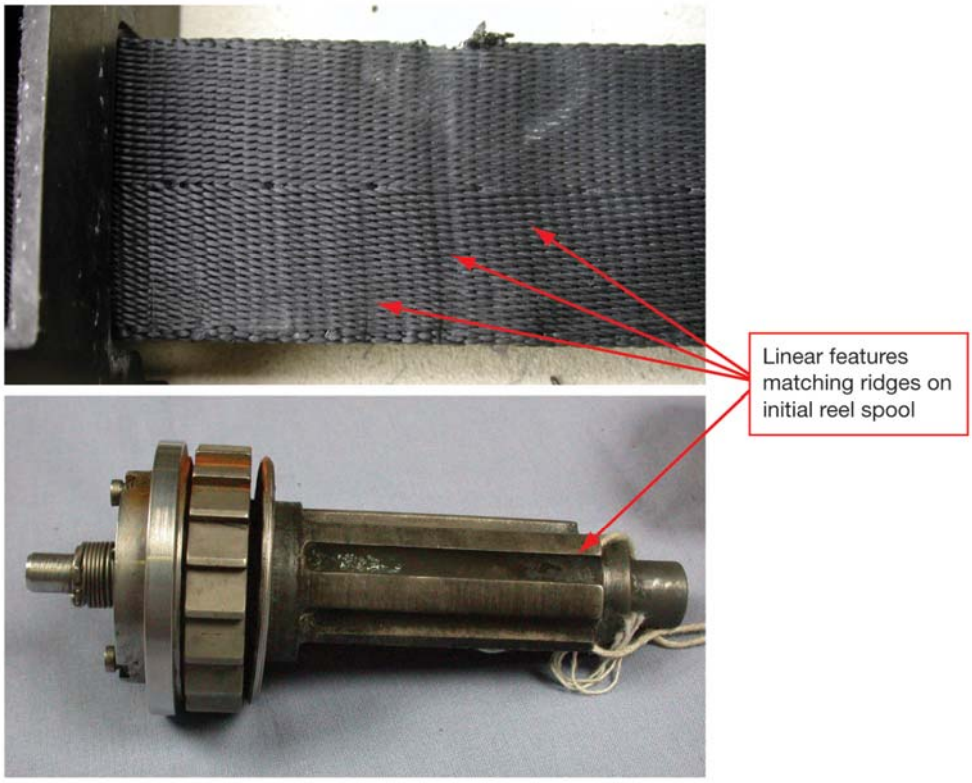


Figure 3.1-25. ***Example of linear waves on inertial reel straps*****.**

The inertial reel strap findings for each upper seatback are summarized below. Table 3.1-2 compares the inertial reel strap findings.

Table 3.1-2. ***Inertial Reel Strap Comparisons***

Seat	Strap Failure Location	Extent of Melting/Condition of Residual Strap Material
Seat 1	The strap failed at approximately 2% of the extended length, in proximity to the attachment to the inertial reel (< 1/2 in. away from the inertial reel, out of the original 22 in. of strap).	Residual strap material exists only at the end of the strap still attached to the inertial reel spool.
Seat 2	The strap failed at approximately 20% of the extended length, in proximity to the attachment to the inertial reel (~4.5–5 in. away from the inertial reel, out of the original 22 in. of strap). The strap failed at an approximately 45-degree angle, over a 1- to 2-in. length.	The strap melted only at the exposed strap pass-through areas and the lateral edges of the inertial reel housing. The strap material that was shielded by the inertial reel housing did not melt. This suggests that mechanical failure occurred before significant thermal exposure.
Seat 3	The strap failed at approximately 90% of the extended length, in proximity to the attachment to the shoulder harness y-split (~20 in. away from the inertial reel, out of the original 22 in. of strap). Strap failure appears to have occurred in close proximity to the shoulder split attachment, probably at the stitch stress concentration. Two raised sections at a slight angle (~20 degrees) from perpendicular to the strap axis, observed 2 to 4 in. from the strap attach location. Marks are similar to "bird-caging" features found in dynamic failures of cable/cord. Approximately 10 small metallic bits of debris, largest approximately 1/8 in. in diameter located approximately 16.5 to 18 in. away from reel attachment.	The strap melted only at the exposed strap pass-through areas and the lateral edges of the inertial reel housing. The strap is melted completely through one layer near the upper pass-through. The strap material that was shielded by the inertial reel housing is not melted. This suggests mechanical failure before significant thermal exposure.
Seat 4	The strap failed at approximately 80% of the extended length, in proximity to the attachment at the shoulder harness y-split (~18 in. away from the inertial reel, out of the original 22 in. of strap). One very small ball (< 1/16 in.) of melted metallic material was located approximately 14 in. from the reel attachment, on the lateral edge of the upper strap surface. (Note that the debris came off during inspection.)	The strap melted only at the exposed strap pass-through areas and the lateral edges of the inertial reel housing. The strap is melted completely through one layer near the upper pass-through. The strap material that was shielded by the inertial reel housing is not melted. This suggests mechanical failure before significant thermal exposure.
Seat 6	The strap failed at approximately 90% of the extended length, in proximity to the attachment at the shoulder harness y-split (~20 in. away from the inertial reel, out of the original 22 in. of strap). Strap failure appears to have occurred in close proximity to the shoulder split attachment, probably at the stitch stress concentration. Three pieces of metallic debris were deposited on the strap surfaces: 1. A small ball of melted material (< 1/32-in. dia.) deposited on the upper surface, located approximately 16 in. from the reel attachment. (Note that described debris came off during inspection.) 2. A small fragment of melted material (< 1/32-in. dia.) deposited on the lateral strap edge, located approximately 13 in. from the reel attachment. 3. A small fragment of melted material (< 1/32-in. dia.) deposited on the lateral strap edge, located approximately 1.75 in. from the reel attachment.	The strap melted only at the exposed strap pass-through areas and the lateral edges of the inertial reel housing. The strap is melted completely through one layer near the upper pass-through. The strap material that was shielded by the inertial reel housing is not melted. This suggests that mechanical failure occurred before significant thermal exposure.

Table 3.1-2. *Inertial Reel Strap Comparisons* (Continued)

Seat	Strap Failure Location	Extent of Melting/Condition of Residual Strap Material
Seat 7	The strap failed at approximately 50% of the extended length (~11 in. away from the inertial reel, out of the original 22 in. of strap). Three pieces of metallic debris were deposited on the strap: 1. One approximately 1/10-in. diameter, roughly spherical piece of metal debris is stuck to the edge (the right side of seat) of the lower surface of the strap, approximately 1.5 in. from the inertial reel. 2. One 0.5 in. × 0.3 in. roughly triangular (~0.05-in.-thick) piece of metal debris is stuck to the lower surface of the strap, approximately 8 in. from the inertial reel. 3. Several small, roughly spherical metal debris items are stuck to the upper side of the strap fragment, along the edge, approximately 2 in. from the end.	The strap melted only at the exposed strap pass-through areas and the lateral edges of the inertial reel housing. The strap is melted completely through one layer near the upper pass-through. The strap material that was shielded by the inertial reel housing is not melted. This suggests that mechanical failure occurred before significant thermal exposure.
Interpretations	Seats 1 and 2 straps failed at or near the end of the strap at the inertial reel (the belt was almost fully extended). The seat 3 strap shows evidence ("birdcage" witness marks near the inertial reel) suggesting that the belt was almost fully extended at the time of failure. The seat 4 strap shows evidence (melted material deposited approximately 14 in. from inertial reel) indicating that the belt was partially (~8 in. or ~36%) extended during the period of material deposition. The seat 6 strap shows evidence (melted material deposited approximately 1.75 in. from the inertial reel) indicating that the belt was mostly (~21.25 in. or ~96%) extended during the period of material deposition. The seat 7 strap shows evidence (melted material deposited approximately 1.5 in. from the inertial reel) indicating that the belt was mostly (~21.5 in. or ~97%) extended during the period of material deposition.	The straps exhibit areas of flexible strap material right next to areas of melted strap. The "border" between the areas is defined by the presence of the inertial reel housing shielding portions of the strap. If thermal exposure (significant strap melting) was a major factor in the failure of the strap, a thermal exposure gradient over some finite length should be present on the straps. The lack of a thermal exposure gradient, and the presence of evidence indicating drastic differences in thermal exposure, point to the strap failure mode as being primarily mechanical in nature.
Conclusions	The seats 1, 2, and 3 straps were mostly extended at the time of strap failure. All others straps failed at locations ranging from 50 to 90% of strap length (away from recoil attach point). The seats 4, 6, and 7 straps were extended during the material globule deposition period: The seat 4 strap was at least 8 in. extended; the seat 6 strap was at least 21.25 in. extended; and the seat 7 strap was at least 21.5 in. extended. The presence of melted metal globules deposited on several of the straps indicates that the mechanical overload of the straps occurred after exposure to a thermal environment resulting in globule deposition. This evidence confirms crew members were in seats 1, 2, 3, 4, 6, and 7 at the time of inertial reel strap failure, and the inertial reels did not lock.	In all cases, mechanical overload of the inertial reel strap occurred independent from significant thermal degradation. Note that the failure of the inertial reel strap may have been affected by material property degradation due to elevated temperatures.

Seat 1 experienced strap failure at the end of the strap next to the inertial reel, and seat 2 experienced strap failure near the end of the strap next to the inertial reel. The seat 1 strap failure (figure 3.1-26) occurred in a straight line along the strap attach point shear plane (the expected failure point if the strap is fully extended).

The seat 2 strap failed at an approximate 45-degree angle, over a 1- to 2-in. length, approximately 4.5 in. from the inertial reel (figure 3.1-26). The way in which the seat 2 belt failed is unique among the

six recovered inertial reel straps. This failure is similar to a failure mode known as strap "dumping."[12] It is theorized that loading of the inertial reel strap laterally against the pass-through slot (figure 3.1-27) resulted in damage to the strap and, eventually, strap failure.

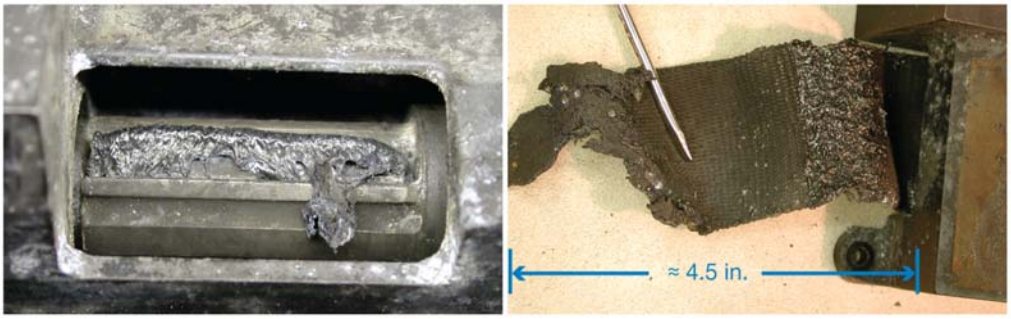


Figure 3.1-26. ***Seat strap failures: seat 1*** **(left)** ***and seat 2*** **(right).**

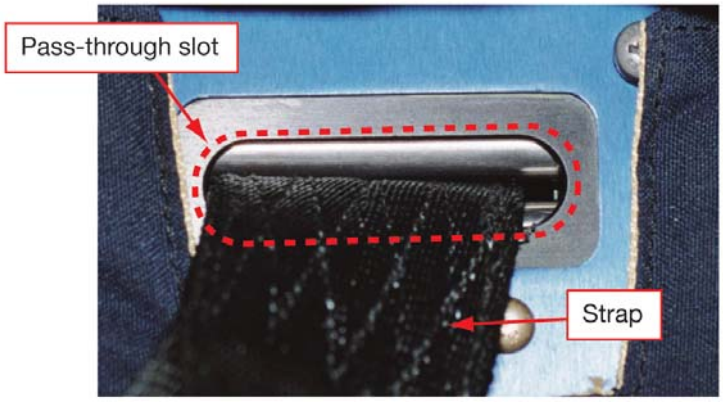


Figure 3.1-27. ***Seat strap pass-through slot (intact seat shown).*** **[Picture from a shuttle training mockup in the JSC Space Vehicle Mockup Facility]**

Finding. The seat 2 inertial reel strap exhibits "strap dumping" failure features. The strap failed progressively, possibly due to damage to the lateral edge of the strap from contact with the sharp edge of the strap pass-through slot.

> ***Recommendation L2-5.*** Incorporate features into the pass-through slots on the seats such that the slot will not damage the strap.

The seat 3 inertial reel strap (figure 3.1-28) failed at approximately 90% of the extended length (~20 in. away from the inertial reel). This strap failure occurred in close proximity to the shoulder harness y-split attachment, which is the expected failure point if a strap is not fully extended (the inertial reel strap will fail at around 5,200 to 5,300 lbs.). Wave features, which were observed approximately 2 to 4 in. from the strap attach location at a slight angle (~20 degrees) from perpendicular to the strap axis, were similar to the "bird-caging"[13] features found in dynamic failures of cable/cord. Because these birdcage features are close to the inertial reel end of the strap, the strap failure occurred with the strap mostly extended (retracted only 2 to 4 in.). Approximately 10 small metallic bits of debris, the larger of which are approximately 1/8 in. in diameter, were located approximately 16.5 to 18 in. away from the inertial reel attachment.

[12]Dumping is a strap failure mode caused by progressive strap failure that can be preceded by damage to the lateral edge of a strap.
[13]Describes the appearance of a multistranded rope or strap that has been subjected to compression or a sudden release of tension load. The outer strands are displaced outward, forming a cage-like appearance.

Figure 3.1-28. ***Seat 3 inertial reel strap*.**

The seat 4 inertial reel strap (figure 3.1-29) failed at approximately 80% of the extended length, which is in proximity to the attachment at the shoulder harness y-split (~18 in. away from the inertial reel). One very small ball of melted metallic material (< 1/16 in.) was located approximately 14 in. from the inertial reel attachment on the lateral edge of the upper strap surface (the debris came off during inspection.). No witness marks were observed along the entire strap length.

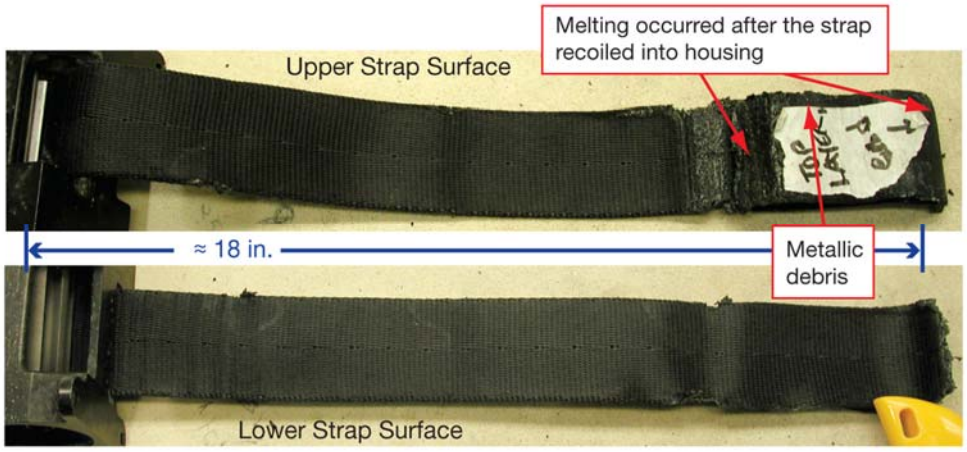


Figure 3.1-29. ***Seat 4 inertial reel strap*.**

The seat 6 inertial reel strap (figure 3.1-30) failed at approximately 90% of the extended length (~20 in. away from the inertial reel). Strap failure occurred in close proximity to the shoulder harness y-split attachment (the expected failure point if the strap was not fully extended). No witness marks were observed along the entire strap length. Three pieces of metallic debris were deposited on the strap surfaces: a small ball of melted material (< 1/32-in. dia.) deposited on the upper surface, approximately 16 in. from the reel

attachment (debris came off during inspection); a small fragment of melted material (< 1/32-in. dia.) deposited on the lateral strap edge located approximately 13 in. from the reel attachment; and a small fragment of melted material (< 1/32 in. dia.), which was deposited on the lateral strap edge located approximately 1.75 in. from the reel attachment. The strap was melted completely through one layer near the strap pass-through opening.

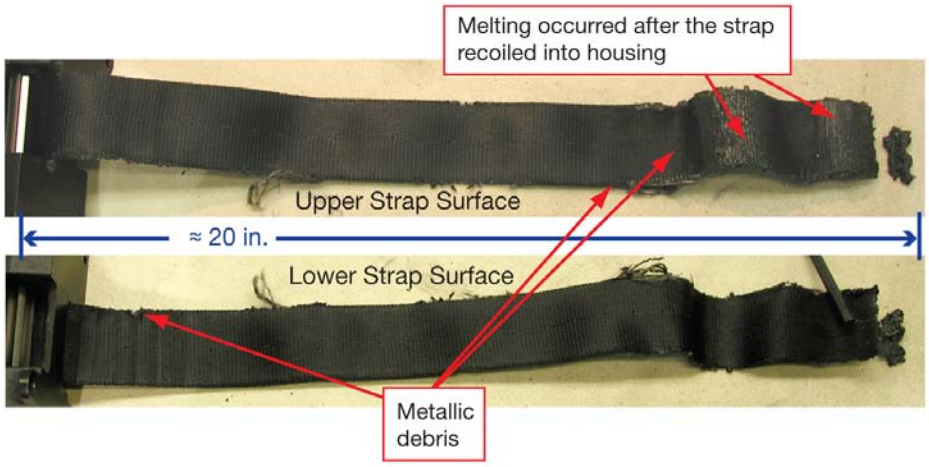


Figure 3.1-30. ***Seat 6 inertial reel strap*****.**

The seat 7 inertial reel strap (figure 3.1-31) failed at approximately 50% of its length (~11 in. away from the inertial reel). No witness marks were observed along the entire strap length. Three areas of metallic debris deposits were noted on the strap: one approximately 1/10-in.-diameter, roughly spherical piece of metal debris stuck to the edge (the right side of seat) of the lower surface of the strap, approximately 1.5 in. from the inertial reel; one 0.5-by-0.3-in. roughly triangular (~0.05-in.-thick) piece of metal debris stuck to the lower surface of the strap, approximately 8 in. from the inertial reel; and several small, roughly spherical metal debris items stuck to the upper side of the strap fragment, along the edge, approximately 2 in. from the end. Localized melting caused the strap to separate into two pieces after the strap had recoiled into the housing.

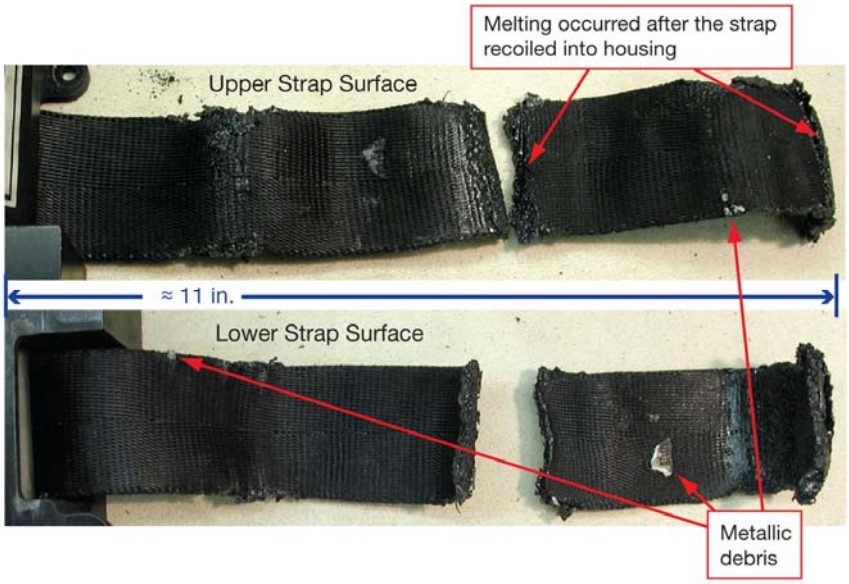


Figure 3.1-31. ***Seat 7 inertial reel strap*****.**

The inertial reel straps are certified to sustain at least 5,000 lbs. of load (at least 2,500 lbs. of load when the strap is fully extended). Vendor testing indicates that the straps fail at approximately 5,200 lbs. (or at ~3,500 lbs. if the strap is fully extended). These values are for straps tested statically at room temperature; the loads required to fail the straps at elevated temperatures are unknown. Additionally, the material properties (combustion vs. chemical degradation vs. melting) in a high-temperature/low-O_2/low-pressure environment are not known, neither are the strap properties in highly dynamic loading situations (high loads over very short time periods). Therefore, the inertial reel straps alone could not provide sufficient evidence for determining the loads at which the straps failed.

Finding. All inertial reel straps are tested with static loads at room temperature. Load testing has not been conducted to determine the loads required to fail the straps at elevated temperatures or under dynamic loads. Testing has not been conducted to determine the material properties (combustion vs. chemical degradation vs. melting) in a high-temperature/low-O_2/low-pressure environment.

> ***Recommendation L2-6.*** Perform dynamic testing of straps and testing of straps at elevated temperatures to determine load-carrying capabilities under these conditions. Perform testing of strap materials in high-temperature/low-oxygen/low-pressure environments to determine materials properties under these conditions.

3.1.6 Inertial reel locking mechanisms

After inspection of the inertial reel straps was completed, each of the inertial reel/recoil mechanisms was disassembled (figure 3.1-32).

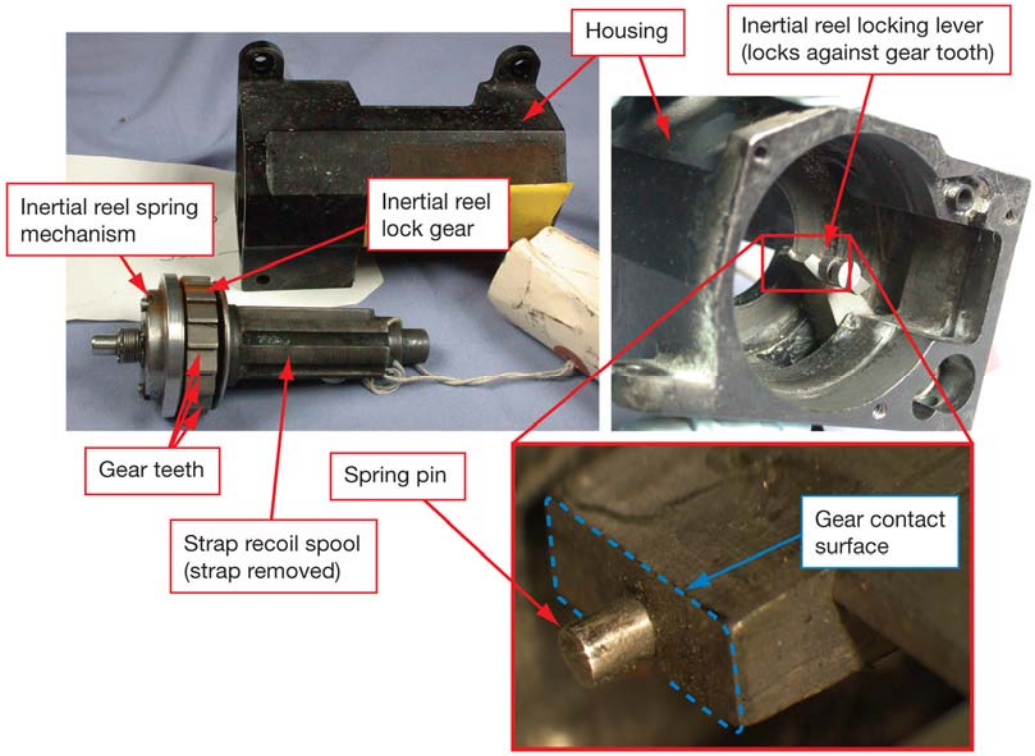


Figure 3.1-32. ***Inertial reel mechanism*****.**

The shuttle inertial reel can be locked manually; it also has an auto-locking feature that will lock with a strap acceleration of 1.78 G to 2 G (accelerations pulling the strap out at ~57 to 64 ft/sec^2). The automatic lock functions when the inertial reel spring mechanism engages the inertial reel locking lever against the corresponding inertial reel locking gear tooth surface. The strap is prevented from further extension, although recoil is possible. A spring pin is in the center of the locking lever contact surface.

All six of the recovered inertial reel mechanism gears and locking levers were inspected under stereomicroscope for evidence of mechanical loading, impact, adhesive wear, cracking, and plastic deformation. The results were consistent in all cases: the gear contact surface of the inertial reel locking lever showed no obvious witness marks; and only one tooth on *each* of the six mechanisms on the inertial reel locking gear showed evidence of witness marks (figure 3.1-33).

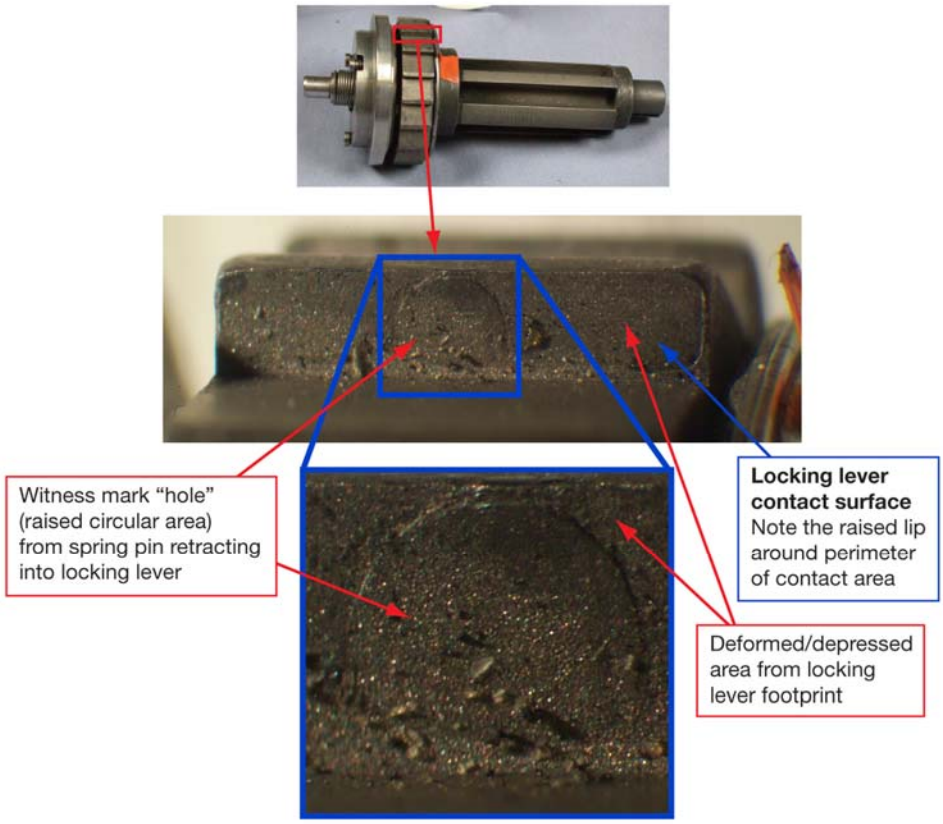


Figure 3.1-33. ***Inertial reel locking gear witness mark*.**

The witness marks can be explained by a significant loading event causing force translation through the functional strap restraint system and the inertial reel locking lever, and finally to the corresponding locking gear tooth surface. This force caused plastic deformation of the locking gear tooth, embossing the locking lever contact area on the gear tooth surface. The raised lip around the perimeter of the contact surface (figure 3.1-33) represents the outer edge of the locking lever contact area. The raised circular feature seen in this figure is caused by the spring pin retracting into the locking lever (below the contact surface of the locking lever). This feature is an area where the locking gear is *not* deformed by the locking lever.

It was initially thought that the witness marks were an indication of inertial reel strap loading during the accident. However, extensive investigation of inertial reels from training seats and other flown shuttle seats as well as a new inertial reel revealed identical circular witness marks (figure 3.1-34).

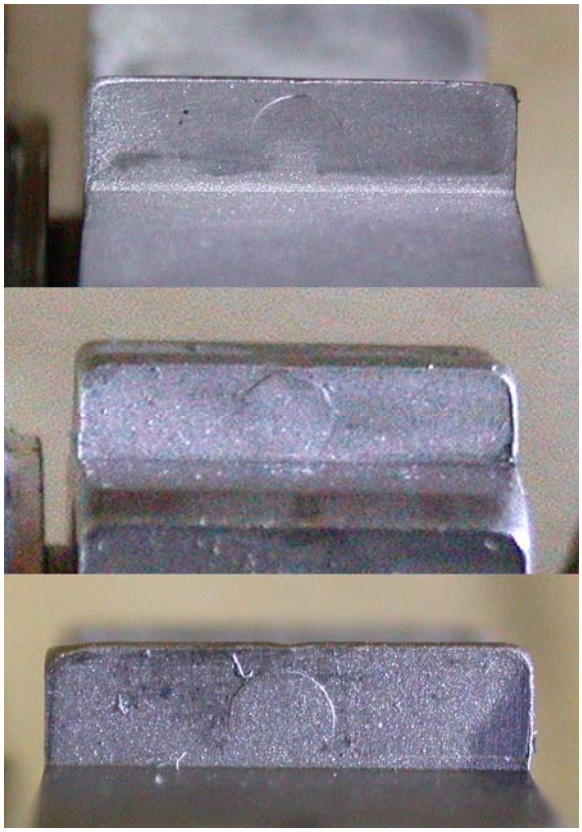


Figure 3.1-34. ***Circular witness marks on inertial reels from a training seat*** **(top),** ***a flight (non-*****Columbia*****) seat*** **(middle),** ***and a new (unused) flight-qualified inertial reel*** **(bottom).**

All inertial reels/straps of this type are proof-tested by the manufacturer with a 3,350-lb. static load prior to delivery to the customer. Because these inertial reels (and the *Columbia* inertial reels) were proof-tested, the investigation concluded that the witness marks on all of the inertial reels are a result of proof-testing. The absence of a second witness mark on the inertial reel led to the conclusion that the inertial reel straps failed at forces below the equivalent of a 3,350-lb. static load. However, it is probable that the straps failed under dynamic loading. As discussed above, the straps' load-carrying capabilities under dynamic loading are unknown.

A relatively large gap was discovered between the inertial locking mechanism and the lock gear on the seat 4 inertial reel (figure 3.1-35).

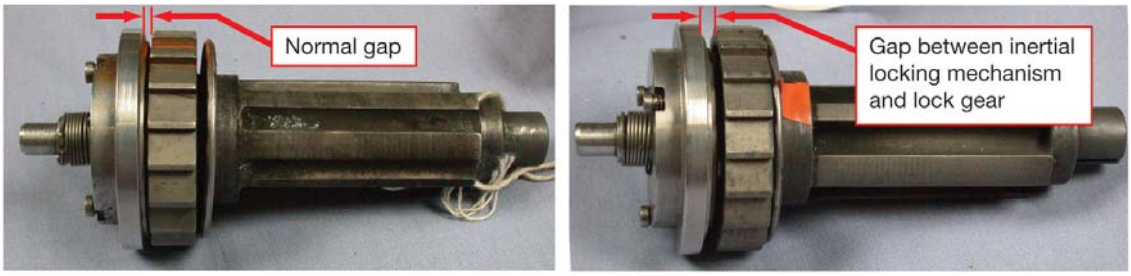


Figure 3.1-35. ***Separation gap on the seat 4 inertial reel*****.**

Upon further inspection, it was noted that the locking mechanism had permanently "jumped" out of the normal track position, creating a gap between the inertial mechanism and the locking gear. Because this configuration would not function nominally, it is concluded that this "jump" occurred during the accident.

The observed gap between the inertial reel locking mechanism and the locking gear on seat 4 is consistent with information in the manufacturer's experience base and occurs in crash events resulting in accelerations above 100 G at the strap input. However, the inertial reel locking mechanism can also be moved away from the locking gear by an impact along the axis of the inertial reel spool. Therefore, the observed failure (and resulting gap) could have been caused by impacts during the breakup dynamics, ground impact, or actual loads on the inertial reel strap. Because the cause of failure cannot be determined positively, no conclusions are possible regarding seat 4 inertial reel strap loads/accelerations.

3.1.7 Belt adjusters

Each seat restraint system includes five seatbelt adjusters (figure 3.1-36). Four of the seatbelt adjusters (out of the 35 that were on *Columbia*) were recovered. All of the belts are the same width and all of the adjusters are identical, so determination of the origin (which seat or which belt) was not possible. The findings are described below.

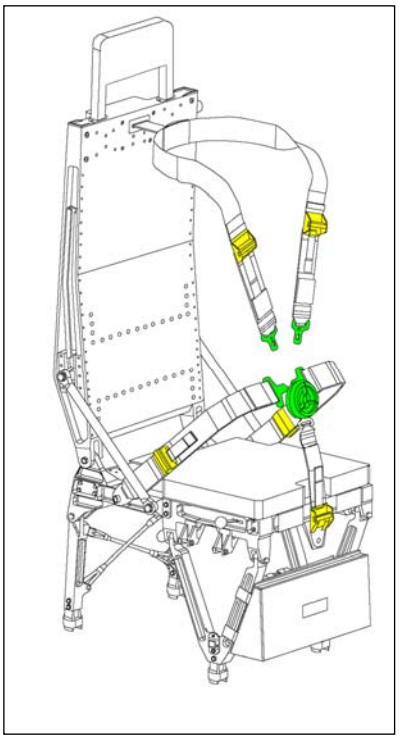


Figure 3.1-36. *Seat belt adjusters* (yellow) *and restraint buckle* (green). [Adapted from the Space Shuttle Systems Handbook]

The belt adjusters exhibited evidence of exposure to material deposition and heating. Molten metal debris impacts were observed in varying degrees (with no apparent directionality). Miniscule amounts of residual melted nylon belt material were observed on the adjusters.

Three of the four adjusters exhibited witness marks within the slider bar slot surface (figure 3.1-37). Tension on the belt will cause the slider bar to contact the slider bar slot as the bar rotates and slides within the slot. The witness marks in the adjusters are a result of significant loading on an intact restraint belt.

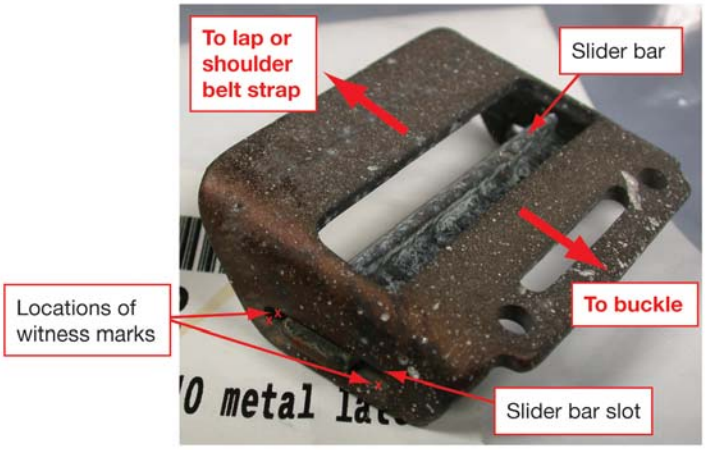


Figure 3.1-37. ***Recovered belt adjuster rear view.***

The fourth adjuster was missing the slider bar, and experienced fractures on both the left and the right slider bar slots. These fractures were apparently caused by the slider bar "blowing out" away from the body surface (figure 3.1-38).

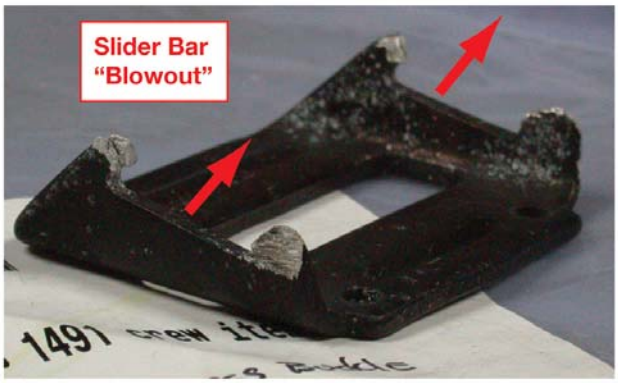


Figure 3.1-38. ***Fractured belt adjuster.***

The fracture surfaces exhibited delamination fractures (the "broom-straw" fractures described in section 3.1.3), indicating material property degradation due to elevated temperature exposure combined with high strain rate loading. The heating occurred quickly and allowed material degradation of the metallic belt adjuster to occur without compromising the material properties of the nylon restraint belt to the point that the belt could not transmit forces to the adjuster.

The slider bar "blowout" on one of the four strap adjusters was the result of significant loading on an intact restraint belt. It cannot be determined positively when this strap adjuster failure occurred.

The witness marks and the slider bar "blowout" fracture are indicative of significant loading events causing force translation through the intact belt restraint system, resulting in the impact of the slider bar against the corresponding slot surface within the belt adjuster housing.

3.1.8 Restraint buckle (figures 3.1-36 and 3.1-39)

One of the seven 5-point seatbelt buckles on board *Columbia* was recovered (figure 3.1-40). Positive identification to a seat position was not possible. Although the buckle experienced significant entry heating, resulting in substantial melting of the outer plastic housing, the structure remained intact with all five belt tongues still in place.

Figure 3.1-39. ***Demonstration of the five-point seatbelt buckle.***

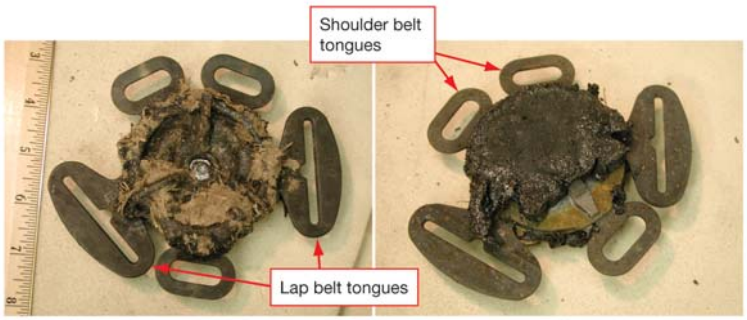


Figure 3.1-40. ***Recovered five-point seatbelt buckle*** **(front and back).**

The buckle assembly was disassembled for analysis. The two shoulder belt tongues and the two lap belt tongues were bent outward slightly (i.e., away from the crew member). Both of the shoulder belt tongue latching pins had very shallow linear features corresponding to the mating surfaces of the belt tongue (figure 3.1-41). These features may be witness marks as a result of dynamic loading events that were experienced in flight by the restraint system or of forces imparted to the five-point attach buckle during disassembly by the CAIB investigation team. A definitive conclusion cannot be made based on these witness marks alone. The other latching pins did not exhibit any noticeable marks or deformations.

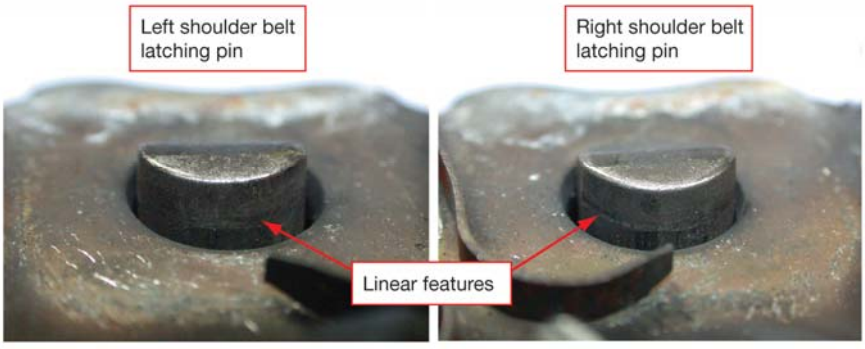


Figure 3.1-41. ***Indentations on latching pins, looking from center of buckle towards outer edge.***

Shadowing[14] on the belt tongues indicates that the belt tongues were in the position shown in figure 3.1-42(b). These positions, when compared to the positions of the tongues being straight out from center of buckle (radially, figure 3.1-42(a)), are: the shoulder belt tongues are moved toward the centerline (medially), the crotch belt tongue is moved toward the right side of the seat, and the lap belt tongues are moved down. It cannot be determined positively when the deposition (and shadowing) occurred, so no conclusions can be made relative to timing.

(a) Belt tongues positioned radially.

(b) Belt tongues positioned as indicated by shadowing.

Figure 3.1-42. *Belt tongue positions*.

Because all of the seats use a common seat restraint design, none of these pieces could be positively identified as being from a specific seat. This means that the belt adjusters and the restraint buckle could be from any of the seven seats on *Columbia*. It is even possible that the items were from the same seat. However, because the items vary in condition (different amounts of heating and debris impacts) and were found in widely spread locations (> a 14-mile spread), it is unlikely that these items originated from the same seat.

Significant forces caused witness marks on the seat restraint buckle tongues and the belt adjusters. Witness marks on the belt adjusters were caused by force transmission through intact functional belts. In the case of the fractured belt adjuster, forces on the belt were sufficient to cause the adjuster slider bar to "rupture" out of the slot. Fracture surfaces indicate that this rupture occurred at an elevated temperature. However, the heating occurred quickly, allowing material property degradation of the metallic belt adjuster without compromising material properties of the nylon restraint belt to the point that the belt could not transmit forces to the adjuster.

[14]An area that lacks or has less material deposition when compared to an adjacent area. Shadowing indicates that another item covered the shadowed area, preventing deposition.

3.1.9 Sequence

From the evidence and conclusions described above, the SCSIIT was able to develop the following sequence of events related to the seats:

- All of the seats were installed.
- Crew members were at least partially strapped into seats 1, 2, 3, 4, 6, and 7.
- During vehicle loss of control (LOC), the dynamics caused the crew members to be pulled forward and/or side-to-side. If the component of the acceleration pulling the inertial reel out of the seat (forward) was less than 1.78 G to 2 G, the inertial reels would *not* lock and the straps could be going in and out as the loads vary in magnitude and direction.
- Cabin breach and depressurization occurred.
- During the period of material deposition (when the cabin was depressurized and molten metal globules were floating around in the vicinity of the seats), the inertial reel straps were extended (the crew members were still in their seats) and the straps received molten material deposits.
- The inertial reel straps failed (predominantly due to mechanical overload, but below the equivalent of a 3,350-lb. static load) and the straps retracted into the seatbacks.
- The remaining shoulder harness belts and crotch and lap belts eventually melted/burned away.
- The seats experienced heating and high strain rates, and broke up due to thermal and then mechanical effects.

3.1.10 Lesson learned – equipment serialization and marking

One of the most useful tools in investigating an aviation accident is reconstructing the vehicle, either physically or virtually, from the recovered debris. Being able to identify the original location within the vehicle of debris items is of utmost importance in achieving an accurate reconstruction. Identifying the origins of debris items is made possible by serializing individual piece parts and subassemblies, and keeping accurate records of the piece part/subassembly serial numbers at the assembly and, ultimately, the vehicle level. This is especially useful when there are multiple units of identical or similar components, such as crew equipment, seats, engines, or structural members.

As discussed above, there are two different types of seats – Pilot seats, which are used by the CDR and the PLT, and Mission Specialists seats. The main difference between these types of seats is the design of the seat pan and the legs. The seatbacks and seat restraints are identical in design and construction for both types of seat. When the seats were manufactured, individual seat components were ink-stamped with individual part numbers and serial numbers. Configuration management records for seat components[15] were not accurately maintained, however, so identifying component locations by any surviving piece-part serial number was futile. The exceptions to this were the components associated with the inertial reels; all six recovered upper seatbacks were identified to specific seat locations.

Initially in the *Columbia* investigation, the only seat debris pieces that could be positively identified to a specific seat location were the upper seatbacks, which contain the inertial reels, and any pieces that remained attached to identifiable floor pieces. For the remaining seat debris items, reconstruction and location identification was a time-consuming, laborious process of matching pieces with the upper seatbacks and those pieces that were attached to floor panels. Eventually, 31 pieces of seat structure debris were positively identified to specific seat locations. However, almost 60 pieces of seat structure debris remained unidentified along with numerous fragments of seat soft goods. Had the individual seat components been permanently marked with serial numbers and those serial numbers tracked to the assembled seats, reconstruction and identification would have been much easier and a higher percentage of pieces could have been identified to

[15]Tracking the serial numbers for seat components to the top-level seat assembly's serial number.

specific seats. Therefore, space flight programs should develop failure investigation marking ("fingerprinting") requirements and policies. Equipment fingerprinting requires three aspects to be effective: component serialization, marking, and tracking to the assembly level. Marking by electronic means, metal stamping, or etching is preferable to labels or ink stamping because labels and ink stamps are not as durable in catastrophic failure scenarios. Marking in multiple locations, and on as many piece-parts in a major assembly as practical, is recommended.

Finding. While all seat piece-parts include serial numbers, only the serial numbers of the inertial reels were recorded and tracked to a specific seat assembly. The lack of configuration management documentation hindered the process of ascribing the seat debris items to specific seat locations.

> ***Recommendation A5.*** Develop equipment failure investigation marking ("fingerprinting") requirements and policies for space flight programs. Equipment fingerprinting requires three aspects to be effective: component serialization, marking, and tracking to the lowest assembly level practical.

3.2 Crew Worn Equipment

Because the crew worn equipment is the hardware that is closest to the crew members, it provides a source of data for the mechanical and thermal environments that the crew members experienced. This section provides background information on crew worn equipment and describes its configuration on STS-107. A brief review of the different types of shuttle suits is presented. Crew worn equipment, which includes the advanced crew escape suit (ACES), the personal parachute assembly (PPA), and the parachute harness, is described. The *Columbia* crew worn configuration is addressed. In addition, some aircraft in-flight breakup case studies are also considered to draw parallels between those mishaps and that of *Columbia*. Finally, *Columbia*-specific topics are addressed: the general, thermal, and mechanical conditions of the helmets and suit neck rings; the glove disconnects; the dual suit controllers (DSCs); the boots; the Emergency Oxygen System (EOS); the Seawater Activated Release System (SEAWARS); the Telonics Satellite Uplink Beacon-A (TSUB-A) search and rescue satellite-aided tracking (SARSAT) beacon; the Army/Navy personal radio communications (A/N PRC)-112 radio; and the ground plot analysis.

The following is a summary of findings, conclusions, and recommendations from this section.

Finding. The current ACES was added after the shuttle cockpit was designed and built. In many cases, the operations that the crew must perform are difficult to perform while wearing the suit. Some crew members choose between not wearing portions of the suit (gloves) to perform nominal tasks efficiently, or wearing their gloves to protect against off-nominal atmospheric situations at the expense of nominal operations or other off-nominal situation responses needing more dexterity.

Finding. Breathing 100% O_2 results in O_2-enriched air being exhaled into the shuttle cabin. Over time, this increases the O_2 concentration in the cabin, amplifying the potential for fire. Therefore, the amount of time that crew members have their visors down and are breathing 100% O_2 is limited operationally to reduce this hazard.

Finding. One crew member did not have the helmet donned at the time of the Crew Module Catastrophic Event (CMCE). Three of the seven crew members did not complete glove donning for entry. The deorbit preparation period of shuttle missions is so busy that crew members frequently do not have enough time to complete the deorbit preparation tasks (suit donning, seat ingress, strap-in, etc.) prior to the deorbit burn.

> ***Recommendation L1-2.*** Future spacecraft and crew survival systems should be designed such that the equipment and procedures provided to protect the crew in emergency situations are compatible with nominal operations. Future spacecraft vehicles, equipment, and mission timelines should be designed such that a suited crew member can perform all operations without compromising the configuration of the survival suit during critical phases of flight.

Finding. Inspection of all seven recovered helmets confirmed that none of the crew members lowered and locked their visors.

> ***Conclusion L5-1.*** The current parachute system requires manual action by a crew member to activate the opening sequence.

Recommendation L1-3/L5-1. Future spacecraft crew survival systems should not rely on manual activation to protect the crew.

Conclusion L4-1. Although the advanced crew escape suit (ACES) system is certified to operate at a maximum altitude of 100,000 feet and to survive exposure to a maximum velocity of 560 knots equivalent air speed, the actual maximum protection environment for the ACES is not known.

Recommendation L3-5/L4-1. Evaluate crew survival suits as an integrated system that includes boots, helmet, and other elements to determine the weak points, such as thermal, pressure, windblast, or chemical exposure. Once identified, alternatives should be explored to strengthen the weak areas. Materials with low resistance to chemicals, heat, and flames should not be used on equipment that is intended to protect the wearer from such hostile environments.

Finding. The current ACES helmets are nonconformal and do not provide adequate head protection or neck restraint for dynamic loading situations.

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments.

Recommendation L2-7. Design suit helmets with head protection as a functional requirement, not just as a portion of the pressure garment. Suits should incorporate conformal helmets with head and neck restraint devices, similar to helmet/head restraint techniques used in professional automobile racing.

Finding. Most of the suit components and subcomponents include serial numbers that are recorded and tracked to a specific crew member. This configuration management documentation aided greatly in the process of ascribing the debris items to specific crew members.

Recommendation A5. Develop equipment failure investigation marking ("fingerprinting") requirements and policies for space flight programs. Equipment fingerprinting requires three aspects to be effective: component serialization, marking, and tracking to the lowest assembly level practical.

Finding. The ACES had no performance requirements for occupant protection from elevated temperatures or fire. The ensemble includes nylon on the parachute harness straps and the boots. The ACES may not provide adequate protection to crew members in emergency egress scenarios involving exposure to heat and flames.

Recommendation L3-5/L4-1. Evaluate crew survival suits as an integrated system that includes boots, helmet, and other elements to determine the weak points, such as thermal, pressure, windblast, or chemical exposure. Once identified, alternatives should be explored to strengthen the weak areas. Materials with low resistance to chemicals, heat, and flames should not be used on equipment that is intended to protect the wearer from such hostile environments.

Crew escape equipment (CEE) enhances the crew members' capability to escape safely from a disabled orbiter on the launch pad, in atmospheric flight, or following landing.[1] This equipment includes the crew worn equipment and the crew escape pole.[2] For in-flight bailout scenarios, the CEE is designed for use during controlled subsonic gliding flight conditions.

[1]Program Requirements Document for Crew Escape Equipment, NSTS-22377, Revision B, October 1994.
[2]The crew escape pole is discussed in Section 2.4.

3.2.1 Shuttle suits

The shuttle was originally designed to be operated in a shirtsleeve (bare-hands) environment for all phases of flight, including launch and landing. The U.S. Air Force SR-71 pressure suit was worn for the first four shuttle missions (STS-1 through STS-4), which were considered test flights. Following these first four missions, the shuttle was declared fully operational and shuttle crews wore standard flight suits (light-weight fabric coveralls) with a helmet and a portable air supply that was intended principally for ground egress and cabin smoke protection. After the *Challenger* accident in January 1986, NASA started to use the launch and entry suit, beginning with STS-26 in 1988, and later phased in the more capable ACES beginning in 1994. This addition of pressure suits to a vehicle that was designed for shirtsleeve operations resulted in human/machine interface incompatibilities, especially with switches and other hand-operated controls. Because of this, some crew members decide between wearing gloves for full protection in emergency scenarios or not wearing gloves to be able to perform nominal tasks (keyboard entries, manipulating displays and controls, etc.) efficiently.

Finding. The current ACES was added after the shuttle cockpit was designed and built. In many cases, the operations that the crew must perform are difficult to perform while wearing the suit. Some crew members choose between not wearing portions of the suit (gloves) to perform nominal tasks efficiently, or wearing their gloves to protect against off-nominal atmospheric situations at the expense of nominal operations or other off-nominal situation responses needing more dexterity.

> ***Recommendation L1-2.*** Future spacecraft and crew survival systems should be designed such that the equipment and procedures provided to protect the crew in emergency situations are compatible with nominal operations. Future spacecraft vehicles, equipment, and mission timelines should be designed such that a suited crew member can perform all operations without compromising the configuration of the survival suit during critical phases of flight.

Crew worn equipment (figure 3.2-1), which is used by shuttle crew members during launch and entry, provides the necessary protection and survival equipment to sustain the crew below 100,000 feet and ensures

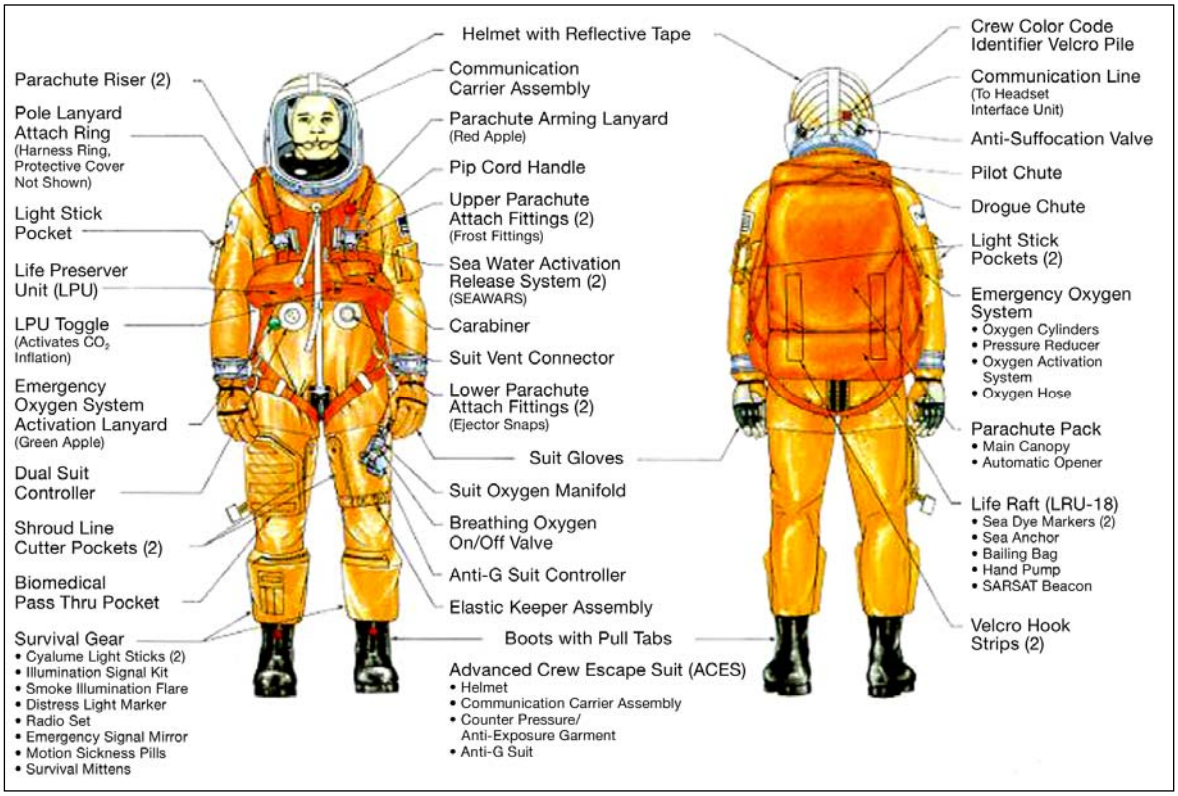


Figure 3.2-1. ***Crew worn equipment: advanced crew escape suit, harness, parachute, and survival gear.***

the crew's safety after vehicle egress. The crew worn equipment consists of the ACES and g-suit, a PPA, a parachute harness, and survival gear. The equipment provides the crew with altitude and atmospheric contamination protection during normal launch/entry operations and emergency conditions.

3.2.1.1 *Advanced crew escape suit*

The ACES is a full-pressure suit that is capable of applying static pressure over the entire body. A positive-pressure regulator delivers orbiter- or EOS-supplied 100% O_2 to the helmet at a pressure that is slightly above suit pressure. Breathing 100% O_2 results in O_2-enriched air being exhaled into the cabin. Over time, this increases the O_2 concentration in the cabin, amplifying the potential for fire. Therefore, the amount of time that crew members have their visors down and are breathing 100% O_2 is limited operationally to reduce this hazard.

Finding. Breathing 100% O_2 results in O_2-enriched air being exhaled into the shuttle cabin. Over time, this increases the O_2 concentration in the cabin, amplifying the potential for fire. Therefore, the amount of time that crew members have their visors down and are breathing 100% O_2 is limited operationally to reduce this hazard (see **Recommendation L1-2**).

The outer covering of the ACES is flame-resistant orange Nomex. Just beneath the outer layer of the Nomex fabric is a woven, open-link net restraint layer made of Nomex cord that provides structural support for the suit. Under the restraint layers is a pressure bladder made of nylon laminated to GORE-TEX® to wick body moisture away when unpressurized, while holding pressure when inflated. The ACES incorporates a rear entry pressure-sealing zipper for suit donning and doffing, a neck ring (figure 3.2-2) for helmet attachment, and wrist rings for glove attachment (figure 3.2-3).

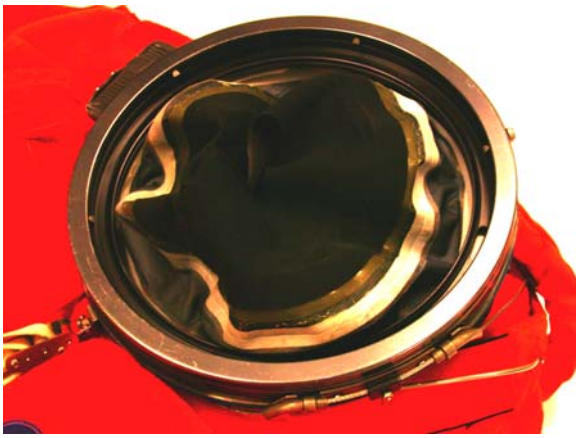


Figure 3.2-2. ***Example of neck ring*****.**

Figure 3.2-3. ***Example of wrist ring*****.**

The helmet (figure 3.2-4) attaches to the ACES neck ring. The neck ring has a latch (figure 3.2-5) that secures the helmet to the suit. Sliding the latch halves together moves six "latch dogs" to secure the helmet on the neck ring. Sliding them apart retracts the dogs, allowing removal of the helmet from the neck ring. Two independently rotating visors on the front of the helmet provide a dark sunshield and a clear pressure visor. The pressure visor is closed and locked by pulling the visor and the bailer bar down into the locked position. To open the pressure visor, a latch on the bailer bar lock must be pushed down and two buttons on either side of the lock must be pressed. This allows the bailer bar to unlock, after which the visor can be opened. O_2 is delivered to the helmet interior by a spray bar.[3] The shell of the helmet is made of fiberglass with a coating of reflective tape. The pressure visor is a laminate made of polycarbonate and polymethylmethacrylate.

[3]The spray bar is a tube with numerous small holes in it that direct O_2 towards the crew member's face.

The helmet must be attached to the neck ring and the pressure visor must be closed and locked to pressurize the suit.

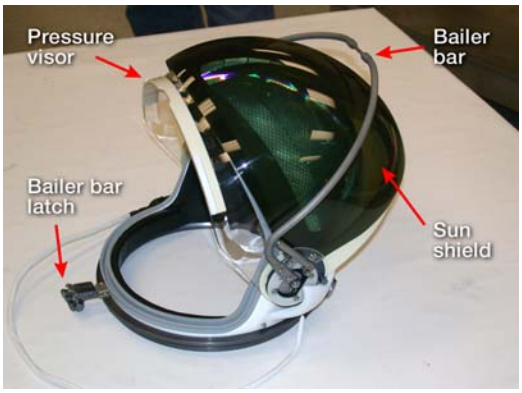


Figure 3.2-4. ***Example of helmet*****.**

(a) Open. (b) Closed.

Figure 3.2-5. ***Example of neck ring latch*****.**

The helmet provides an interface between the communication carrier assembly (CCA) and the orbiter communications system. The CCA, or "comm cap," contains microphones and earphones. The communications cable passes through the lower left side of the helmet and connects to a headset interface unit, which in turn connects to the orbiter communications system.

Detachable gloves (figure 3.2-6) attach to the ACES sleeve via mating rings and must be worn for the ACES to provide full protection. The rings provide an airtight seal and allow the gloves to swivel for improved mobility. The gloves have adjustable straps around the palm to prevent "ballooning" during suit pressurization and to allow for flexion at the palms.

Figure 3.2-6. ***Example of gloves*****.**

The ACES is worn in conjunction with Rocky 911 commercial off-the-shelf boots (worn over the pressure bladder) that have rubber soles and leather and nylon upper sections (figure 3.2-7).

Figure 3.2-7. ***Example of boots.***

Crew members wear a g-suit (figure 3.2-8) under the ACES during shuttle entry and landing. The g-suit bladders surround the abdomen, thighs, and calves and apply pressure to the crew member's lower abdomen and legs. The g-suit is used to counteract the effects of orthostatic intolerance upon return to 1-G conditions after exposure to microgravity. The g-suit is made from Nomex and nylon and has lacing to achieve a proper fit. It is pressurized with suit O_2. Pressure is controlled manually by the crew member. The g-suit connects to the ACES O_2 manifold via a quick disconnect (QD) hose.

Various garments are worn under the ACES and the g-suit for crew comfort. These garments include a liquid cooling garment (LCG), thermal underwear, wool socks, and a diaper. The LCG (figure 3.2-9) consists of thermal underwear shirt and pants with tubes sewn into the fabric on the inside. A cooling unit (thermal electric liquid cooling unit (TELCU) or individual cooling unit (ICU)) cools and pumps water through the LCG's network of tubing to cool the crew member. The water supply and return lines are fed through a plug located on the right thigh area of the ACES.

Figure 3.2-8. ***Example of g-suit.***

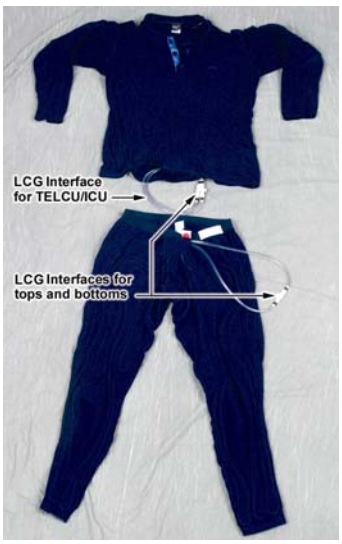


Figure 3.2-9. ***Example of liquid cooling garment.***

One AN/PRC-112 radio (figure 3.2-10) is flown per crew member and is located in a survival gear pouch inside a pocket on the right shin area of the ACES. The PRC-112 radio is constructed of an outer aluminum casing, plastic external switches, a plastic external battery pack, and various internal electronic components.

Figure 3.2-10. ***Example of Army/Navy personal radio communications-112 radio.***

3.2.1.2 *Parachute harness*

The parachute harness (figure 3.2-11) is a system of interwoven nylon webbing straps that provides an interface between the PPA and the crew member. The straps provide body support for crew members during bailout, emergency egress, and water rescue operations. Integrated into the harness are a carabineer, an EOS, an emergency water pack, and a life preserver unit (LPU).

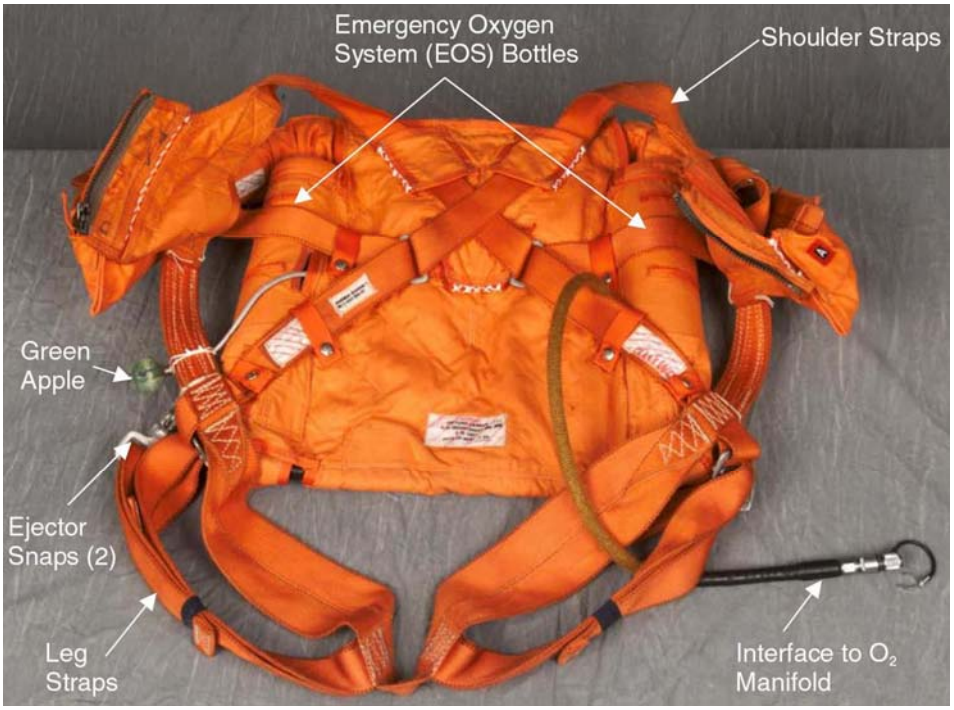


Figure 3.2-11. ***Example of parachute harness.***

The EOS (figure 3.2-12), which is located within the parachute harness, consists of two bottles that are pressurized with O_2 at 3,000 pounds per square inch (psi). Each bottle has a pressure regulator that reduces the pressure down to 70 psi. A common manifold delivers the 70-psi O_2 from both bottles to the O_2 hose that connects to the ACES O_2 manifold via a QD. The system, activated by pulling the "green apple" activation knob on the right side of the harness, provides 381 liters of O_2.

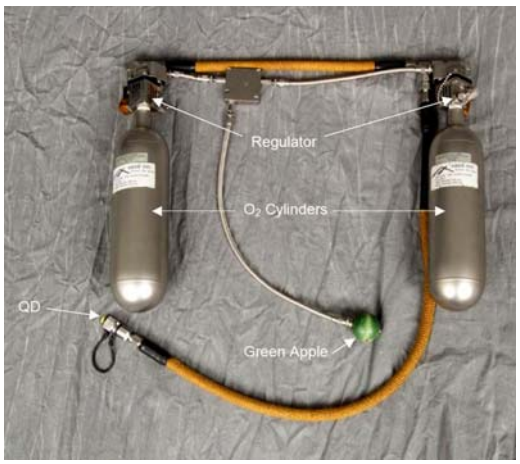


Figure 3.2-12. ***Example of Emergency Oxygen System*.**

3.2.1.3 *Personal parachute assembly*

All crew members wear a PPA (figure 3.2-13). It is secured to the crew member's parachute harness at four locations. The parachute risers are connected to two attachment points on the harness which are called Frost fittings, and the metal triangular rings are secured to the ejector snaps on the harness. Two SEAWARS, one on each parachute riser, are part of the fittings on the PPA risers that interface to the harness. These two SEAWARS are designed to automatically release the risers from the harness upon immersion in saltwater. The SEAWARS consist of an outer aluminum casing, plastic external components, various internal electronic components, and a small pyrotechnic device.

Figure 3.2-13. ***Example of personal parachute assembly*.**

The outer covering of the parachute pack is made of Nomex. The parachute pack has three compartments: an upper compartment that contains a pilot chute and drogue chute, a middle compartment that contains the main parachute and the automatic opening device (AOD), and a lower compartment that contains a one-person life raft. The right riser houses the D-ring bridle, which connects to a lanyard hook on the escape pole for use during in-flight bailout. The D-ring bridle and lanyard hook initiate the parachute opening

sequence. The left riser has a manual rip cord, which can be used to initiate the parachute opening sequence. Both methods require some crew action (either attaching the D-ring to the escape pole lanyard or pulling the rip cord) to initiate the parachute sequence.

Conclusion L5-1. The current parachute system requires manual action by a crew member to activate the opening sequence.

Recommendation L1-3/L5-1. Future spacecraft crew survival systems should not rely on manual activation to protect the crew.

The main parachute is a circular canopy made from nylon with Kevlar reinforcement.

The life raft is rubber and is inflated by two carbon dioxide (CO_2) bottles. A TSUB-A SARSAT beacon (figure 3.2-14) is secured to the life raft and activates automatically upon main parachute deployment. The SARSAT beacon consists of an outer aluminum casing, plastic external switches, and various internal electronic components.

Figure 3.2-14. ***Example of search and rescue satellite-aided tracking beacon.***

3.2.2 Crew worn equipment configuration

Recovered videotape from *Columbia* revealed information related to the configuration of the crew worn survival equipment, including the helmets and the ACESs. The recovered middeck video shows the seat 5 crew member suited (except for the helmet and gloves) and the seat 1 and seat 6 crew members donning their suits. The middeck video, which does not include views of other flight deck crew members donning their suits, ends prior to the seat 7 crew member donning the ACES.

Recovered flight deck video shows the flight deck crew members suited with helmets and gloves on (except one crew member, who had not completed donning gloves by the end of the video). This video also shows the helmet and helmet neck rings in close proximity to the crew members' chins. Because the helmets appeared to be restrained, investigators concluded that the crew members had the proper tension on the neck ring tie-down straps.

Although all crew members were wearing the main portion of the suit at the time of the accident, at some point the suits completely failed and separated. The SCSIIT investigated similar cases to understand the mechanism of suit failure.

3.2.3 Aircraft in-flight breakup case studies

The following civil aviation accidents provide examples of cases of passenger clothing being shed (body denuding) during in-flight breakups:

- Air India Flight 182 was flying at 31,000 feet over the Atlantic Ocean on June 23, 1985 when a terrorist bomb exploded in the baggage compartment. The Boeing 747 aircraft broke up in flight, and at least 21 of the 131 recovered bodies were denuded.
- Iran Air Flight 655 was mistakenly shot down by a U.S. Navy ship on July 3, 1988 while flying over the Persian Gulf. After missiles hit it, the Airbus A300 aircraft broke up in flight at an altitude of 13,500 feet. The denuded bodies of the passengers were recovered from the Persian Gulf waters.

- Pan Am Flight 103 was blown up by a terrorist bomb over Lockerbie, Scotland on December 21, 1988. The bomb went off when the Boeing 747 aircraft was at roughly 31,000 feet and 313 knots airspeed; numerous passengers who had separated from the aircraft prior to ground impact were denuded.
- COPA Flight 201 broke up over the jungle in Panama on June 6, 1992. The Boeing 737 aircraft broke up at approximately 13,000 feet while in a high-speed dive (the pilots entered the dive because of a faulty attitude indication that was due to a wiring problem). Many of the passengers' bodies were denuded.

These aviation accidents involved lower altitudes and slower speeds than those associated with the *Columbia* accident. However, a military accident more similar to the *Columbia* accident occurred on January 25, 1966 involving an SR-71 test flight.[4] The pilot lost control of the aircraft, and the SR-71 broke up while flying at approximately Mach 3 at over 75,000 feet (~400 knots equivalent airspeed (KEAS)). The pilot survived, but the reconnaissance systems officer was killed. Points of similarity to the *Columbia* accident include:

- The SR-71 accident occurred at high speed and relatively high altitude.
- The SR-71 aircraft breakup dynamics resulted in a fatality.
- The SR-71 aircraft breakup dynamics included crew member separation from the vehicle.
- The seat restraint straps failed.
- The SR-71 pressure suit is very similar to the shuttle ACES in design and construction.
- The dynamic pressure at the *Columbia* CMCE was roughly 405 pounds per square foot (psf) and the dynamic pressure at SR-71 aircraft breakup was roughly 398 psf, a difference of less than 2%.

However, there are also notable differences between the two accidents. These include:

- The SR-71 pilot's suit pressurized automatically (as designed) when the cockpit depressurized due to the aircraft breakup. The pilot attributed his survival to the pressurized suit, which protected him from the low-pressure/low-O_2 environment as well as the aerodynamic forces (windblast) that he experienced when he separated from the aircraft. As discussed in the sections below, the *Columbia* suits did not pressurize because the crew members did not lower visors or activate the suit O_2 system. Additionally, three crew members did not complete donning gloves, which is required for the suit to pressurize.
- While the *Columbia* crew members were exposed to a similar dynamic pressure environment as the SR-71 crew members, the thermal environment of the *Columbia* accident was much more severe than that experienced during the SR-71 breakup.
- Because of the altitude differences, the chemical environment (higher concentration of more reactive monatomic oxygen) of the *Columbia* accident differs from that of the SR-71 breakup.
- The *Columbia* suits did not remain intact, whereas the SR-71 pressure suits did remain intact.

Aerodynamic analysis indicates that the equivalent airspeed of the CM at the CMCE (GMT 14:00:53) was roughly 400 KEAS, and that it increased to 560 KEAS by the time of Total Dispersal (TD) (GMT 14:01:10). The ACES is designed to maintain structural integrity and pressure response capability when exposed to at least a 560-KEAS windblast. Since the suit is certified by NASA to meet this requirement based on its similarity to the pressure suit used by the U.S. Air Force, it was not subjected to windblast tests for certification. By contrast, the U.S. Air Force suit was tested in a certification program in 1990 during which it was exposed to a 600-KEAS windblast (the suit was worn by a manikin that was restrained in an ejection seat with the helmet visor down and locked). During the first test (suit not pressurized), the helmet sun shield separated from the helmet and a life preserver unit inflation tube separated from the life preserver unit. During the second test (the suit was pressurized to 2.99 psi), both shin pockets (survival gear storage pockets) were forced open. No other relevant anomalies were observed.

In the U.S. Air Force windblast test configuration, the helmet visors were lowered, which is notably different from the position of the *Columbia* visors. Debris evidence indicates that the *Columbia* helmet

[4] *Aviation Week & Space Technology*, August 8, 2005, pp. 60–62.

visors were up. With the helmet visor up, the helmet cavity presents a high drag configuration that could contribute to a mechanical failure of the suit/helmet interface, leading to suit disruption.

Standard materials testing data exist for the suit materials (GORE-TEX®, Nomex, nylon, etc.). However, the data are for tests conducted at "normal" environmental conditions (sea-level atmospheric temperature, pressure, and composition). Little laboratory test data exist on the performance of the materials in extreme environments. The lack of laboratory data presents an information gap regarding how the materials properties of the ACES are affected by exposure to the thermal and chemical environments at the altitudes and speeds experienced by *Columbia.* Although the more severe thermal and chemical environment of the *Columbia* accident may have weakened the suit materials, hastening suit disruption, the extent to which the thermal/ chemical environment contributed to suit disruption cannot be determined from the debris and because the environment's affects on suit materials is not understood.

Conclusion L4-1. Although the advanced crew escape suit (ACES) system is certified to operate at a maximum altitude of 100,000 feet and to survive exposure to a maximum velocity of 560 knots equivalent air speed, the actual maximum protection environment for the ACES is not known.

Recommendation L3-5/L4-1. Evaluate crew survival suits as an integrated system that includes boots, helmet, and other elements to determine the weak points, such as thermal, pressure, windblast, or chemical exposure. Once identified, alternatives should be explored to strengthen the weak areas. Materials with low resistance to chemicals, heat, and flames should not be used on equipment that is intended to protect the wearer from such hostile environments.

3.2.4 Recovered debris

Although only a small percentage of ACES fabric was recovered, many hard suit components were recovered (Table 3.2-1). There was no obvious pattern to explain why the hard components that were associated with some crew members were recovered while those associated with other crew members were not recovered. Figures 3.2-15 through 3.2-21 show the major crew worn equipment components that were recovered and ascribed to specific crew members.

Table 3.2-1. ***Recovered Crew Worn Components***

Advanced Crew Escape Suit	
Helmets – seven flown	All seven helmets recovered and identified to crew members.
Suit-side helmet neck rings – seven flown	Six recovered. Seat 6 not recovered; seat 4 recovered separately from helmet; all others recovered attached to helmets.
Glove disconnect rings – 14 flown (seven right and seven left)	Nine recovered (five right and four left). • Seat 2 right side (attached to suit-side ring) • Seat 3 left side (attached to suit-side ring) • Seat 4 right side (not attached to suit-side ring) • Seat 4 left side (not attached to suit-side ring) • Seat 5 right side (attached to suit-side ring) • Seat 5 left side (attached to suit-side ring) • Seat 6 left side (not attached to suit-side ring) • Seat 6 right side (not attached to suit-side ring) • Seat 7 right side (not attached to suit-side ring)
Suit-side glove disconnect rings – 14 flown (seven right and seven left)	Seven recovered (three right and four left). • Seat 2 right side (attached to glove ring) • Seat 3 left side (attached to glove ring) • Seat 4 left side (not attached to glove ring) • Seat 5 right side (attached to glove ring) • Seat 5 left side (attached to glove ring) • Seat 6 right side (not attached to glove ring) • Seat 7 left side (not attached to glove ring)

Table 3.2-1. *Recovered Crew Worn Components* (Continued)

Advanced Crew Escape Suit	
A/N PRC-112 radios – seven flown	Four recovered. • Seat 1 • Seat 2 • Seat 4 • Seat 7
Boots – 14 flown (seven right and seven left)	Fifty-three items identified as ACES boot fragments recovered. Eight sole fragments identified from six different boots. Four matches of the sole fragments made, resulting in a total of one complete left sole, three complete right soles, a left heel fragment, and a left toe fragment.
Suit fabric	Approximately 30 fragments of suit material recovered (only one-third is cover layer material); none could be ascribed to a specific crew member.
Miscellaneous ACES components: DSC, suit breathing regulator, suit pressure relief valve, bio-instrumentation pass-through (BIP) plug (one spare BIP plug flown; it was not recovered), O_2 manifold and g-suit controller, suit vent inlet and elbow fitting, etc. – seven each flown	Four DSCs recovered. • Seat 1 • Seat 2 • Seat 3 • Seat 4 One suit breathing regulator (seat 1) recovered. Three suit pressure relief valves recovered. • Seat 3 • Seat 4 • Seat 7 Three BIP plugs recovered. • Seat 4 • Seat 6 • Seat 7 Four O_2 manifold/g-suit controllers recovered. • Seat 4 • Seat 5 (with entire EOS O_2 hose and ~34 in. of suit O_2 hose attached) • Seat 6 • Seat 7 (g-suit controller portion missing) One O_2 hose quick disconnect (seat 5) recovered. One suit vent inlet with elbow fitting (seat 6) recovered.
Parachute Harness	
EOS bottles – 14 flown (two per crew member)	Ten whole bottles and three bottle fragments (not ascribed to a specific crew member) recovered. • Seat 1 • Seat 3 • Seat 4 (two) • Seat 5 • Seat 6 (two) • Seat 7 • Two whole bottles not ascribed to a specific crew member • Three bottle fragments not ascribed to a specific crew member.
FLU-8 (life preserver unit inflation devices) – 14 flown (two per crew member)	Seven recovered. • Seat 1 • Seat 2 • Seat 3 (two) • Seat 4 • Seat 5 • Seat 6

Table 3.2-1. *Recovered Crew Worn Components* (Continued)

Parachute Pack	
SEAWARS – 14 flown (seven left, seven right)	Six recovered: four left and two right (none had parachute riser strap attached, all mated to the Frost fittings and had ~12 in. of parachute harness strap attached). • Seat 1 left • Seat 2 right • Seat 4 right • Seat 5 left • Seat 6 left • Seat 7 left
AODs – seven flown	Two recovered. • Seat 4 • Seat 7
SARSAT beacons – seven flown	Six recovered. • Seat 1 • Seat 3 • Seat 4 • Seat 5 • Seat 6 • Seat 7
Fabric	More than 180 fragments of parachute canopy, parachute cord, parachute pack, and parachute harness strap material recovered; none could be ascribed to a specific crew member.

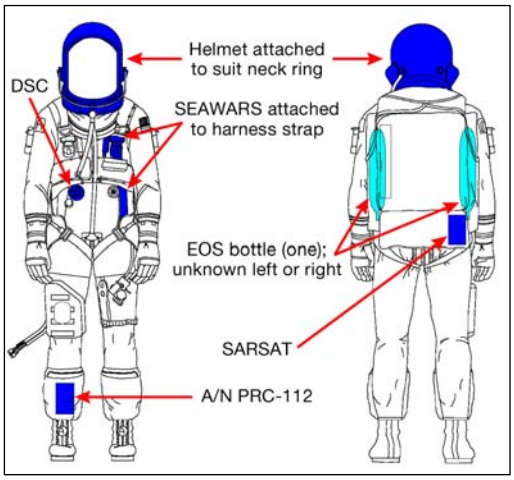


Figure 3.2-15. *Seat 1 recovered crew worn equipment.*

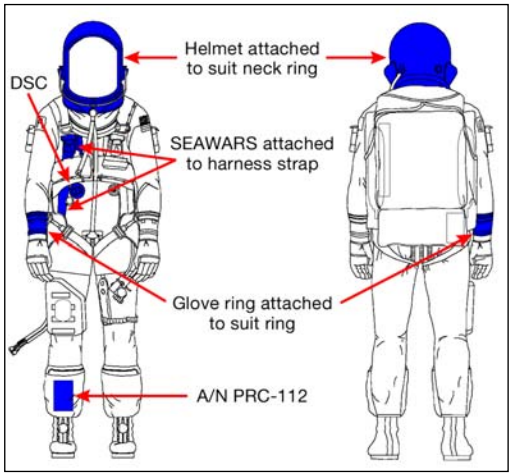


Figure 3.2-16. *Seat 2 recovered crew worn equipment.*

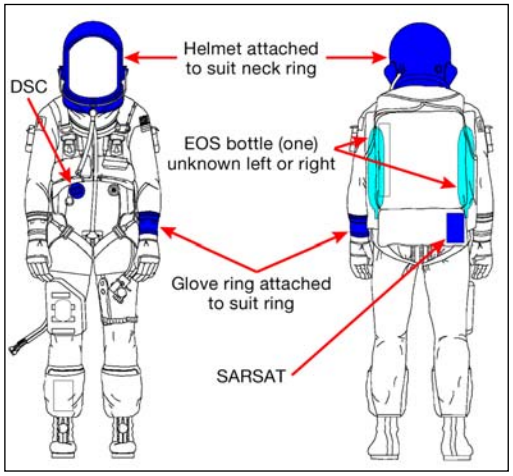


Figure 3.2-17. ***Seat 3 recovered crew worn equipment.***

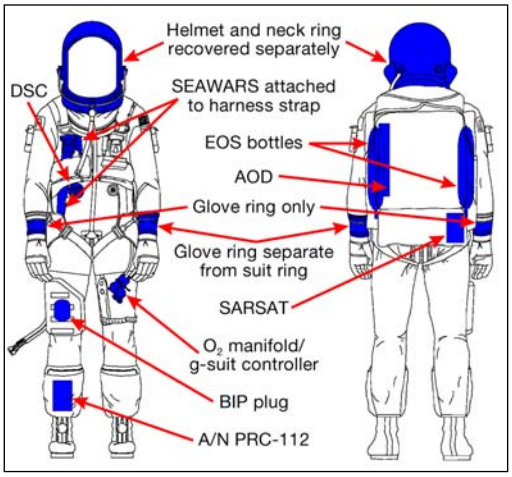


Figure 3.2-18. ***Seat 4 recovered crew worn equipment.***

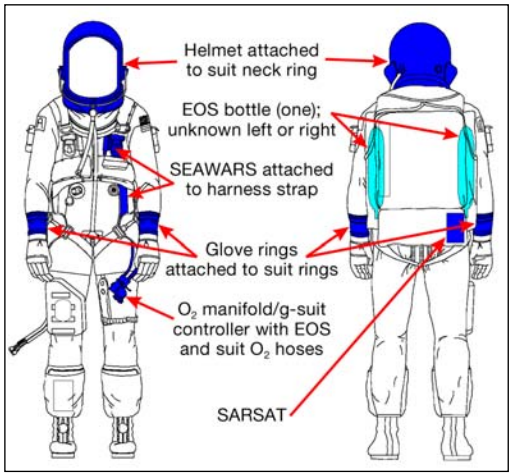


Figure 3.2-19. ***Seat 5 recovered crew worn equipment.***

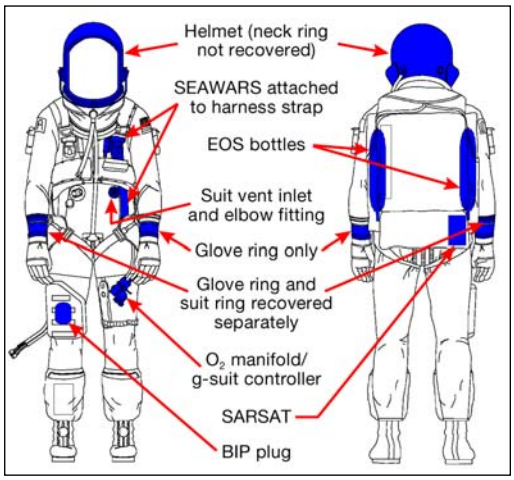


Figure 3.2-20. ***Seat 6 recovered crew worn equipment.***

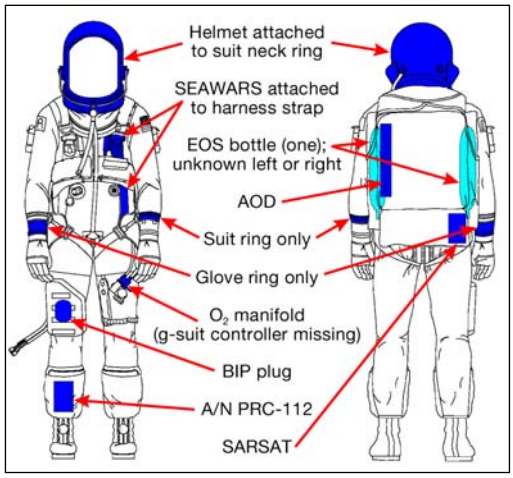


Figure 3.2-21. ***Seat 7 recovered crew worn equipment.***

Most of the suit components and subcomponents include serial numbers that are recorded and tracked to a specific crew member. This documentation aided greatly in the process of ascribing debris items to specific crew members. Thus, a very high percentage of recovered crew worn equipment was identified to specific crew members, aiding the post-accident analysis.

3.2.5 Helmets

Undamaged helmets and neck rings are shown in figure 3.2-4 and figure 3.2-2, respectively.

3.2.5.1 *General condition*

All seven helmets and six of the seven neck rings were recovered. Five of the seven helmets were recovered with the neck ring attached. One neck ring was recovered separate from the helmet. Inspection revealed that this neck ring had been mechanically removed from the helmet due to fracture of the latch mechanism. Detailed inspection of all helmets and helmet-to-neck-ring interfaces indicates that all crew members except one had their helmets on and latched at the time of the CMCE.[5] Helmet separation from the suit occurred between the suit-side neck ring and the suit fabric interface.

Finding. One crew member did not have the helmet donned at the time of the CMCE. Three of the seven crew members did not complete glove donning for entry. The deorbit preparation period of shuttle missions is so busy that crew members frequently do not have enough time to complete the deorbit preparation tasks (suit donning, seat ingress, strap-in, etc.) prior to the deorbit burn (see **Recommendation L1-2**).

The condition of each helmet shows effects from mechanical loading and thermal exposure. Effects from thermal exposure were generally consistent across all helmets, except for the helmet that was not donned at the time of the CMCE. This helmet had more pressure visor material remaining. The effects from mechanical loading were generally consistent across all seven helmets. The magnitude and distribution of mechanical damage was not severe, except for damage caused by ground impact.

[5]According to experienced astronauts, shuttle crews often struggle to complete all actions in the time allotted, giving priority to time-critical orbiter systems activities and reordering the tasks as necessary. Deorbit preparation activities frequently extend into the time after the deorbit burn and entry interface. Per the STS-107 crew's deorbit preparations plan, the crew member whose helmet was not donned was the last crew member scheduled to ingress the seat and don the helmet.

3.2.5.2 *Thermal condition*

Thermal effects were apparent throughout all helmet surfaces. Significant variations in thermal conditions were noted from helmet to helmet (both interior and exterior helmet surfaces). The reflective tape was missing from all of the helmets, and fiberglass delaminations of various sizes and depths were observed. Some white paint remained, except in the areas removed via fiberglass delamination. Residual paint on the exterior helmet surfaces shows signs of damage (pitting) that are consistent with impacts with many small debris items. Figure 3.2-22 shows examples of delaminations and pitting damage.

Figure 3.2-22. *Delamination and pitting damage*.

Small amounts of residual melted suit material were discovered, all of which were confined to the helmet/neck ring area. Melted suit bladder materials (nylon and Teflon) were observed on both sides of the helmet/neck ring interface on all helmets (except on the one helmet for which the suit-side neck ring was not recovered). Nomex material was absent from the internal and external helmet surfaces. Inspection of the suit bladder clamp interface on the neck ring yielded only nylon and Teflon (no Nomex) materials. This indicates that the Nomex material failed mechanically before the thermal decomposition temperature of 932°F (500°C) was reached. Helmet separation from the suit occurred primarily due to mechanical (aerodynamic) forces; the helmets were not "melted off" the suit. Mechanical (aerodynamic) disruption of the suit occurred prior to completion of the heating period. Melted suit material was deposited onto the helmet and neck ring areas after mechanical separation of the neck ring (small fragments of suit material were still clamped into the neck ring upon mechanical separation).

On three of the seven helmets, the upper visor reinforcement bar was recovered with some pressure visor material still attached; no sun shield material remained on any of the helmets. The upper and lower visor bars along with visor materials on each of the other four helmets were not recovered. The visor is constructed of a laminate of polycarbonate and polymethylmethacrylate. These materials do not have a true melting point but instead have a glass transition temperature.[6] The glass transition temperature for polymethylmethacrylate is approximately 230°F (110°C). The glass transition temperature for polycarbonate is approximately 300°F (149°C). Thermal gravimetric analysis testing was conducted to determine the temperature at which thermal decomposition (pyrolysis) in air begins. The thermal decomposition temperatures for polymethylmethacrylate, fiberglass epoxy resin, and polycarbonate are 572°F (300°C), 735°F (391°C), and 752°F (400°C), respectively. These temperatures are for pyrolysis in air. Tests conducted in nitrogen (N_2) yielded thermal decomposition temperatures roughly 55°F (13°C) to 90°F (32°C) higher. It is unknown

[6]The temperature above which the mechanical properties of a material are reduced significantly and the material will flow.

whether these temperatures would be higher or lower in a low-pressure, monatomic oxygen (highly reactive) environment.

On all three helmets that have remaining pressure visor material, the polymethylmethacrylate flowed and pyrolyzed and the polycarbonate flowed in some places but did not pyrolyze (figure 3.2-23). Therefore, helmet visor materials experienced at least 300°F (149°C), which is the glass transition temperature of polycarbonate, to over 572°F (300°C), the pyrolysis temperature of polymethylmethacrylate, but certainly less than 752°F (400°C), which is the pyrolysis temperature of polycarbonate. Although there were small localized areas of fiberglass pyrolysis, in no case was there global pyrolysis of the helmet fiberglass material, indicating that the helmets did not experience temperatures globally above 735°F (391°C).

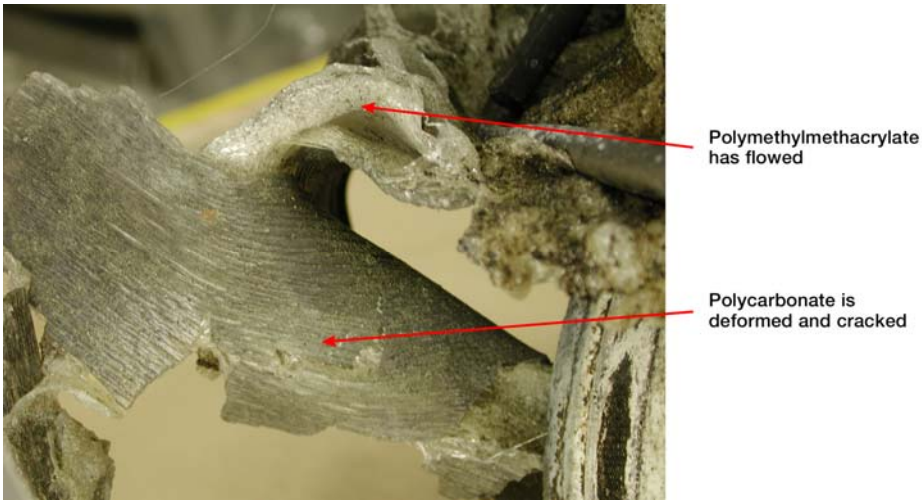


Figure 3.2-23. ***Helmet pressure visor thermal effects*****.**

The Object Reentry Survival Analysis Tool (ORSAT) was used to predict thermal damage for helmets released at various times in the trajectory. The helmet was modeled as an 11-in.-diameter sphere weighing 6.3 lbs., with an initial temperature of 80°F (27°C). The analysis concluded that free-flying helmets released around GMT 14:01:03 would have received heating sufficient to cause damage similar to that seen on the visors of the three recovered helmets. Ballistics analysis provided helmet release times consistent within 10 seconds of the ORSAT-predicted time, confirming the approximate time of helmet separation from the CM. No significant inconsistencies were noted among ballistics analysis, ORSAT analysis, materials testing, and debris observations.

Despite the flow of the visor material, this material is notably absent from the helmet visor seal around the face opening, indicating that the visors were not in contact with the visor seal when heating occurred and were not down and locked.

Finding. Inspection of all seven recovered helmets confirmed that none of the crew members lowered and locked their visors (see **Recommendation L1-3/L5-1**).

3.2.5.3 *Mechanical condition*

In all cases, the helmet structure remained intact. The helmets experienced a range of localized mechanical damage (fractures), but did not experience massive structural damage from external impacts prior to ground impact. External helmet impacts were insignificant in size and random in distribution. Detailed inspections differentiated the sources of internal impacts.

Finding. The current ACES helmets are nonconformal and do not provide adequate head protection or neck restraint for dynamic loading situations.

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments.

Recommendation L2-7. Design suit helmets with head protection as a functional requirement, not just as a portion of the pressure garment. Suits should incorporate conformal helmets with head and neck restraint devices, similar to helmet/head restraint techniques used in professional automobile racing.

The hold-down cables on each neck ring were severed at the attach points to the cable guide tubes due to mechanical overload (figure 3.2-24). Most cable guide tubes experienced significant plastic deformation. The guide tubes display evidence of external contaminants (i.e., melted metal and suit material) and thermal effects on top of the fractures and localized deformation. This indicates that mechanical loading preceded exposure to the thermal environment. Rotation of the helmet relative to the normal forward position was observed on all neck rings varying from 90 to 180 degrees. Major cable guide tube deformation and helmet rotation indicates that a significant loading event occurred where helmets were removed via a mechanical (nonthermal) mechanism.

Figure 3.2-24. ***Hold-down cable guide tube.***

One of the seven helmets was recovered with the bailer bar still attached. All other helmets had the bailer bar mechanically removed, although the bailer bar cam mechanism remained in place on the starboard and portside helmet interfaces.

The bailer bar latch mechanisms on five of the seven helmets remained attached to the helmets in good condition (figure 3.2-24). This would not be expected if the crew members had lowered and locked their visors. Mechanical separation of the bailer bar would be accompanied by fracture of the latch assembly if the visor was down and the bailer bar was locked. The other two helmets experienced latch mechanism separation due to failure of the fasteners that attach the latch mechanism to the helmet before subsequent deposition of melted suit materials. This suggests that latch separation was followed by suit melting. Neither of the two helmets shows evidence of the indentation or deformation that would be associated with forces expected if the bailer bar ripped the latch from the neck ring. Combined with the absence of melted visor material on the visor seal, this confirmed the conclusion that none of the crew members lowered and locked visors.

3.2.6 Glove disconnects

Undamaged glove disconnects are shown in figure 3.2-25.

Figure 3.2-25. ***Examples of intact suit-side*** **(left)** ***and glove-side*** **(right)** ***glove disconnects*****.**

Twelve glove disconnect debris items were recovered, corresponding to six of the seven *Columbia* crew members (no glove disconnect rings were recovered for seat 1). Although evidence of exposure to entry heating was noted on all disconnect rings, the level of heating varied from item to item, with differences between the left and right sides of items from the same crew member. Inspection of recovered disconnect rings indicates that three crew members did not have their gloves mated to their suits for entry. Inspection indicates that three crew members had gloves mated to their suits. The recovered flight deck entry video supports these conclusions. The video also indicates that the crew member in seat 1 also had gloves mated to the suit.

Finding. One crew member did not have the helmet donned at the time of the CMCE. Three of the seven crew members did not complete glove donning for entry. The deorbit preparation period of shuttle missions is so busy that crew members frequently do not have enough time to complete the deorbit preparation tasks (suit donning, seat ingress, strap-in, etc.) prior to the deorbit burn (see **Recommendation L1-2**).

Melted aluminum deposits and/or tiny craters were observed on most of the glove and suit rings. Melted suit material (figure 3.2-26) was discovered on all recovered disconnect rings. Close inspection of the suit bladder clamp interface on the suit-side disconnect rings revealed mainly nylon and Teflon materials. In all cases, minimal amounts of Nomex remained clamped to the interface. Overall, the amount of residual melted suit material seems to correlate with the general magnitude of heating; that is, higher-magnitude heating resulted in the pyrolysis of residual suit material. As with the helmets, deposition of melted suit material on the glove disconnect areas occurred after mechanical separation. Small fragments of suit material were still clamped in the disconnect after mechanical separation. The failure modes at the disconnect ring (the ring-to-suit material interface) were similar to those observed in the suit-side helmet disconnect rings (see Section 3.2.5).

Figure 3.2-26. ***Recovered suit-side*** **(upper left),** ***glove-side*** **(upper right),** ***and mated*** **(bottom)** ***glove disconnects*****.**

3.2.7 Dual suit controllers

An undamaged DSC is shown in figure 3.2-27.

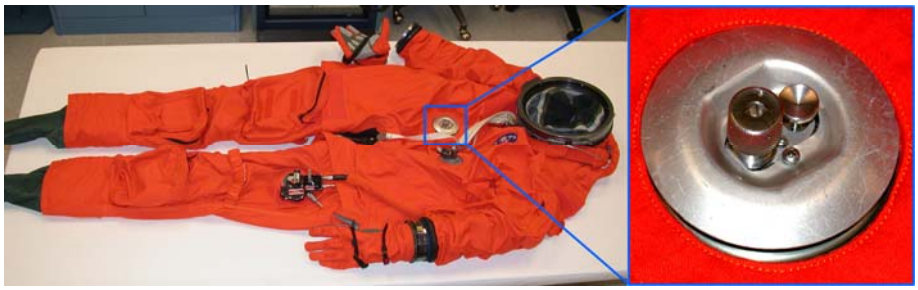


Figure 3.2-27. ***Example of an intact dual suit controller*****.**

Four DSCs were recovered, all from flight deck crew members and all presenting similar appearances. One DSC (figure 3.2-28) was disassembled and inspected to determine the thermal environment exposure. The suit material edges were ragged with some localized melting around the edges, indicating that there was a mechanical disruption of the suit followed by thermal exposure to the suit fabric that was still attached to the DSC body. The DSC body back surface (i.e., the surface inside the suit) had many craters and one penetration (~0.5 in. × 0.25 in.). This indicates a rapid disruption of the suit, releasing the DSC while it was still in close proximity to the CM debris cloud. Some melting of the suit material to the DSC body caused the suit flange to adhere to the DSC body. All internal soft goods (O-rings, seals, diaphragms) were intact and showed no signs of mechanical or thermal damage. Both of the aneroids[7] were still hermetically sealed with the sealing solder intact. The suit cover-layer Velcro was melted to the suit restraint-layer Velcro in some

[7]Small, sealed metal bellows, sensitive to air pressure, that are part of the suit pressurization control mechanism.

areas. Although melting of the torn fabric edges and the Velcro® indicates thermal exposure, the complete lack of thermal damage to any of the internal soft goods suggests that the thermal exposure was limited in intensity and/or duration.

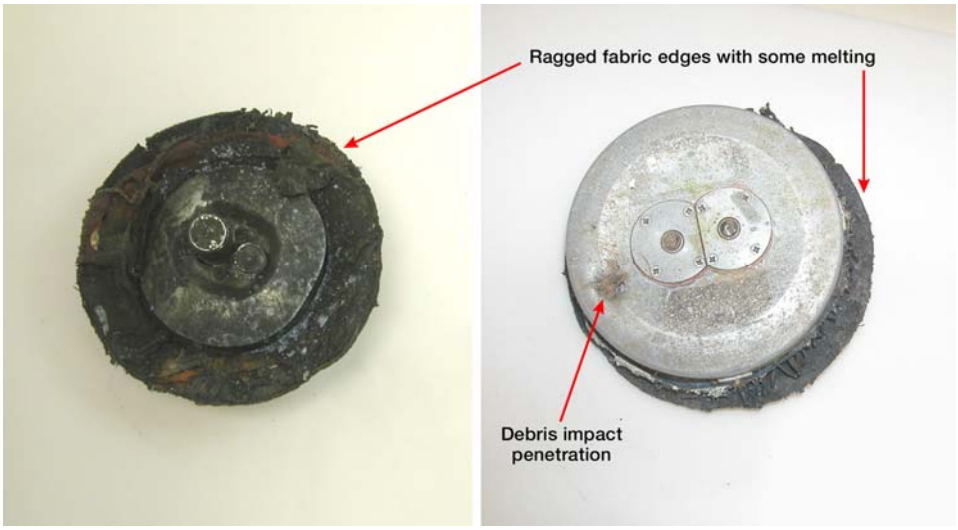


Figure 3.2-28. ***Recovered dual suit controller front*** **(left)** ***and back*** **(right).**

3.2.8 Boots

Undamaged boots are shown in figure 3.2-7.

A total of 53 possible boot fragments were recovered, including eight sole fragments from six boots and fragments of leather uppers and inserts. All fragments exhibited mechanical and thermal damage. No identifiable pieces of the shoe laces or nylon sections of the boots were recovered.

The condition of each recovered boot sole shows effects from mechanical loading and thermal exposure. Mechanical loading resulted in the removal of the boot leather uppers from five of the six recovered soles and fracture of four of the six recovered boot soles. Forensic evidence indicates that the boots were worn by the crew members during thermal exposure and that the boots failed mechanically prior to the conclusion of thermal exposure. Evidence indicates that the soles of these boots failed, with either the toe section or the complete sole being removed first followed by the remainder of the boot. The nylon lower sections of some of the boots appear to have been thermally penetrated prior to mechanical removal of the leather upper. Effects from thermal exposure were generally consistent across all soles. Edges, including the fracture edges, exhibited thermal erosion.

In an attempt to match the observed thermal damage, boot soles of flight-like boots were heated in an oven to identify the range of thermal effects with varying thermal exposure. The test samples were exposed to 750°F (399°C), 1,000°F (538°C), or 1,250°F (677°C) at normal atmospheric pressure conditions (~14.7 psi, ~20% O_2) for 15, 30, 45, or 60 seconds. The materials showed no significant changes in appearance until they combusted. This initially puzzled the team until it became clear that the presence of O_2 was affecting the results. The tests were repeated using new samples that were heated in an N_2 purge (<3% O_2). Results of the revised test protocol appeared to be similar to the recovered boot sole fragments. The test samples that most closely matched the recovered debris items were those that were exposed to 1,000°F (538°C) for 30 to 45 seconds or 1,250°F (677°C) for 15 to 30 seconds. However, no credible scenario could be envisioned in which the *Columbia* boots would be exposed to these temperatures for the length of time indicated by the tests, so the test results could not be correlated directly to the debris observations. Because the test conditions (~14.7 psi, 97% to 99% N_2, 1% to 3% O_2) did not sufficiently approximate the entry environment

conditions (low ambient pressure, monatomic O_2, and possibly high dynamic pressure), they are a potential source of error in this analysis.

3.2.9 Emergency Oxygen System

An intact EOS is shown in figure 3.2-12.

Ten whole EOS bottle/reducers were recovered (eight were ascribed to specific crew members), each with no O_2 remaining. All 10 of the recovered EOS bottle/reducers have similar appearances, with some variance in the amounts of material deposition. Additionally, three fragments of bottles were also recovered (figure 3.2-29).

Figure 3.2-29. ***Recovered whole Emergency Oxygen System bottle/reducers*** **(left and center)** ***and a bottle fragment*** **(right).**

No nylon material from the parachute harness adhered to the bottle/regulator assemblies. Minor evidence of elevated temperatures, directional burn marks, and discrete external impacts are visible. As with most of the *Columbia* hardware, corrosion that occurred while the debris was on the ground is also evident. X ray revealed that all of the EOSs were activated. However, the crew members were trained not to activate them unless their visors were down. Therefore, activation of the EOS (achieved by applying tension to the activation cables) probably occurred as the bottles separated from the harnesses rather than by crew action.

The overall appearance of the 10 recovered whole EOS bottle/reducers suggests that each EOS bottle/reducer assembly experienced similar thermal and mechanical environments. Each EOS assembly was mechanically extracted from the harness as temperatures were rising; then for a short duration and nearly simultaneously, they experienced ballistic heating and some metal pellet-like impacts. This indicates a rapid disruption of the parachute harness, releasing the EOS while it was still in close proximity to the CM debris cloud.

The EOS bottle fragments exhibited irregular edges along the fracture surfaces, some outward bent edges, evidence of heating on the inner surfaces, and some deposited/flowed black material along a fracture surface.

Heating on the inner surfaces indicates that the bottle failure occurred before the end of entry heating. Neither the cause of the bottle failures nor the status of the bottles at the time of failure (pressurized or unpressurized) could be determined.

3.2.10 Seawater Activated Release System

Six SEAWARS were recovered. None had automatically ignited, and all had both of the Frost fitting male and female halves still mated. All SEAWARS had similar appearances (figure 3.2-30), consisting of the SEAWARS assembly still attached to approximately 12 in. of nylon parachute harness strap. None had any parachute riser material attached.

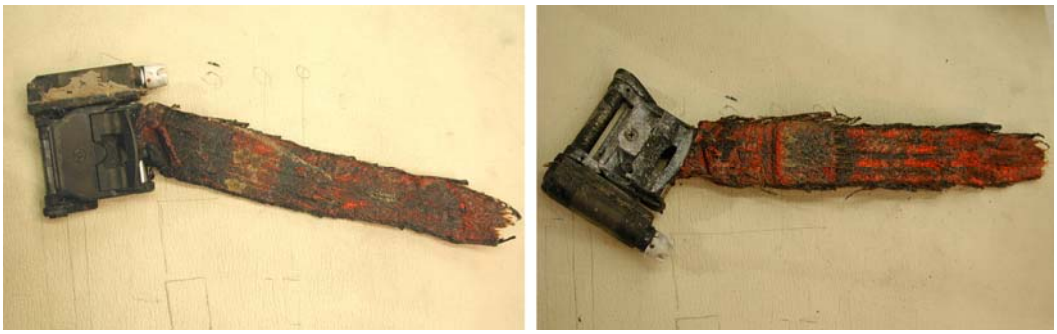


Figure 3.2-30. ***Seawater Activated Release System with Frost fittings (male/female) still mated, with a segment of harness strap attached*** **(two different SEAWARS are shown).**

Each SEAWARS/strap item shows evidence of directional melting, burning, and mechanical loading. The surviving length of parachute harness strap is consistent with a harness strap failure at the waist. While the terminating ends show evidence of melting on the top layer of fabric, these ends appear to have failed primarily due to mechanical overload, not melting. Some surface melting occurred along the length of the straps, with distinct directionality away from the SEAWARS towards the broken end of the strap, corresponding to a head-to-foot direction when the harness is on a crew member. These directional heating/melting features are present on both sides of the straps, with little difference between the front and the back of the straps. Because both sides of the straps show signs of heating, the heating and melting must have occurred after the straps had separated from the crew member. Localized heating on the metallic SEAWARS Frost fitting buckle suggests intense, short-duration heat exposure. Each SEAWARS and strap was mechanically extracted from the harness webbing before experiencing this short-duration heating. Directionality suggests that each piece of riser trimmed with the SEAWARS into the airflow (figure 3.2-31).

Figure 3.2-31. ***Directional melting on Seawater Activated Release System/parachute harness strap*****.**

3.2.11 Telonics Satellite Uplink Beacon-A search and rescue satellite-aided tracking beacon

An undamaged TSUB-A SARSAT beacon is shown in figure 3.2-14.

One TSUB-A SARSAT beacon is flown per crew member; it is located in the survival raft packed in the PPA. Six SARSAT beacons were recovered; none of them were activated during the accident. All six recovered SARSAT beacons show similar thermal and mechanical damage (figure 3.2-32).

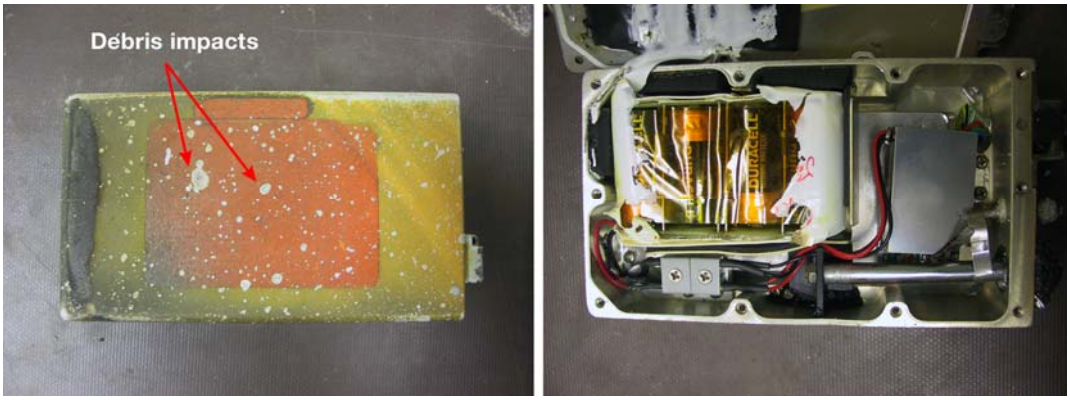


Figure 3.2-32. ***External*** **(left)** ***and internal*** **(right)** ***views of a recovered search and rescue satellite-aided tracking beacon*****.**

No material from the raft or the PPA was adhered to the outer aluminum casing. Various amounts of paint remained, displaying evidence of impacts with small, hot metal pellets. Corrosion was also present. All external plastic was melted. Internal inspection of the beacons revealed minor solder re-flow, with most of the components mechanically and electrically intact. Two SARSAT beacons were tested and, when externally powered, functioned properly. For each of the six SARSAT beacons, the processing module was extracted and interfaced with ground support equipment to read the beacon's unique identifier, thereby allowing all six beacons to be ascribed to individual crew members. Each SARSAT beacon was mechanically extracted from the PPA as temperatures were rising; then for a short duration and nearly simultaneously, it experienced high heating and a hot metal pellet-like shower. This indicates a rapid disruption of the parachute packs and release of the SARSAT beacons while they were still in close proximity to the CM debris cloud.

3.2.12 Army/Navy personal radio communications-112 radio

An undamaged A/N PRC-112 radio is shown in figure 3.2-10.

Four A/N PRC-112 radios were recovered, all of which looked similar (figure 3.2-33). No Nomex material from either the suit pocket or the survival gear pouch was adhered to the outer aluminum casing. Various amounts of paint showed evidence of impact with small, hot metallic pellets. Corrosion was also present. All external plastic was melted or missing. Internal inspection of the radios revealed evidence of moderate heating, with the center-most components experiencing only minor solder re-flow. For each of the four radios that were recovered, the control module was extracted and interfaced with ground support equipment to read the unique identifier of the radio. Thus, all four radios were ascribed to a specific crew member. Overall appearance suggests that each A/N PRC-112 radio experienced similar thermal and mechanical environments. Each radio was mechanically extracted from the ACES pocket and the survival gear pouch as temperatures were rising and, for a short duration and nearly simultaneously, each of the radios experienced high heating and a hot metal pellet-like shower. This indicates a rapid disruption of the suit survival gear pockets, releasing the radios while they were still in close proximity to the CM debris cloud.

Figure 3.2-33. ***External*** **(left)** ***and internal*** **(right)** ***views of a recovered Army/Navy personal radio communications-112 radio.***

3.2.13 Ground plot analysis

To help understand the breakup sequence of the CM, the recovery locations of pieces of crew equipment were analyzed to determine the order in which the crew members separated from the CM. The human body's complex geometry is difficult to model for precise ballistic analysis and can result in significant variations in trajectory. As a result, the recovery locations of the crew remains is unreliable data for determining the order in which the crew members separated from the CM. Therefore, recovery locations of the crew remains were not included in this analysis.

Items selected for analysis (helmets, SARSAT beacons, and A/N PRC-112 radios) were chosen due to the small variations in the conditions of like items (i.e., all helmets were recovered in the same general condition) and their "regular" shapes (the helmets are roughly spherical, and the SARSAT beacons and A/N PRC-112 radios are rectangular prisms), which have known aerodynamic properties and result in predictable free-flying trajectories.[8] Thus, all like items had similar flight characteristics in the fall to the ground, and can be used to determine the relative order of crew member separation from the CM. It was assumed that the items separated from each crew member in the same manner and roughly the same amount of time after each crew member separated from the CM.

As discussed above, all seven helmets were recovered and ascribed to crew members.[9] The helmets impacted the ground from west to east in the order of seat 7, seat 6, seat 5, seat 4, seat 2, seat 3, and seat 1 (figure 3.2-34).

Six SARSAT beacons were recovered (all except seat 2) and ascribed to crew members. The SARSAT beacons impacted the ground from west to east in the order of seat 6, seat 7, seat 5, seat 4, seat 1, and seat 3 (figure 3.2-34).

Four A/N PRC-112 radios were recovered and identified to crew members (seat 1, seat 2, seat 4, and seat 7). The longitude data for the seat 4 radio is highly suspect, however. These data indicate that it was recovered more than 50 miles east of all other seat 4 crew equipment debris. It is assumed that the original data point was recorded incorrectly and is off by 1 degree, so a "corrected" data point was used in the analysis. The A/N PRC-112 radios impacted the ground from west to east in the order of seat 4, seat 7, seat 1, and seat 2 (figure 3.2-34).

[8]Other crew equipment items were not selected for detailed ground plot analysis (EOS bottles, SEAWARS, etc.) due to their irregular shapes, which would result in lower confidence in the flight characteristics being similar.
[9]One of the middeck crew members did not have the helmet attached to the suit at the time of the breakup. It is possible that the helmet was released from the CM at a different time than it would have been if it had been attached to the suit.

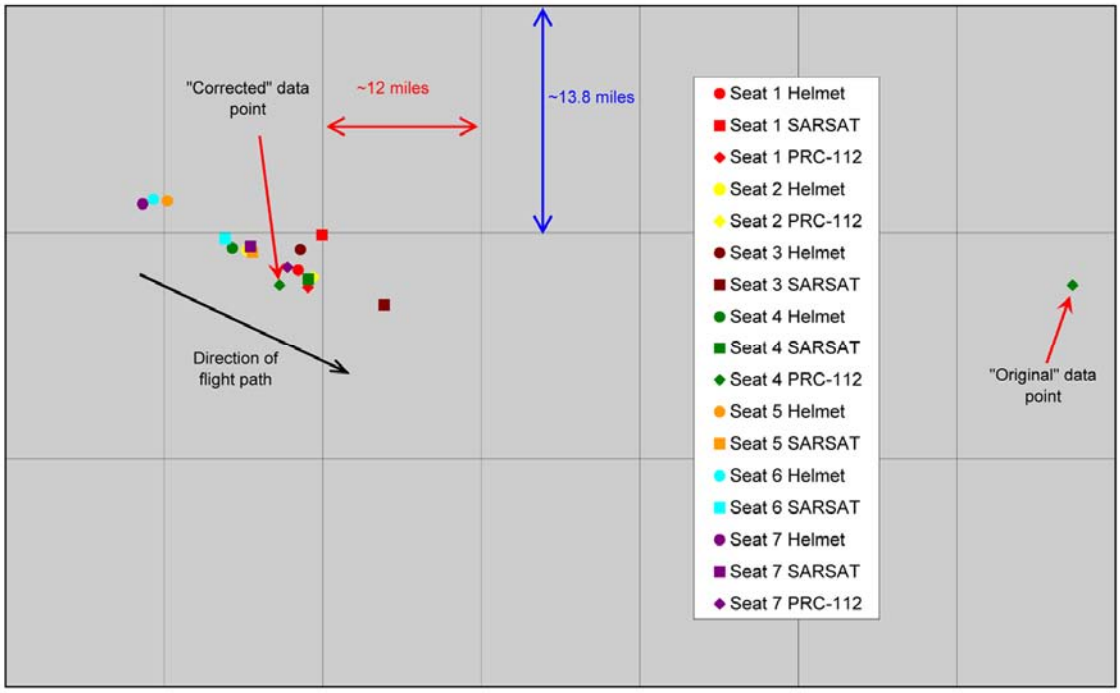


Figure 3.2-34. ***Relative location of helmets, search and rescue satellite-aided tracking beacons, and Army/Navy personal radio communications-112 radios*****.**

The investigation team concluded that the middeck crew members separated from the CM before the flight deck crew members. Orders within the middeck/flight deck groups cannot be determined conclusively, but it appears that seat 6 and seat 7 equipment items were first out of the middeck and seat 4 equipment was first out of the flight deck. All CEE associated with specific crew members was plotted by crew member. Analysis looking at the relative centers of the areas of recovered items for each crew member supports this conclusion. Figure 3.2-35 shows the locations of the recovered items for each crew member. The upper plot shows items for the flight deck crew members; the lower plot show items for the middeck crew members. Both plots are the same scale and represent the same geographic area.

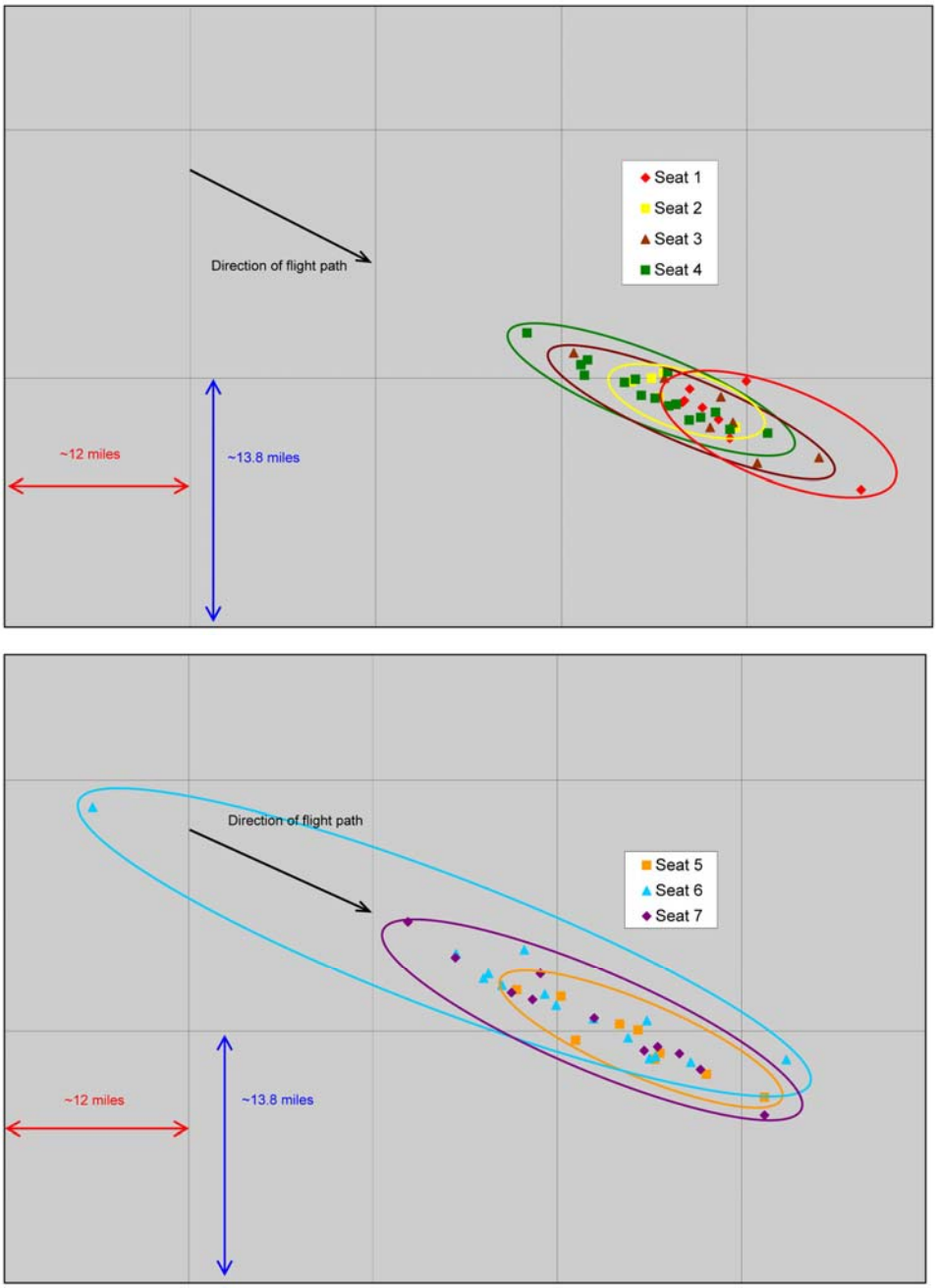


Figure 3.2-35. *Relative locations of all crew escape equipment* – flight deck crew members (top) and middeck crew members (bottom).

3.2.14 Lessons learned

3.2.14.1 *Equipment serialization and marking*

One of the most useful tools in investigating an aviation accident is physically or virtually reconstructing the vehicle from the recovered debris. Being able to identify the original location within the vehicle of debris items is of utmost importance in achieving an accurate reconstruction. Identifying the origins of debris items is made possible by serializing individual piece parts and subassemblies, and keeping accurate records of the piece part/subassembly serial numbers at the assembly and, ultimately, the vehicle levels.

This is especially useful when there are multiple units of identical or similar components, such as crew equipment, seats, engines, or structural members.

The hard components of the ACES, parachute harness, and parachute pack are serialized and tracked to the top-level assembly. Records are kept regarding which crew member is using which suit, harness, and parachute pack. Because of this meticulous recordkeeping, the recovered helmets, glove rings, SEAWARS, DSCs, ACES pressure relief valves, BIP plugs, O_2 manifolds, EOS bottles, and AODs were identified to a specific crew member. In most cases, the serial numbers are etched or physically stamped on the components, aiding identification. For the SEAWARS, the SARSAT emergency beacons, and the A/N PRC-112 survival radios, the identification labels were damaged or destroyed by entry heating. Identification was possible by disassembling the units and inspecting subcomponent serial number labels (in the case of the SEAWARS), or by reading the programmed unique transponder information in the beacon and radio electronics.

Finding. Most of the suit components and subcomponents include serial numbers that are recorded and tracked to a specific crew member. This configuration management documentation aided greatly in the process of ascribing the debris items to specific crew members.

> ***Recommendation A5.*** Develop equipment failure investigation marking ("fingerprinting") requirements and policies for space flight programs. Equipment fingerprinting requires three aspects to be effective: component serialization, marking, and tracking to the lowest assembly level practical.

3.2.14.2 *Suit requirements and design*

The crew escape suits (the ACESs) were designed to enable survival of crew members during egress and escape from the shuttle in emergency situations. There were specific requirements for the suit to protect crew members from contaminated atmosphere and smoke. As with other materials used in the shuttle, the suit materials were required to be nonflammable or self-extinguishing. However, the suit assembly did not have functional requirements to protect the crew members from environments involving elevated temperatures or fire, as might be present during an emergency egress due to a fire at the launch pad.

As part of the certification testing of the U.S. Air Force suit, suits were subjected to flame pit tests in which suited manikins were placed in a jet fuel fire for 3 seconds and then removed. The suits performed well, with no structural failures and no expected burns to the occupant (based on temperature sensors on the manikin). The ACES is similar to the U.S. Air Force suit, so it may be expected that the ACES would perform well in similar tests. However, the ACES ensemble has some design and materials differences from the U.S. Air Force suit. One notable difference is the use of nylon on the ACES parachute harness straps and the boots. The use of nylon presents a potential weakness in the suit if the suit is used in an environment entailing elevated temperatures or fire.

Finding. The ACES had no performance requirements for occupant protection from elevated temperatures or fire. The ensemble includes nylon on the parachute harness straps and the boots. The ACES may not provide adequate protection to crew members in emergency egress scenarios involving exposure to heat and flames.

> ***Recommendation L3-5/L4-1.*** Evaluate crew survival suits as an integrated system that includes boots, helmet, and other elements to determine the weak points, such as thermal, pressure, windblast, or chemical exposure. Once identified, alternatives should be explored to strengthen the weak areas. Materials with low resistance to chemicals, heat, and flames should not be used on equipment that is intended to protect the wearer from such hostile environments.

3.3 Crew Training

Crew training, while not a factor in causing the *Columbia* accident, is nonetheless an important element of this report. This section will provide an overview of generic astronaut training and examine *Columbia*-specific crew training. Finally, this section will provide an in-depth analysis and discussion of the impact of training on the actions taken by the STS-107 crew as events unfolded during vehicle entry.

The following is a summary of findings, conclusions, and recommendations for this section.

Finding. The current training regimen separates vehicle systems training from emergency egress training. Emergency egress training sessions exercise the procedures and techniques for egressing the shuttle CM without emphasizing the systems failures that caused the emergency condition. The egress training events are performed on different days from the systems training events, with little discussion of the transition between systems malfunctions and the decision to egress the vehicle. Crew members become conditioned to focus on problem resolution rather than crew survival; the training does not adequately prepare the crew to recognize impending survival situations. It is possible that the STS-107 crew members did not close and lock their visors during the vehicle LOC dynamics (before cabin depressurization) because they were more focused on solving vehicle control problems rather than on their own survival.

> ***Recommendation L1-1.*** Incorporate objectives in the astronaut training program that emphasize understanding the transition from recoverable systems problems to impending survival situations.

Finding. Emergency egress training for a vehicle LOC/breakup is based on extrapolated data and basic assumptions from the *Challenger* accident for aerodynamic modeling and CM dynamics. The vehicle LOC emergency egress procedures taught to shuttle crews do not address a vehicle LOC occurring during entry.

> ***Recommendation L2-1.*** Assemble a team of crew escape instructors, flight directors, and astronauts to assess orbiter procedures in the context of ascent, deorbit, and entry contingencies. Revise the procedures with consideration to time constraints and the interplay among the thermal environment, expected crew module dynamics, and crew and crew equipment capabilities.
>
> ***Recommendation L2-2.*** Prior to operational deployment of future crewed spacecraft, determine the vehicle dynamics, entry thermal and aerodynamic loads, and crew survival envelopes during a vehicle loss of control so that they may be adequately integrated into training programs.
>
> **Recommendation L2-3.** Future crewed spacecraft vehicle design should account for vehicle loss of control contingencies to maximize the probability of crew survival.

3.3.1 Overview

NASA's astronaut training program is designed to provide the systems familiarization and flight skills that are required for astronauts to operate the shuttle and carry out mission tasks effectively and efficiently. The training is structured in a building-block format, beginning with workbooks and briefings and progressing to lessons that use sophisticated trainers and simulators.

Individuals selected as astronaut candidates (ASCANs) undergo a training and evaluation period, which lasts over 1 year. This training introduces ASCANs to generic shuttle systems and flight operations, and it prepares them for more in-depth follow-on training as assigned crew members. The curriculum includes training on the Data Processing System; the Guidance, Navigation, and Control System; vehicle control and propulsion systems; the Communications and Tracking System; crew habitability; and shuttle crew escape (emergency egress) equipment and procedures. Additionally, ASCANs are given International Space Station (ISS) systems training. Upon completion of the ASCAN training, the student possesses a functional knowledge of the shuttle systems, ISS systems, and flight operations procedures.

After the initial training period, ASCANs receive advanced training that may lead to assignment to a shuttle flight crew. While awaiting flight assignment, ASCANs maintain proficiency in shuttle systems and flight operations through recurring proficiency lessons in the shuttle mission simulator (SMS). They may also elect to take single system trainer refresher lessons in various orbiter systems. In addition, ASCANs receive mission-specific courses such as Payload Deployment and Retrieval System, Mobile Servicing System, rendezvous and proximity operations, and extravehicular activity (spacewalk).

Upon assignment to a specific flight, crew members progress to flight-similar operations (ascent, orbit, and deorbit/entry) lessons and begin the appropriate mission-specific courses. Assigned crew training also includes flight-specific shuttle mockup training sessions on crew habitability and crew escape. Ascent/entry flight operations training prepares orbiter crew members for crew ingress, orbital insertion, deorbit burn, and landing.

Manual flying techniques are covered in several lessons. All CDRs and PLTs become proficient in manual skills for nominal[1] ascents, aborts, and entries. Crew coordination (space flight resource management (SFRM)) objectives are included to enhance team effectiveness and to ensure mission safety and success.

All training courses lead to integrated simulations. The integrated simulations build the team coordination between the crew and the flight control teams in the Mission Control Center (MCC) that is necessary to ensure a successful mission.

The training regimen encourages systems knowledge and problem resolution through appropriate analysis of displays and use of checklists. While the regimen incorporates scenarios that involve multiple systems failures, in general it is considered nonproductive to train scenarios from which there is no recovery and so those cases are not simulated. No simulation cases are intentionally scripted to result in explosive cabin depressurization or vehicle LOC. Unrecoverable conditions are not intentionally presented to the crew during training. There have been isolated cases in which simulations have ended in a vehicle LOC, but those instances are usually a result of an unrealistic number of simulated systems failures occurring at the same time, seemingly unrelated simulated failures interacting in unforeseen ways, failures being entered into the simulation computer system in the wrong order, crew or flight control team error or miscommunication, or an unexpected failure of the simulation computer.

Training is segregated based on the topic, the activity, and the limitations of each training facility. Sessions for systems failures, which may eventually result in the need to perform an emergency egress of the vehicle, are conducted in a facility that adequately simulates the software and hardware responses of the orbiter and provides an accurate representation of the flight deck interior only. In contrast, emergency egress procedure training is conducted in a volumetrically correct mockup of the entire CM that lacks the capacity for simulating systems malfunctions. The purpose of the emergency egress training sessions is to exercise the procedures and techniques for egressing the shuttle CM without emphasizing the systems failures that "caused" the emergency egress. The training events are performed at different times on different days with little discussion of the transition between a systems malfunction and the decision to egress the vehicle.

[1]Within acceptable boundaries.

Finding. The current training regimen separates vehicle systems training from emergency egress training. Emergency egress training sessions exercise the procedures and techniques for egressing the shuttle CM without emphasizing the systems failures that caused the emergency condition. The egress training events are performed on different days from the systems training events, with little discussion of the transition between systems malfunctions and the decision to egress the vehicle. Crew members become conditioned to focus on problem resolution rather than crew survival; the training does not adequately prepare the crew to recognize impending survival situations. It is possible that the STS-107 crew members did not close and lock their visors during the vehicle LOC dynamics (before cabin depressurization) because they were more focused on solving the vehicle control problems rather than on their own survival.

> ***Recommendation L1-1.*** Incorporate objectives in the astronaut training program that emphasize understanding the transition from recoverable systems problems to impending survival situations.

The emergency egress training program includes classroom sessions on shuttle CM egress procedures in the event of an LOC and vehicle breakup. This training is given to new astronauts as part of the ASCAN training program, and is given to flight-assigned shuttle crews in flight-assigned training just prior to launch in the escape systems refresher class. The training discusses procedures (figure 3.3-1) that are based on extrapolated data from the *Challenger* accident. The analysis of the *Challenger* data used basic assumptions for the vehicle/CM attitudes and dynamics.[2] However, this analysis had not been updated using the more sophisticated techniques available since the *Challenger* accident. Aerodynamic modeling performed for the current investigation provided estimates of CM dynamics following vehicle breakup. These dynamics differed from the assumptions that were made in the Shepherd-Foale Report, and have significant bearing on the LOC/breakup egress procedures. Additionally, the emergency egress procedures and training cover the case of a vehicle breakup during ascent,[3] which has a relatively benign thermal environment when compared to the entry trajectories. The *Columbia* accident brought to the forefront the higher aerodynamic and heat stresses that adversely affect the shuttle CM survival (and, therefore, crew survival) found in the entry environment.

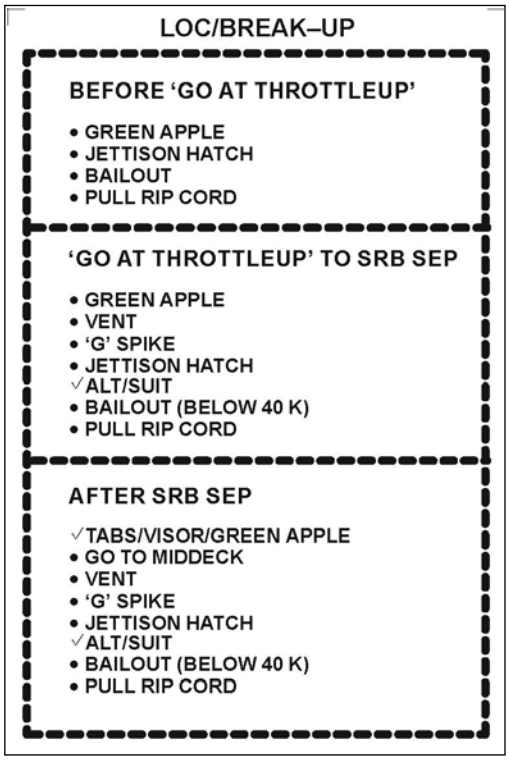


Figure 3.3-1. ***Loss of control/breakup cue card as flown on STS-107.***

[2]Crew Bailout Procedure for LOC/Breakup Report, B. Shepherd and M. Foale, September 25, 1989.
[3]Prior to STS-107, NASA had not experienced a vehicle breakup during entry.

Finding. Emergency egress training for a vehicle LOC/breakup is based on extrapolated data and basic assumptions from the *Challenger* accident for aerodynamic modeling and CM dynamics. The vehicle LOC emergency egress procedures taught to shuttle crews do not address a vehicle LOC occurring during entry.

Recommendation L2-1. Assemble a team of crew escape instructors, flight directors, and astronauts to assess orbiter procedures in the context of ascent, deorbit, and entry contingencies. Revise the procedures with consideration to time constraints and the interplay among the thermal environment, expected crew module dynamics, and crew and crew equipment capabilities.

Recommendation L2-2. Prior to operational deployment of future crewed spacecraft, determine the vehicle dynamics, entry thermal and aerodynamic loads, and crew survival envelopes during a vehicle loss of control so that they may be adequately integrated into training programs.

Recommendation L2-3. Future crewed spacecraft vehicle design should account for vehicle loss of control contingencies to maximize the probability of crew survival.

3.3.2 *Columbia* crew training

All STS-107 crew members completed the applicable ASCAN, core systems refresher, and ascent/entry flight operations training programs. The crew also completed all prescribed flight-specific training programs, including ascent and entry proficiency simulator training sessions, crew habitability, and crew escape/crew survival training as well as the prescribed ascent and entry integrated simulations prior to the launch delay in June 2002. An additional five ascent integrated simulations and three entry integrated simulations were completed prior to the January 2003 launch. The prescribed post-insertion and deorbit preparation simulations were also completed.

Launch delays, which were caused by main engine flowliner issues, resulted in the need to repeat some training. The entire crew repeated the water survival lessons (classroom and in-water sessions) in November 2002. The terminal countdown demonstration test (TCDT) was postponed several months. The crew had already completed the prelaunch ingress/egress mockup training session in June 2002, just prior to the postponement. Once the TCDT date was finalized, the CDR opted to retake the prelaunch ingress/egress lesson in November 2002. The escape systems refresher lesson was given in June 2002 but was also repeated in December 2002 at the CDR's request. The crew also performed additional ascent and entry proficiency training sessions in the simulator following the launch delay in June 2002.

Throughout their training, the STS-107 crew members displayed expert orbiter systems knowledge, correct and thorough procedure execution, and excellent SFRM techniques. The crew was very rigorous in verbalizing and verifying procedural steps and routinely took time to brief SFRM topics before each simulation. The launch delays kept the crew in training together for more than 2 years, resulting in a well-trained and finely tuned team. For example, during a simulation run, the *Columbia* crew exercised crew coordination skills by performing the entire run without verbal communication. The crew worked through the systems failures and malfunctions by knowing the systems, procedures, and each other's duties, while using nonverbal communication when appropriate.

3.3.3 Analysis and discussion

Following the loss of *Columbia*, the STS-107 training records were reviewed and the crew instructors were questioned with respect to failure scenarios that would result in the crew members closing visors. This research established that the crew had experienced numerous simulation scenarios in training with procedures requiring lowered visors (e.g., smoke/fire, cabin leaks, broken window panes, contingency aborts, systems failures resulting in an in-flight bailout, etc.). In line with the standard training regimen, very few simulations were performed with the crews wearing the ACES. When performing simulations unsuited, the STS-107 crew would verbalize the suit-specific steps of the procedures.

The CDR, Mission Specialist 2 (MS2), and MS3 were veterans of previous space flights. The CDR, PLT, MS1, MS3, and Payload Specialist 1 (PS1) were all professional military pilots, and the MS2 and MS4 were experienced civilian pilots. All members of the *Columbia* crew were highly trained and outstandingly competent. The training regimen that they underwent emphasized systems knowledge and problem resolution through appropriate analysis of displays and the use of checklists. As was previously stated, unrecoverable conditions are not presented to a shuttle crew during training. It is likely that the STS-107 crew members did not close and lock their visors before cabin depressurization because they were focused on solving the problems that had been presented to them rather than on their own survival. Upon cabin depressurization, a survival situation would be immediately apparent from a physiological perspective. The fact that they still did not close and lock their visors indicates that they were rapidly incapacitated and unable to do so.

3.3.4 Training effectiveness case study

Analysis of switch positions on recovered control panels can reveal crew actions. This analysis can provide insight into the crew members' thought processes and motives, which reveals their knowledge of vehicle systems.

Following the 1986 loss of *Challenger*, a review of the recovered equipment showed that the crew took a few actions after the breakup and prior to losing consciousness.[4] During the *Columbia* CM reconstruction, the switch panels were examined to determine whether any switches were out of the expected positions. Of the recovered flight deck panels (highlighted green in figure 3.3-2), an estimated 10% to15% of the switches were found out of position with respect to the expected positions for the entry timeframe.

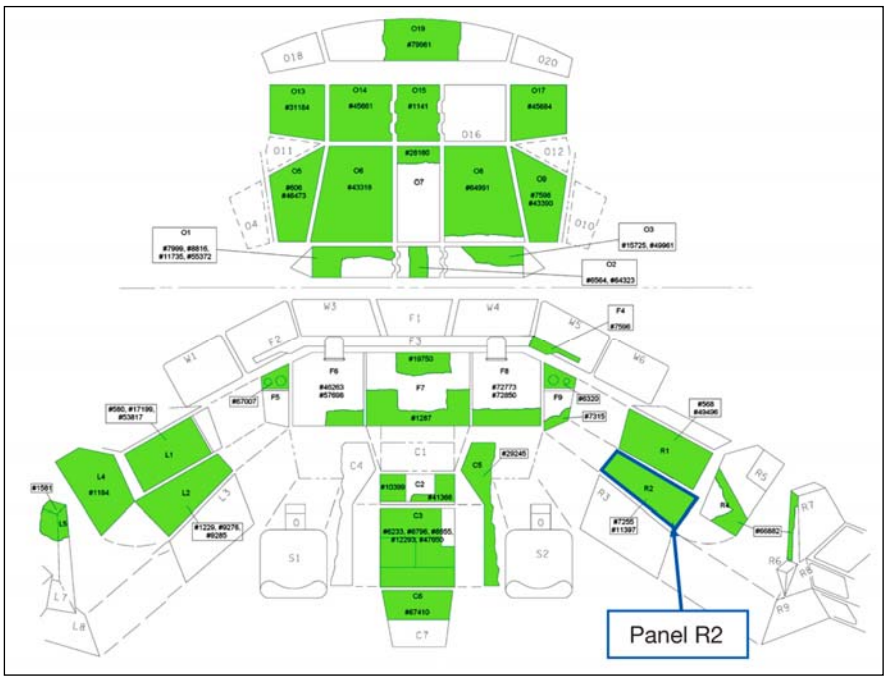


Figure 3.3-2. Columbia *recovered flight deck forward panels* (highlighted in green).

[4]Report from Dr. Joe Kerwin to Rear Adm Truly, http://history.nasa.gov/kerwin.html, July 28, 1986.

No further analysis was performed at that time as it was not possible to know whether the switch positions were due to crew action, the mishap, or handling during debris recovery. However, the R2 panel, to the immediate right of the PLT on the flight deck (figure 3.3-2), warranted further investigation. This panel contains the primary controls for the auxiliary power units (APUs). The APUs drive the hydraulic pumps that provide hydraulic pressure to the flight control surface actuators. During entry, these surfaces become increasingly important to vehicle control, and loss of hydraulic pressure can have catastrophic results. Reconstructed general purpose computer (RGPC)-2 data revealed that the hydraulic systems failed prior to GMT 14:00:03 while the crew was conscious and capable of taking action.

When recovered, the R2 panel was folded back in on itself (figure 3.3-3), protecting the innermost switches from manipulation during recovery operations. The switch positions were considered to be unaltered by external factors, making the switches valuable in determining crew actions.

The panel was pried open during the investigation and switch positions were reviewed. In figure 3.3-4, all out-of-position switches are outlined with a pink box and noted with a pink dot.

Figure 3.3-3. ***Recovered R2 panel from*** **Columbia.**

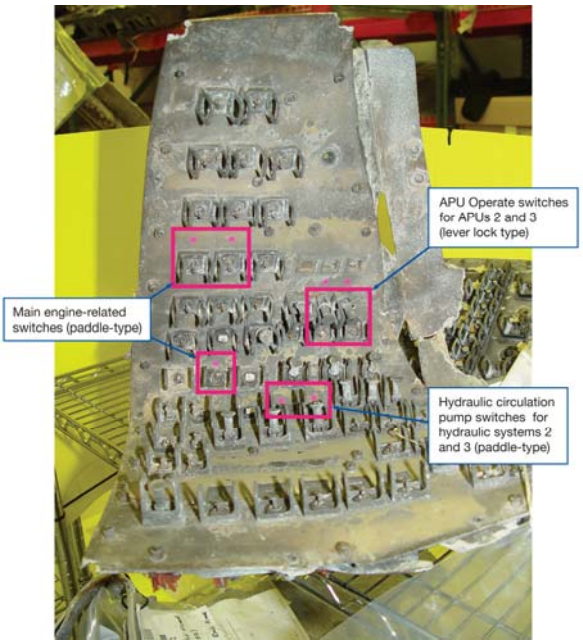


Figure 3.3-4. ***Recovered R2 panel from*** **Columbia** ***(after it was unfolded). Pink dots are above the out-of-configuration switches*****.**

The APU 2 and APU 3 Operate switches ("lever lock" switches that require two independent actions to change the switch position) were found in the "injector cool" position as opposed to the nominal "start/run" position. These switches are used when starting and stopping the APU. The "injector cool" position is used to cool down the APU after a shutdown prior to restarting it.

The remaining out-of-position switches are paddle-type switches that require just a push to move out of position. The switches on the left side of the panel are main engine-related switches that are not used during

entry.[5] The two paddle-type switches in the lower center of the panel are the circulation pump switches for hydraulic systems 2 and 3. These were found in the "on" position as opposed to the normal "off" position. The circulation pump is used for thermal conditioning of the hydraulic fluid while the orbiter is on orbit. The pump also is used to keep the hydraulic reservoir pressurized. This pump is not powerful enough to deploy the landing gear, but it can provide some hydraulic pressure if activated. The pump is neither used nominally on entry nor is it used in off-nominal procedures.

At the end of RGPC-2 (GMT 14:00:05), all three APUs were operating but the hydraulic systems pressures and quantities were zero, presumably due to a loss of hydraulic fluid from damage to the left wing. While the crew members could not know the reason for the low hydraulic pressure, they would know from training that a loss of hydraulic pressure would result in a vehicle LOC such as they were experiencing.

In response to a hydraulic system failure, the procedures require shutting down the APUs by placing the APU Operate switches in the "off" position, then moving the APU Operate switches into the "injector cool" position to cool down the APUs before attempting a restart of the APUs. RGPC-2 data indicate that the APU Operate and hydraulic circulation pump switches were in their nominal, expected positions.[6] Therefore, these switches changed position after GMT 14:00:05, 13 seconds prior to the Catastrophic Event (CE).

Because the R2 panel was recovered folded in half and the APU Operate switches were not accessible, it is concluded that these switch positions were not altered during recovery operations. While the possibility exists for the lever lock switches to move due to random debris-debris interaction, the requirement for specific physical actions to enable switch movement makes it much more probable that the PLT deliberately moved the switches in an effort to regain hydraulic pressure and control of the vehicle. The paddle switches for the circulation pumps would be more subject to movement due to debris contact. However, the switches that were out of position (for hydraulic systems 2 and 3) correspond to the same APUs (APUs 2 and 3), lending credence to the theory that the actions were deliberate.

The catastrophic events that led to the loss of *Columbia* are not simulated in training or covered by existing systems procedures. The crew's attempt to recover at least two APUs by selecting the "injector cool" position and, in the interim, providing some hydraulic pressure to the flight control surfaces through the use of the circulation pump demonstrates remarkable aplomb. Their effort to regain hydraulic pressure to recover vehicle control shows excellent knowledge of the orbiter systems and problem-resolution techniques. This also indicates that deliberate crew actions (such as manipulating specific switches) were possible for some period of time after GMT 14:00:05, indicating that the CM was still pressurized and the dynamics of the out-of-control vehicle were not incapacitating.

[5]These were paddle-type switches, so the positions could have been changed by any of the various methods described above. They are not relevant to entry systems, so no analysis was necessary.
[6]E-mail: from Jeff Kling, STS-107 Ascent/Entry Mechanical Maintenance and Crew Systems Officer, to Pam Melroy, November 8, 2005.

3.4 Crew Analysis

This section contains the analyses and results regarding what happened to the STS-107 crew. It encompasses the awareness that the crew had of events, crew actions in response to those events, and the events of lethal potential to which the crew was exposed. This analysis is meant to aid current and future spacecraft designers in developing vehicles and systems that incorporate the lessons learned from this accident.

The analysis is based on two types of data: objective data (e.g., medical forensic findings, on-board and downlinked vehicle instrumentation data, recovered on-board video data, and air-to-ground crew communications) and derived data (e.g., ballistics, thermal analysis, aerodynamic analyses, shock wave interactions, motion modeling, thermal injury mapping, and material testing). Although this section describes the best "data fit," it is subject to some inherent uncertainty due to the lack of data, both actual and experimental, on human exposure to conditions that are similar to the atmospheric entry environment.

Evidence indicates that the crew was aware of the LOC and was taking actions that were consistent with an attempt to recover hydraulic pressure. Once the depressurization event occurred, the crew was rendered unconscious or deceased and was unaware of the subsequent physical and thermal events. There is no evidence of crew error contributing to this accident.

[REDACTED.] Cause of death of the crew was unprotected exposure to high altitude and blunt trauma.

The first section discusses crew awareness. Next, medical findings are described by injury categories. A chronological sequence of the events with lethal potential is presented followed by a summary.

The following is a summary of the findings, conclusions, and recommendations from this section:

[REDACTED.]

> ***Conclusion L1-1.*** After loss of control at GMT 13:59:37 and prior to orbiter breakup at GMT 14:00:18, the *Columbia* cabin pressure was nominal and the crew was capable of conscious actions.
>
> ***Recommendation L1-4.*** Future suit design should incorporate the ability for crew members to communicate visors-down without relying on spacecraft power.

Finding. Tissue samples revealed evidence of ebullism.[1]

> ***Conclusion L1-3.*** The crew was exposed to a pressure altitude above 63,500 feet, indicating that the cabin depressurization event occurred above this altitude.

[1]Ebullism is defined as the formation of bubbles in bodily fluids under reduced environmental pressure.

Finding. The depressurization event occurred prior to the loss of circulatory function.

Finding. No conclusion could be drawn as to the rate of cabin depressurization based on medical evidence.

Conclusion A8-1. Spacecraft accidents are rare, and each event adds critical knowledge and understanding to the database of experience.

Recommendation A8. As was executed with *Columbia*, spacecraft accident investigation plans must include provisions for debris and data preservation and security. All debris and data should be cataloged, stored, and preserved so they will be available for future investigations or studies.

Finding. None of the six crew members wearing helmets closed their visors.

Conclusion L1-5. The depressurization incapacitated the crew members so rapidly that they were not able to lower their helmet visors.

Recommendation L1-3/L5-1. Future spacecraft crew survival systems should not rely on manual activation to protect the crew.

Finding. One crew member appears to have been restrained only by the shoulder harness and crotch strap.

Recommendation L1-2. Future spacecraft and crew survival systems should be designed such that the equipment and procedures provided to protect the crew in emergency situations are compatible with nominal operations. Future spacecraft vehicles, equipment, and mission timelines should be designed such that a suited crew member can perform all operations without compromising the configuration of the survival suit during critical phases of flight.

Finding. Injuries were consistent with the crews' upper bodies not being securely held to the seatbacks and with evidence indicating that the inertial reel straps were extended at the time of failure.

Finding. Injuries were consistent with the crews' upper bodies not being supported during the time of dynamic motion.

Conclusion L2-3. Lethal injuries resulted from inadequate upper body restraint and protection during rotational motion.

Recommendation L2-7. Design suit helmets with head protection as a functional requirement, not just as a portion of the pressure garment. Suits should incorporate conformal helmets with head and neck restraint devices, similar to helmet/head restraint techniques used in professional automobile racing.

Recommendation L2-8. The current shuttle inertial reels should be manually locked at the first sign of an off-nominal situation.

Recommendation L2-9. The use of inertial reels in future restraint systems should be evaluated to ensure that they are capable of protecting the crew during nominal and off-nominal situations without active crew intervention.

Finding. Crew members experienced traumatic injuries in areas corresponding to the seat restraint system.

Conclusion L3-4. The seat restraint system caused lethal-level injuries to the unconscious or deceased crew members when they separated from the seat.

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments.

Recommendation L3-1. Future vehicles should incorporate a design analysis for breakup to help guide design toward the most graceful degradation of the integrated vehicle systems and structure to maximize crew survival.

Finding. No significant levels of carbon monoxide or cyanide (combustion by-products) were identified in any of the body fluids.

Finding. There was no evidence of thermal injury to the respiratory tracts.

Conclusion L1-4. The crew was not exposed to a cabin fire or thermal injury prior to depressurization, cessation of breathing, and loss of consciousness.

[REDACTED.]

3.4.1 Crew awareness

3.4.1.1 *Preflight*

The crew members of STS-107 were placed in protective quarantine at the Johnson Space Center astronaut crew quarters on January 9, 2003 where their health was monitored by the assigned crew surgeons; no health issues were observed.

3.4.1.2 *Launch*

As reported in the CAIB Report, at 81.9 seconds mission elapsed time, post-launch video showed a piece of insulating foam striking the left wing of the orbiter. The remainder of the ascent phase went without incident; and the crew, which was unaware of the debris impact at this time, proceeded with the mission as planned.

3.4.1.3 *Orbital operations*

The *Columbia* orbiter performed satisfactorily on orbit and the crew worked well as a team, accomplishing all scientific goals. On Flight Day 8, the crew was notified via email about the foam strike, but was told it was "not even worth mentioning other than wanting to make sure that [the crew is] not surprised by it in a question from a reporter." The capsule communicator (CAPCOM)[2] also relayed that there was "no concern for [reinforced carbon-carbon] or tile damage" and that there was "absolutely no concern for entry". A video clip of the strike was included with the e-mail.[3] No changes in the mission profile were thought necessary or recommended by the shuttle Mission Management Team, and the entry was flown as originally planned.

[2]The main individual with whom astronauts on a flight communicate with on the ground at the MCC in Houston.
[3]*Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 36.

3.4.1.4 *Deorbit preparations*

The crew started the planned deorbit activities on Flight Day 16 (February 1, 2003). Per pre-mission planning, the crew began working items on the De-orbit Preparation checklist at GMT 09:15:30. At GMT 11:11:18, the flight deck crew entered a computer command (OPS 301) to initiate the *Pre-deorbit Coast* sequence. The Commander, Pilot, Mission Specialist 2, and Mission Specialist 4 seats were located on the flight deck. The Mission Specialist 1, Mission Specialist 3, and Payload Specialist 1 seats were located on the middeck. Figures 3.4-1 and 3.4-2 show the seating arrangements in the CM.

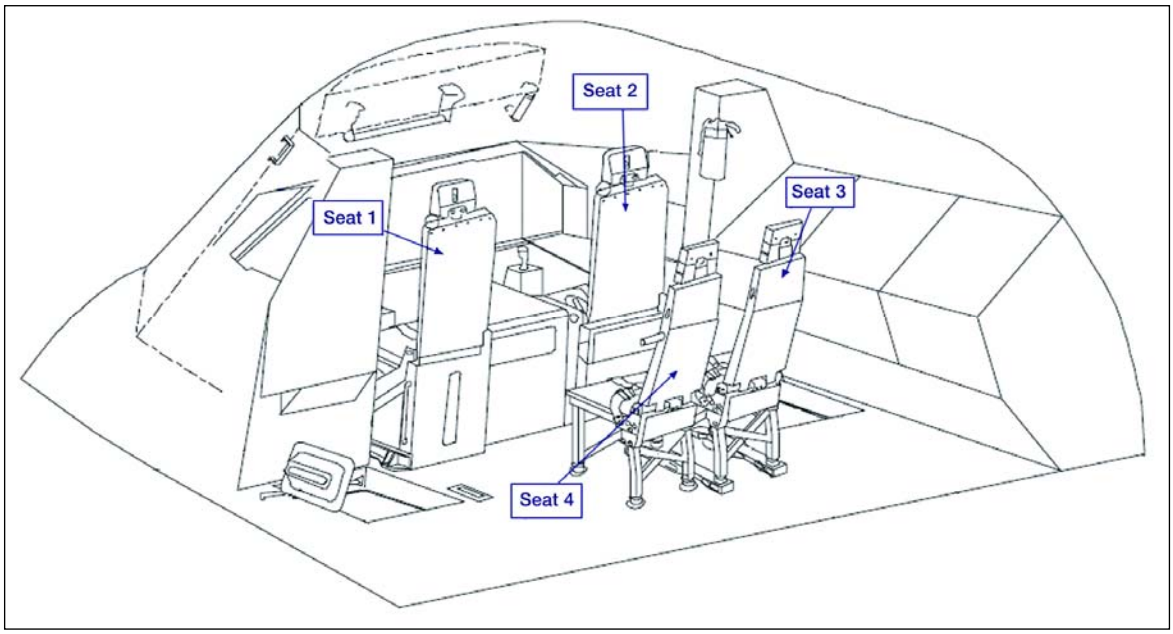


Figure 3.4-1. ***Depiction of the flight deck seats*****.**

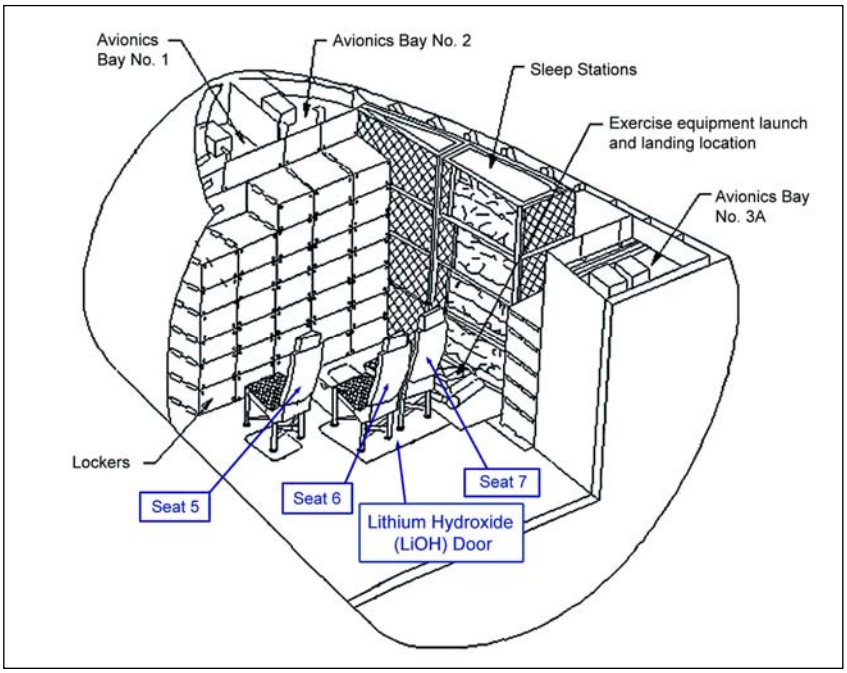


Figure 3.4-2. ***Depiction of the middeck seats*****. [Adapted from the Shuttle Crew Operations Manual]**

Recovered middeck video, documenting events from approximately GMT 11:40:00 to GMT 12:10:00, shows the middeck crew members donning their ACESs and preparing the middeck for return. Figure 3.4-3 is a video frame-capture from the beginning of the tape showing the crew members in various states of landing preparation.

At GMT 12:10:10, the video ends with the middeck crew starting the escape pole installation procedure. Two of three crew members seen in the video were wearing their ACESs, but their gloves and helmets were not mated (typically not performed until after the crew member is strapped in their seat) and one crew member had not yet donned the ACES.

3.4.1.5 *Entry*

The initial phase of entry went without incident. A recovered flight deck video (figure 3.4-4), which runs from approximately GMT 13:35:34 to GMT 13:48:45, provides insight into the crew events taking place on the flight deck.

Figure 3.4-3. ***Video frame-capture from the recovered middeck video.***

Figure 3.4-4. ***Video frame-capture from the recovered flight deck video.***

The video shows that, at GMT 13:36:04, the CDR bumped the RHC accidentally (figure 3.4-5). Movement of the RHC out of the centered position caused the digital autopilot (DAP) to "downmode" from "Auto" mode to "Inertial" mode. When this occurred, a "DAP DOWNMODE RHC" caution-and-warning message was displayed, the INRTL button on the C3 panel was illuminated, and a tone, which can be heard in the recovered flight deck video, was annunciated. An immediate reactivation of the autopilot was performed by the CDR. The CAPCOM in the MCC then requested the CDR to enter "another Item 27," which is a command to fully recover the vehicle attitude from the bumped RHC. The CDR complied, and is heard on the recovered videotape commenting that he had forgotten that he needed to perform an Item 27 after a stick bump but had noticed the guidance needles "weren't really down where they needed to be." This indicates that the CDR was scanning the displays, noticing that the guidance needles were not centered after the stick bump, and was properly processing the information.

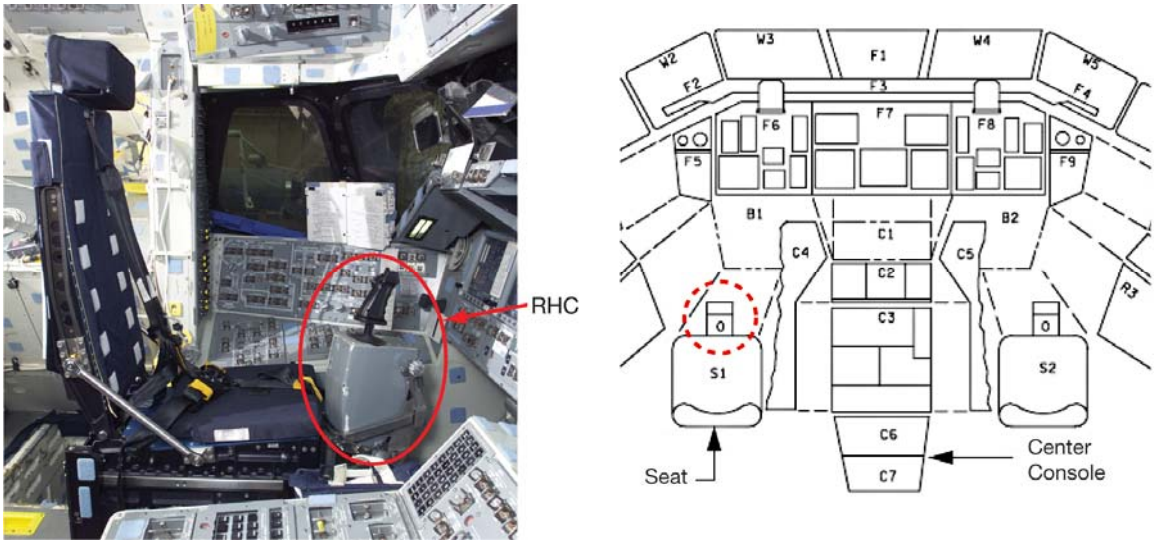


Figure 3.4-5. ***Location of the Commander's seat and the rotational hand controller.*** **[Left picture from a shuttle training mockup in the JSC Space Vehicle Mockup Facility, looking from starboard to port; right picture adapted from the Space Shuttle Systems Handbook, looking from aft to forward]**

At GMT 13:39:09, the CDR executed the OPS 304 command to load the computer software that was used to execute entry. The CDR and PLT both verified that the OPS 304 command was properly executed.

The CDR is next seen finishing a drink bag, as part of his required fluid-loading protocol,[4] and floating it back to a crew member for disposal. When the CDR went to pass the last drink bag to the crew member for disposal, that crew member requested that the CDR wait so that the crew member could finish donning the ACES gloves before the gravity levels increased. The video shows that the crew member had partially donned the gloves but did not mate the connecting rings to the ACES. Investigators, therefore, concluded that dealing with the disposal of the water bags and other loose items plus performing flight engineer duties (e.g., assisting the CDR and PLT with checklist items and throwing switches) caused the delay in configuring the ACES.

Recorded telemetry indicates that entry interface (EI) occurred at GMT 13:44:09. Upon observing the time cue for EI on the displays, the CDR states, "Just past EI." The crew then remarks on the flashes of plasma that are visible through the windows (these flashes are a normal part of entry). Shortly afterwards, the CDR requests that everyone perform suit integrity and communications checks. At GMT 13:45:24, three of the four flight deck crew members are observed performing successful communications and suit pressure integrity checks. One of the flight deck crew members could not participate in the suit pressure integrity check since that crew member's gloves were not completely donned at this time.[5] After completion of the check, a crew member asked the CDR whether the crew members were to keep their visors down after the test. The CDR replied, "No."

It should be noted that due to a limitation of the orbiter-suit system, the normal configuration for entry is with the visors up. If the crew keeps the visors down and the O_2 flowing for entry, the O_2 that is being vented from the ACES would increase the cabin O_2 concentration to a level that would violate the hazard controls for fire prevention. In addition, flown astronauts and crew trainers who were interviewed concerning this indicate that the visors also restrict the crews' field of vision and can interfere with nonverbal communication. With the visors down, inter-crew verbal communication is dependent on orbiter main power; there is no battery backup. A loss of power requires either verbal communications with the visor open or nonverbal communications, which can be hindered by having the visor down.

[4]Fluid loading is one of the medical countermeasures that is used to mitigate orthostatic intolerance due to the fluid shift and blood plasma level changes that are experienced during space flight and the return to a gravity environment.
[5]The gloves must be mated to the suit for the ACES to pressurize.

By the end of the recovered video at GMT 13:48:45, plasma is visible through the windows (this is normal for this phase of entry) and three of the four flight deck crew members are observed with their ACES suits and helmets on, visors open, and gloves mated and are seated with restraint harnesses on. One crew member was suited with helmet on, visor open, left glove on but not mated, and right glove off and was seated with restraint harness on. There was no indication that the crew was aware of any problems with the orbiter.

Analysis of the telemetry from the O_2 supply system at GMT 13:54:30 shows a signature that is consistent with a second suit pressure check and/or g-suit pressurization by three to five crew members. Since no video or audio was recovered from this timeframe, it is unknown which crew members performed this check. It is possible that the flight deck crew member who was not ready for the first suit and communications check participated in this one.

Between GMT 13:58:39 and GMT 13:58:56, four left tire pressure fault messages were recorded by the Backup Flight Software. These messages were annunciated on the crew displays and accompanied by an audio tone. The fault messages indicated a loss of pressure on the left main landing gear tires. These indications also were presented to the flight control team in the MCC. The CDR and PLT called up the fault page for these messages and reviewed the information. One of the failure scenarios that the crew practiced during training was a circuit breaker trip that resulted in one-half of the tire pressure sensors being disabled. A circuit breaker trip would disable some sensors for all of the tires (left main gear, right main gear, and nose gear), but the failure signature during the accident involved all of the tire pressure sensors on the left main gear only. So the indications that the crew saw would be familiar, although different from what they saw in training. At GMT 13:58:48, the crew began a call to the MCC but that call was broken and not repeated. Brief interruptions of communications often occur due to the tracking and data relay satellite antenna pointing angles changing relative to the orbiter's transceivers. This specific dropout of communication was expected.

At GMT 13:59:06, 10 seconds after the fourth of four tire pressure fault messages, telemetry indicated that the "LEFT MAIN GEAR DOWN" lock sensor transferred to "ON." Other sensors indicated that the landing gear door was still closed and the landing gear was locked in the "up" (stowed) position. These mixed signals caused the left landing gear position indicator to display a "barber pole" (figure 3.4-6), which indicates an indeterminate landing gear position. Post-accident analysis of the data and recovered debris indicates that the left landing gear was locked in the "up" position and the landing gear door was closed. The signal indicating that the gear was down was a false signal that was likely triggered by damage to the sensor system (sensor, wiring harness, etc.). Based on training experience, the crew was probably attempting to diagnose the situation given that it involved the same landing gear as the tire pressure messages and indicated a potential landing gear deployment problem.

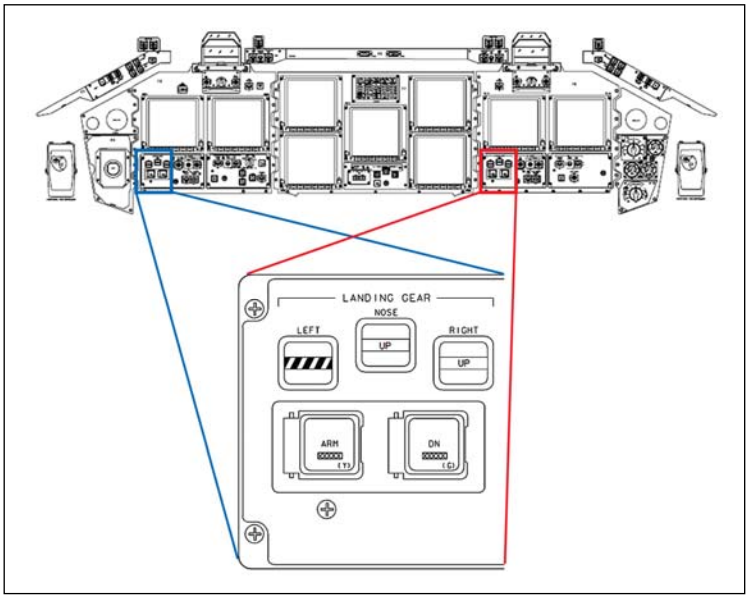


Figure 3.4-6. ***Landing gear indicator panel, identical on both sides of the flight deck forward display panels.*** **Left indicator showing "barber pole" (indeterminate position). [Adapted from the Space Shuttle Systems Handbook]**

Twenty-six seconds after the left main gear "talk-back" displayed the barber pole, the last audio transmission from the crew, "Roger, uh …," was received (GMT 13:59:32). The CAPCOM replied to the partial transmission to let the crew know that the flight controllers saw the tire pressure fault messages and did not "copy" the last transmission.

Analysis of RGPC data indicated that the Primary Avionics Software System recorded a fault message that was associated with the removal of Flight Control System (FCS) Channel 4 (CH4) from the control loop at GMT 13:59:33. This message would result in the annunciation of a Master Alarm. While there is no crew action associated with this frequently trained FCS fault message other than to perform a message reset, the crew likely called up a display to analyze the failure. Crews are trained to troubleshoot systems errors, and this crew would have been evaluating this new message along with the previous tire pressure and landing gear down-lock indications to assess whether there was a common system fault that could account for all of these messages.

3.4.1.6 *Loss of control*

Based on engineering analysis and modeling (see Section 2.1), hydraulic pressure, which is required to move the flight control surfaces, was lost at approximately GMT 13:59:37. At that time, the Master Alarm would have sounded for the loss of hydraulics and the crew would have become aware of a serious problem. It is probable that the loss of hydraulic pressure as a result of the damage to the left wing resulted in an uncontrolled pitch-up and loss of vehicle control. A visual simulation of the pitch-up associated with this LOC scenario is shown in figure 3.4-7.[6] The flight deck crew would have been the first to be aware of this owing to the changing light levels, the view of the horizon through the windows, and the information on the flight displays. Space-adapted crews are reported to be very sensitive to motions and G-loads. As the orbiter motion dynamics began to increase, all of the crew members likely would have sensed this motion and been aware of the off-nominal[7] situation. At GMT 13:59:46, a "Roll Ref" alarm message was annunciated, indicating that the orbiter had exceeded the limits of the entry drag profile. In conjunction with the hydraulics messages and the unusual motion of the orbiter, the "Roll Ref" message would have reinforced the fact that a serious problem had developed.

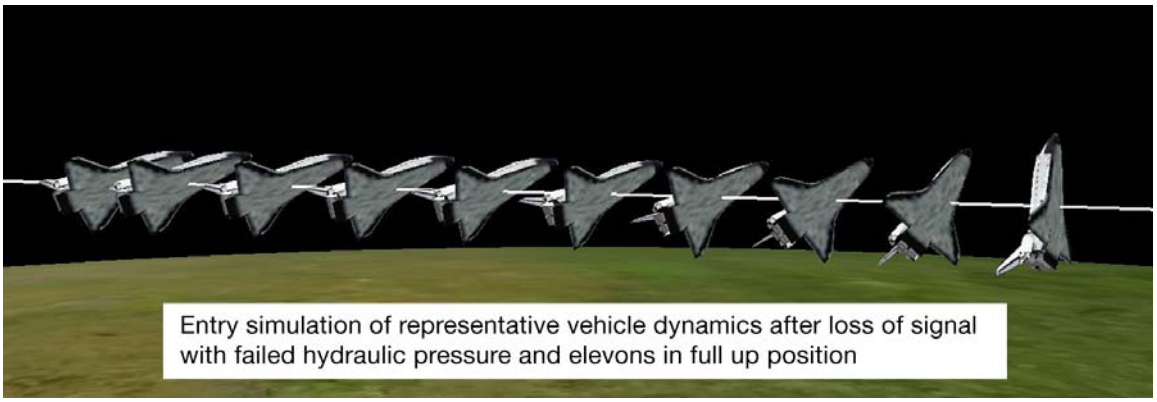


Figure 3.4-7. ***Sequence (1-second intervals) showing a simulation of orbiter loss of control pitch-up from GMT 13:59:37 to GMT 13:59:46*****. White line indicates vehicle trajectory relative to the ground.**

Based on the orbiter LOC entry simulation, the representative motion showed that the predominant orientation of the orbiter remained "belly-into-the-wind" with large excursions in pitch, roll, and yaw. This

[6]Vehicle dynamics are based on aerodynamic modeling using orbiter aerodynamic models and accelerometer data.
[7]Outside of acceptable limits.

motion can be characterized[8] as a slow (30 to 40 degrees per second), highly oscillatory spin. Analysis of the accelerations describes the overall motion of the crew as a swaying to the left and the right ($\pm Y_{Crew}$ axis, eyeballs right and left[9]) combined with a pull (deceleration) forward ($+X_{Crew}/-G_x$, eyeballs out) against the seat harness straps. Z-axis accelerations pushed the crew (vertically) down into the seat ($-Z_{Crew}/+G_Z$ axis, eyeballs down). Figure 3.4-8 shows a depiction of the sign convention and the resulting motion for accelerations.

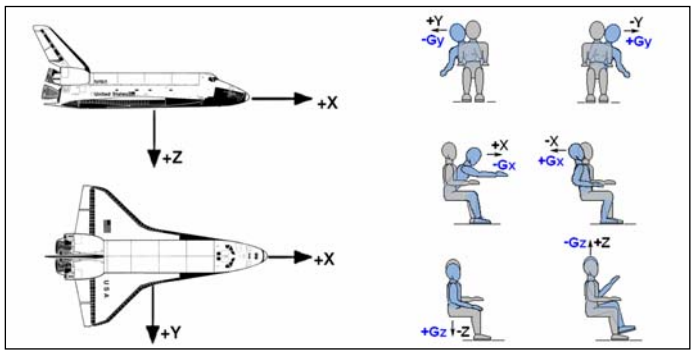


Figure 3.4-8. ***Depiction of the orbiter and crew member axes conventions.*** **(Note: Z axis sign convention for crew is opposite from orbiter.)**

Figure 3.4-9 shows representative loads based on modeling in all three axes, including the effects of increasing rotational loads. Models showed that accelerations were initially low, and peaked between 2 G and 3.5 G by the time of the CE (separation of the forebody from the midbody). The dashed black lines (upper and lower) on the chart indicate human performance limits based on NASA-STD-3000.[10] The representative loads, which are based on modeling, were well within these human performance limits.

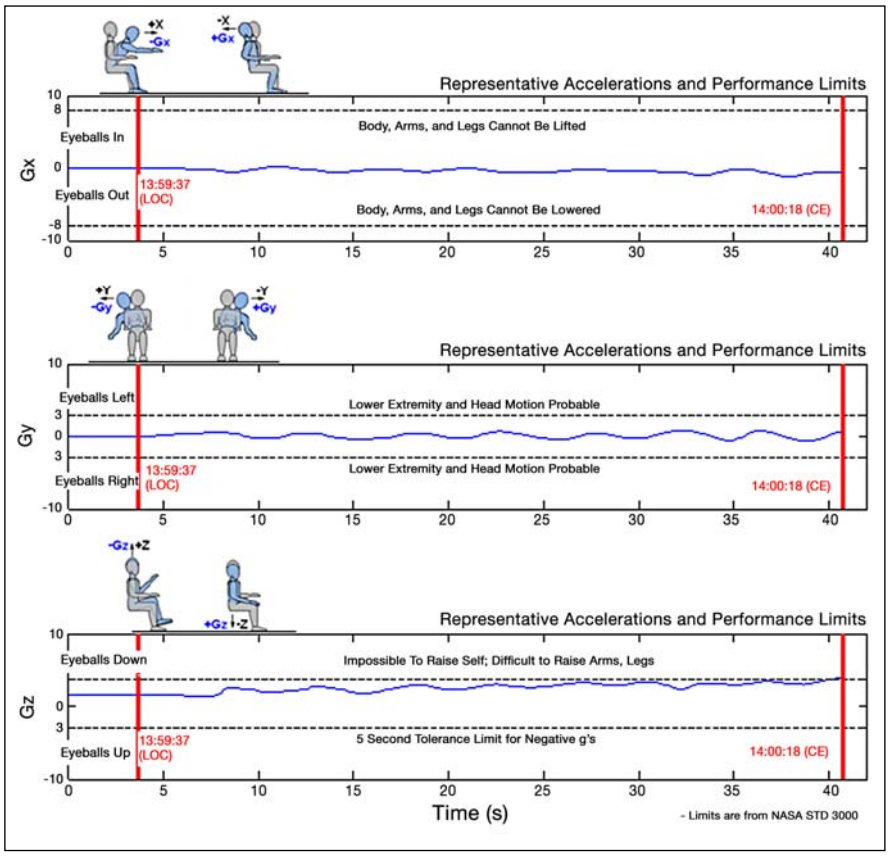


Figure 3.4-9. ***Representative accelerations from modeled motion analysis from loss of control to the Catastrophic Event*** **(see Section 2.1). Black dashed lines show human performance limits.**

[8]See descriptors in Flight Test Demonstration Requirements for Departure Resistance and Post-Departure Characteristics of Piloted Airplanes, Air Force MIL-F-83691B, Change 1, May, 31, 1996.
[9]In the crew axis convention, the physiological reaction is in the opposite direction of the acceleration vector; i.e., an acceleration in the $+X_{Crew}$ direction pushes crew members into their seat (i.e., forces the crew in the –X direction, resulting in a +Gx reaction, also known as "eyeballs in").
[10]NASA-STD-3000, *Man-Systems Integration Standards*, Volume I, Section 5, Revision B, 1995.

Post-flight analysis of crew equipment revealed that none of the six recovered seat restraint inertial reel mechanisms locked prior to failure. This resulted in the upper bodies of the crew members being unrestrained. Loads of these magnitudes and rates would not be expected to produce crew injuries or prevent the crew from performing most actions. However, these loads, augmented by the loose harness configuration and mass of the crew worn equipment, would require the crew members to brace themselves.

As the LOC scenario progressed, the dynamic motion environment would be expected to increase the susceptibility to motion sickness and disorientation, particularly in those who had no visual reference (i.e., those on the middeck) or who were novice space flyers.

Based on seat debris and medical analyses, one crew member was not fully restrained before loss of consciousness. Only the shoulder and crotch straps of this crew member appear to have been connected. The normal sequence for strap-in is to attach the lap belts to the crotch strap first, followed by the shoulder straps. Analysis of the seven recovered helmets indicates that this same crew member was the only one not wearing a helmet. Additionally, this crew member was tasked with post-deorbit burn duties. This suggests that this crew member was attempting to become seated and restrained when the LOC dynamics began. Given the motion of the orbiter, the lap belts hanging down between the closely spaced seats would have been difficult to locate and grasp.

RGPC data indicate that between GMT 14:00:02 and GMT 14:00:04, the vehicle was experiencing a right yaw rate of at least 21 deg/sec, which was the sensor limit, and a right roll rate of 7 deg/sec, followed by a left roll rate of 23 deg/sec that was associated with a nose-down pitch rate of 5 deg/sec (because of possible inertial measuring unit saturation, these values may be inaccurate). All available data indicate that the crew cabin environment (temperature, atmosphere) and systems (APUs, fuel cells, lighting, etc.) were still generally nominal; however, the hydraulic pressures and quantities were indicating zero.

RGPC data show that a message reset was performed by the CDR or PLT sometime between GMT 13:59:37.4 and GMT 14:00:05. This action is a normal crew response to a fault message and requires a crew member to manually acknowledge the message by keyboard entry on the center panel. RGPC-2 data indicate that the RHC was moved beyond neutral sometime between GMT 14:00:01.7 and GMT 14:00:03.6, triggering a “DAP DOWNMODE RHC” message at GMT 14:00:03.637. This message was likely due to a crew member bumping the RHC out of the null position due to the oscillatory motion of the orbiter. At GMT 14:00:03.678, the orbiter autopilot was returned to the AUTO mode. Returning the DAP to AUTO mode requires either the CDR or the PLT to press one or two buttons that were located on the glare shield. These actions indicate that the CDR or the PLT was still mentally and physically capable of processing display information and executing commands, and that the orbiter dynamics were still within human performance limitations.

Recovered debris revealed that the APU Operate switches on flight deck panel R2 were in positions that were consistent with an attempted restart of two of the three APUs[11] (figures 3.4-10 and 3.4-11). The hydraulic circulation pump paddle switches for the two hydraulic systems corresponding to the two APUs were also turned on. While turning on the hydraulic circulation pump is not in any crew emergency checklist, the pump can provide some hydraulic pressure, and this action shows good systems knowledge by the crew members as they responded to the limited information presented to them and worked to restore orbiter control. The APU Operate switches are “lever-lock” switches that require three actions to change the position. They must be (1) pulled outwards to disengage the switch lever from the lock, (2) moved to the desired position, and (3) released (figure 3.4-12). The switches are spring-loaded to hold them in the detents. The RGPC data indicate that all of these switches were in the nominal configuration up to GMT 14:00:04.826. These findings strongly suggest that despite the very dynamic vehicle motion, the PLT was still capable of taking appropriate actions to attempt a recovery of the hydraulic pressure by performing an APU restart at some time after GMT 14:00:04.826. Based on the panel R2 switch throws and the lack of visors being lowered, it is probable that the crew never realized that the vehicle LOC situation was unrecoverable and had become a survival situation.

[11]APUs supply the hydraulic pressure to the flight control surfaces.

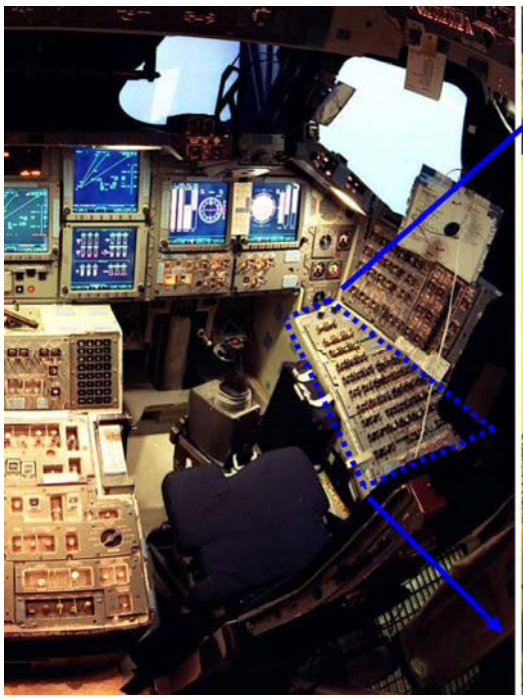


Figure 3.4-10. ***Location of the R2 panel.*** **[Picture from the Shuttle Training Simulator]**

Figure 3.4-11. ***Recovered R2 panel from*** **Columbia** ***(after it was unfolded).*** **Pink dots are above the out-of-configuration switches.**

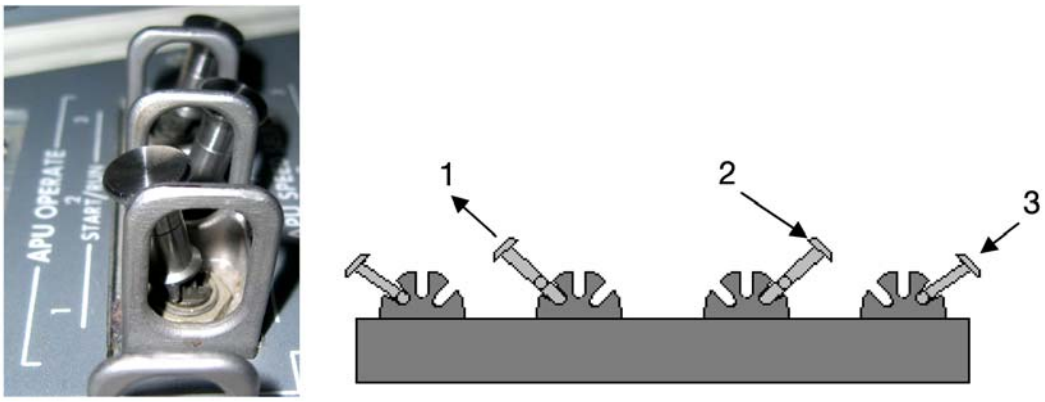


Figure 3.4-12. ***Operation of the lever-lock controller switches*** **(side view).**

Analysis of several sources of information indicates that the forebody separated from the midbody at or shortly after GMT 14:00:18. It is unknown what accelerations occurred during separation of the orbiter forebody from the midbody; however, ballistic analysis estimates that the translational G that was experienced by the orbiter forebody at the CE decreased from approximately 3.5 G to 1 G. It is also likely that there were additional translational and rotational loads acting on the crew at this time. Analysis of structural debris supports that multiple small impacts occurred between the forward fuselage (FF) and the CM, including in the area of middeck Volume E (figures 3.4-13a and 3.3-13b).

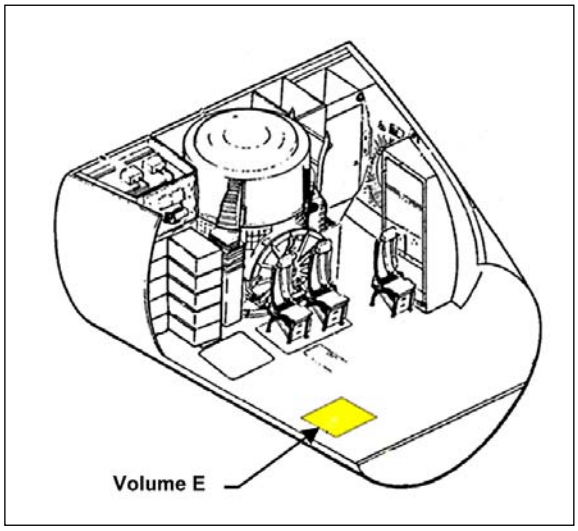
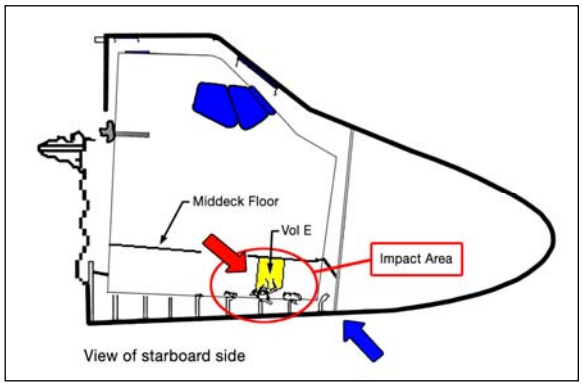


Figure 3.4-13a. ***View of middeck floor and Volume E, looking aft.***

Figure 3.4-13b. ***Scenario showing how the crew module pressure vessel could impact the forward fuselage, and the middeck Volume E could impact the crew module pressure vessel, with resultant damage.***



[REDACTED.]

[REDACTED. Figures 3.4-14a through 3.4-13d.]

[REDACTED.]

Conclusion L1-1. After loss of control at GMT 13:59:37 and prior to orbiter breakup at GMT 14:00:18, the *Columbia* cabin pressure was nominal and the crew was capable of conscious actions.

When the vehicle forebody separated from the rest of the vehicle, all resources from the midbody were lost, including power from the fuel cells. This resulted in the loss of all powered lighting, crew displays, radio, intercom, ventilation, and main O_2 supply. The flight deck would still have had light entering the cabin from the windows as well as from the activated chemical light sticks on each arm of the ACES and positioned throughout the cabin. The middeck would have been in total darkness except for some light filtering through the two inter-deck openings and from the activated chemical light sticks. This would indicate a survival situation.

Recommendation L1-4. Future suit design should incorporate the ability for crew members to communicate visors-down without relying on spacecraft power.

It was concluded that the crew was incapacitated and incapable of action at or shortly after the CE. As a survival situation, one of the first crew actions that would be expected after the CE would be for the crew to manually lower their visors and turn on their EOS. As detailed in Section 3.2, none of the crew members wearing helmets closed their visors. The accelerations derived from the representative motion modeling (figure 3.4-10) would not have prevented this action. Since they did not, it was concluded that the crew members were incapacitated due to other factors. This will be discussed in the following section.

This concludes the discussion of crew awareness. Injury classifications are discussed next.

3.4.2 Injury classifications

3.4.2.1 *Exposure to high altitude*

[REDACTED.] Given the level of tissue damage, the crew could not have regained consciousness even with re-pressurization. Survival was possible, but not likely, even with immediate and extensive medical intervention at this point. Although respiration would cease after depressurization, circulatory functions can exist for a short period of time.

[REDACTED.]

Finding. Tissue samples revealed evidence of ebullism.

> ***Conclusion L1-3.*** The crew was exposed to a pressure altitude above 63,500 feet, indicating that the cabin depressurization event occurred above this altitude.

[REDACTED.]

Finding. The depressurization event occurred prior to the loss of circulatory function.

There is very limited data on human exposure to space-equivalent vacuum. **[REDACTED.]** Although the *Soyuz 11* cabin depressurization was relatively slow (reportedly taking more than 3.5 minutes to depressurize to 0 psi), it was stated that the depressurization was fatal to the crew in roughly 30 seconds.[12] Because the exact scenario cannot be positively identified, no conclusions with respect to the rate or timing of cabin depressurization can be made from the medical findings. **[REDACTED.]**

Depressurization events in aviation have led to extensive studies on "time of useful consciousness (TUC)." TUC is generally based on the remaining amount of O_2 in the tissues that is permitting brain functions to continue. Various factors affect the TUC (i.e., exertion, depressurization rate, pre-exposure O_2 partial pressure, G-loads, adrenaline loading, etc.). Since the shuttle cabin uses air, the pre-exposure O_2 partial pressure was only 21% O_2 (the normal for sea-level). Based on debris and structural evidence, the most likely time for the initiation of cabin depressurization was at orbiter breakup (CE) at GMT 14:00:18. Based on video evidence, the depressurization was complete no later than GMT 14:00:59 (figure 3.4-15), and likely much earlier (see Section 2.3). This corresponded to an altitude range of 181,000 feet to approximately 140,000 feet. Traditional aviation TUC would correlate a rapid depressurization at these altitudes to a TUC of 12 seconds.[13] This would have been enough time for the crew to close their visors and initiate O_2 flow, and yet they did not (see Section 3.2).

[12] "A History of the Apollo-Soyuz Project, Midterm Review." http://history.nasa.gov/SP-4209/ch8-2.htm.
[13] *Joint Aerospace Physiology, Air Education and Training Command/Bureau of Medicine and Surgery*, February 1998.

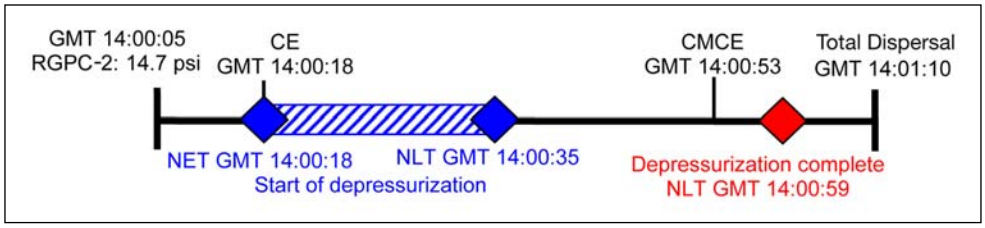


Figure 3.4-15. ***Cabin depressurization timeline.***

However, additional research discussed in *Joint Aerospace Physiology, Air Education and Training Command/Bureau of Medicine and Surgery* shows that the physiological response to hypoxia during a rapid depressurization event at this extreme altitude (181,000 feet) would have reduced the conventional TUC interval by 50% (i.e., 12 seconds would have been reduced to 6 seconds). In addition to the depressurization effects, the physical exertion against the G-forces that the crew experienced at this time would further reduce the available metabolic O_2 reserves[14] and increase the CO_2 partial pressure. Also, NASA research data indicate that de-conditioned crews have a reduced tolerance to G-loads.[15] Further, anecdotal reports from accidental exposure to vacuum confirm much shorter periods of awareness as reported by survivors.[16]

The 51-L *Challenger* accident investigation showed that the *Challenger* CM remained intact and the crew was able to take some immediate actions after vehicle breakup, although the accelerations experienced were much higher as a result of the aerodynamic loads (estimated at 16 G to 21 G[17]). The *Challenger* crew became incapacitated quickly and could not complete activation of all breathing air systems, leading to the conclusion that an incapacitating cabin depressurization occurred.[18] By comparison, the *Columbia* crew experienced lower loads (~3.5 G) at the CE. The fact that none of the crew members lowered their visors[19] strongly suggests that the crew was incapacitated after the CE by a rapid depressurization.

From this time forward, the crew members would have been unconscious, totally unaware of events, and unable to brace against the loads. With the configuration of the ACES (i.e., visors up and three crew members without gloves donned), the depressurization was an event of lethal potential. Had the ACES been configured with the visors down and locked, gloves on, and EOS activated, the depressurization event by itself probably would have been survivable.

Finding. No conclusion could be drawn as to the rate of cabin depressurization based on medical evidence.

Finding. None of the six crew members wearing helmets closed their visors.

Conclusion L1-5. The depressurization incapacitated the crew members so rapidly that they were not able to lower their helmet visors.

Recommendation L1-3/L5-1. Future spacecraft crew survival systems should not rely on manual activation to protect the crew.

Conclusion A8-1. Spacecraft accidents are rare, and each event adds critical knowledge and understanding to the database of experience.

[14]Naval Aviation Survival Training Program, G-Tolerance Brief: *G-Tolerance Improvement Program*, 2004.
[15]K. V. Kumar and W. T. Norfleet, "Issues on Human Acceleration Tolerance After Long-Duration Space Flights," NASA Technical Memorandum 104753, October 1992.
[16]Description of Altitude Chamber Mishap, *Roundup*, Volume 6, No. 6, Jan. 6, 1967.
[17]JSC-22175, STS-51L, JSC Visual Data Analysis Sub-Team Report, Appendix D9, June 1986.
[18]Report from Dr. Joe Kerwin to Rear Adm. Truly, http://history.nasa.gov/kerwin.html, July 28, 1986.
[19]See Section 3.2 Crew Worn Equipment.

Recommendation A8. As was executed with *Columbia*, spacecraft accident investigation plans must include provisions for debris and data preservation and security. All debris and data should be cataloged, stored, and preserved so they will be available for future investigations or studies.

3.4.2.2 *Mechanical injuries*

Mechanical injuries were isolated to the period of time at which they most likely occurred, based on engineering analyses of motions and accelerations.

Pre-Catastrophic Event
[REDACTED.]

Catastrophic Event to Crew Module Catastrophic Event
A very dynamic motion environment existed after the CE (GMT 14:00:18); this environment became more intense as the CMCE (the breakup of the forebody) approached at GMT 14:00:53. Figure 3.4-16 shows representative loads on the unconscious or deceased crew members based on aerodynamic modeling of the forebody dynamics post-CE. The black dashed lines showing human performance limits[20] are for conscious crew members. Based on the conclusion that the rapid depressurization occurred at or close to the time of the orbiter forebody separation, the crew was unconscious or deceased and unable to brace against these loads.

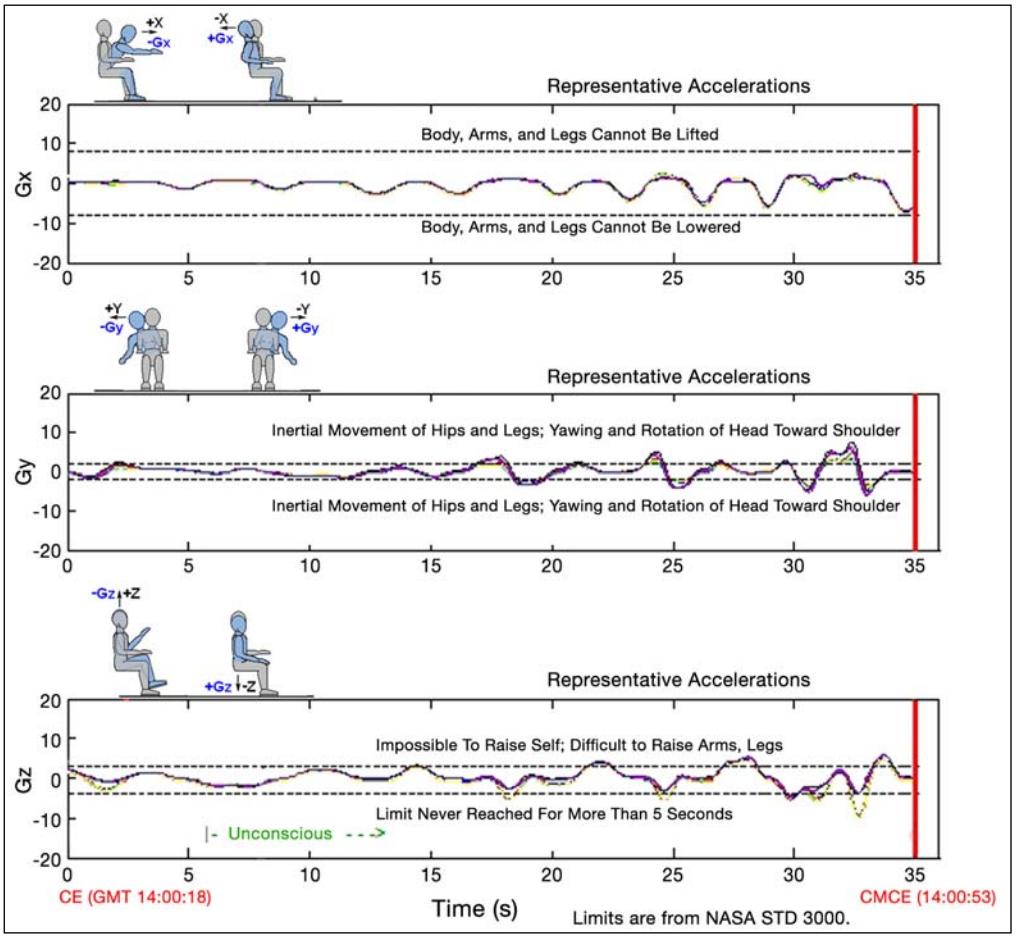


Figure 3.4-16. ***Representative acceleration profiles from the orbiter breakup (Catastrophic Event) to the orbiter forebody breakup (Crew Module Catastrophic Event) based on aerodynamic modeling*****. Black dashed lines show human performance limits.**

[20]NASA-STD-3000, *Man-Systems Integration Standards*, Volume I, Section 5, Revision B, 1995.

For the first 15 to 20 seconds, the modeled loads would not cause serious injuries to a conscious crew member who was capable of active bracing. An unconscious or deceased crew member would have been more susceptible to injury.

The crew is normally restrained in the seats by a five-point harness system (figure 3.4-17). A lap belt secures the lower torso. A crotch strap prevents "submarining."[21] Two shoulder harnesses, which attach to an inertial reel via the inertial reel strap, secure the upper torso.

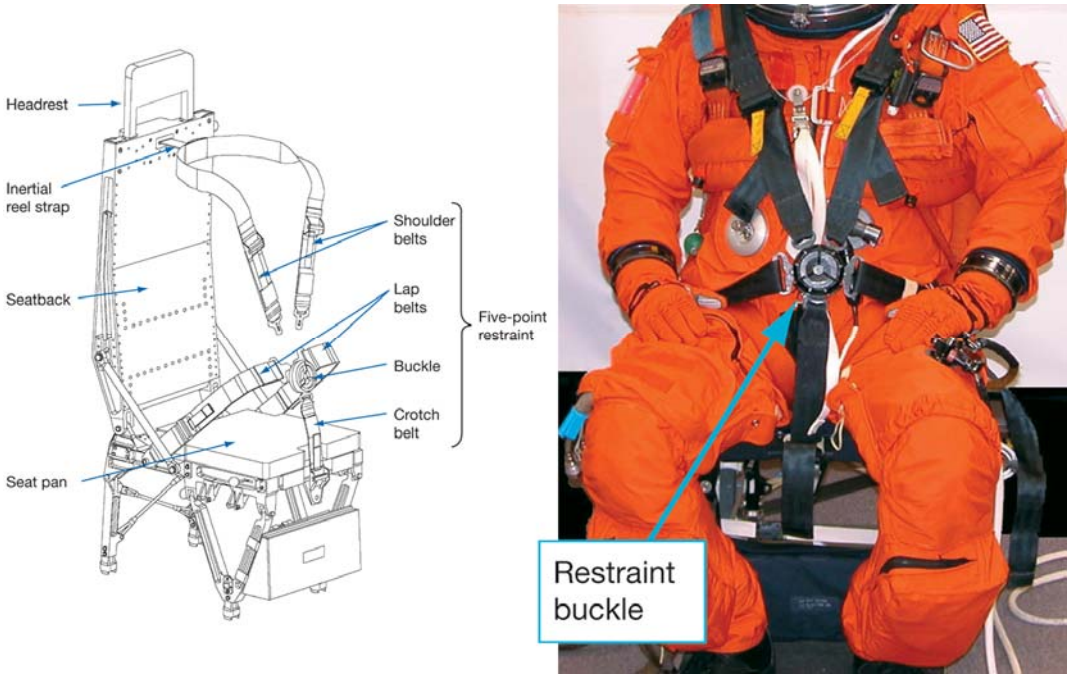


Figure 3.4-17. ***Detail of the five-point harness.***

Engineering analysis of the STS-107 restraints indicates that most of the inertial reel straps were extended and did not lock or retract prior to failure of the straps. The inertial reels are normally unlocked to allow the crew to access displays and controls with a full range of motion. With the inertial reels unlocked, the crew members' upper bodies were left unrestrained during the forebody dynamics. **[REDACTED.]**

Finding. One crew member appears to have been restrained only by the shoulder harness and crotch strap.

> ***Recommendation L1-2.*** Future spacecraft and crew survival systems should be designed such that the equipment and procedures provided to protect the crew in emergency situations are compatible with nominal operations. Future spacecraft vehicles, equipment, and mission timelines should be designed such that a suited crew member can perform all operations without compromising the configuration of the survival suit during critical phases of flight.

[REDACTED.]

Figure 3.4-18 provides a demonstration of an integrated seat/suit/crew member in entry configuration.

[REDACTED.] Figure 3.4-19 shows an interior view of an intact, pristine ACES helmet demonstrating exposed hardware. **[REDACTED.]**

[21] "Submarining" is when the occupant slides forward and beneath the lap belt.

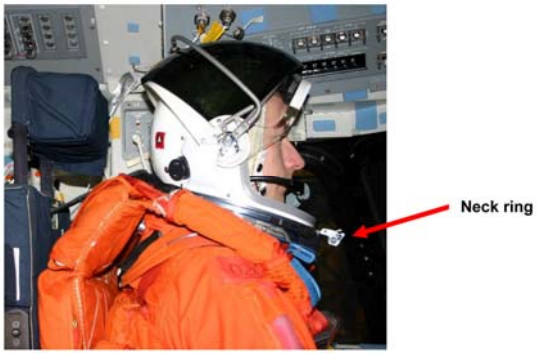


Figure 3.4-18. ***Demonstration of an integrated seat/suit/crew member in entry configuration.***

Figure 3.4-19. ***Example of an intact, pristine advanced crew escape suit nonconformal helmet (not an STS-107 helmet).***

[REDACTED. Figure 3.4-20.]

Finding. Injuries were consistent with the crew's upper bodies not being securely held to the seatbacks and with the evidence indicating that the inertial reel straps were extended at the time of failure.

Finding. Injuries were consistent with the crew's upper bodies not being supported during the time of dynamic motion.

Conclusion L2-3. Lethal injuries resulted from inadequate upper body restraint and protection during rotational motion.

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments.

Recommendation L2-7. Design suit helmets with head protection as a functional requirement, not just as a portion of the pressure garment. Suits should incorporate conformal helmets with head and neck restraint devices, similar to helmet/head restraint techniques used in professional automobile racing.

Recommendation L2-8. The current shuttle inertial reels should be manually locked at the first sign of an off-nominal situation.

Recommendation L2-9. The use of inertial reels in future restraint systems should be evaluated to ensure that they are capable of protecting the crew during nominal and off-nominal situations without active crew intervention.

Crew Module Catastrophic Event

[REDACTED.] Engineering and ballistic analyses of the orbiter forebody failure indicate that the mid-deck separated prior to the flight deck. Crew members on the middeck separated along with the middeck accommodations rack, middeck lockers, sub-floor components, and Modular Auxiliary Data System/orbiter experiment data recorder – a scenario that is supported by debris plots (see Section 2.2). Based on structural design analysis, thermal damage, and position in the debris field, the flight deck "pod" and the CM aft bulkhead stayed intact for a longer time. The location of the recovered flight crew equipment, which is plotted in figure 3.4-21, supports the middeck departing prior to the flight deck.

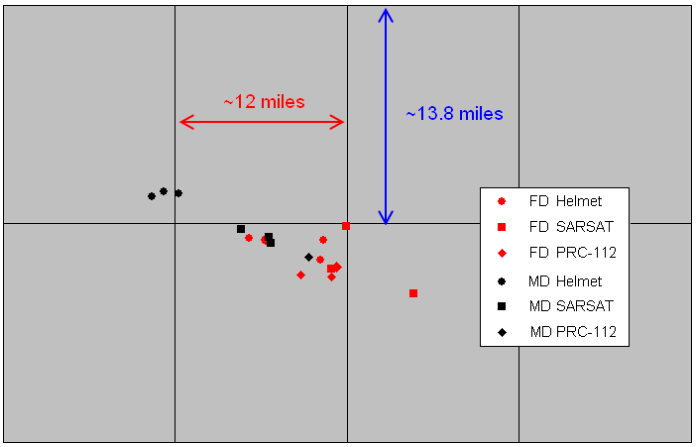


Figure 3.4-21. ***Ground location of the recovered flight crew equipment.***

[REDACTED.]

[REDACTED. Figure 3.4-22.]

[REDACTED.]

Finding. Crew members experienced traumatic injuries in areas corresponding to the seat restraint system.

Conclusion L3-4. The seat restraint system caused lethal-level injuries to the unconscious or deceased crew members when they separated from the seat.

Recommendation L2-4/L3-4. Future spacecraft suits and seat restraints should use state-of-the-art technology in an integrated solution to minimize crew injury and maximize crew survival in off-nominal acceleration environments.

Recommendation L3-1. Future vehicles should incorporate a design analysis for breakup to help guide design toward the most graceful degradation of the integrated vehicle systems and structure to maximize crew survival.

3.4.2.3 *Thermal exposure*

[REDACTED.]

Finding. No significant levels of carbon monoxide or cyanide (combustion by-products) were identified in any of the body fluids.

Finding. There was no evidence of thermal injury to the respiratory tracts.

> ***Conclusion L1-4.*** The crew was not exposed to a cabin fire or thermal injury prior to depressurization, cessation of breathing, and loss of consciousness.

[REDACTED.]

[REDACTED. Figure 3.4-23.]

[REDACTED.]

The ambient absolute pressure condition at separation was approximately 0.03 psi.

3.4.3 Identified events with lethal potential

1. **The first event with lethal potential was depressurization of the CM, which started at or shortly after orbiter breakup.** Existing crew equipment protects for this type of lethal event, but operational practices and hardware limitations were such that the ACESs were not in a protective configuration. The current shuttle ACES relies on the crew to lower and lock the visor; therefore, complete protection from a depressurization event depends on a permissive environment. Design solutions that do not require crew action are achievable.

2. **The second event with lethal potential was unconscious or deceased crew members exposed to a dynamic rotating load environment with nonconformal helmets and a lack of upper body restraint.** Current shuttle seat and helmet design and operational practices did not protect the crew members from this lethal event. Complete strap-in with inertial reels locked would reduce the risk of injury/death; however, even in this configuration, the current seat-suit restraint system provides limited protection from dynamic G events (i.e., no lateral restraints, no control of extremity motion, and no head-neck support). Better restraint designs that include head-neck support (i.e., conformal helmets), extremity control, and spine support are achievable to reduce the risk of injury/death.

3. **The third event with lethal potential was separation from the crew module and the seats with associated forces, material interactions, and thermal consequences. This event is the least understood due to limitations in current knowledge of mechanisms at this Mach number and altitude. Seat restraints played a role in the lethality of this event.** Although the seat restraints (e.g., narrow width) played a significant role in the lethal mechanical injuries, there is currently no full range of equipment to protect for this event. The event was not survivable by any means currently known to the investigative team, with the exception of ensuring the integrity of the CM until the airspeed and altitude are within survival limits. This is not possible for the current space shuttle design; however, future vehicle designs incorporating a principle of "graceful degradation" and CM stabilization are possible.

4. **The fourth event with lethal potential was exposure to near vacuum, aerodynamic accelerations, and cold temperatures.** Although current crew survival equipment may be capable of protecting the crew, it is not certified to protect the crew above 100,000 feet. At the altitude and speeds at which the unconscious or deceased crew members departed from the CM, the environmental risks include lack of O_2, low atmospheric pressure, high thermal loads as a result of deceleration from high Mach numbers, shock wave interactions, aerodynamic accelerations, and exposure to cold temperatures. Existing shuttle

CEE is certified to protect up to 100,000 feet and 600 KEAS; however, the ACES is not designed to provide protection from high-temperature exposures. Anecdotal evidence from the survival of the pilot of an SR-71 mishap [*Aviation Week & Space Technology*, August 8, 2005, pp. 60–62] suggests that an intact, pressurized suit similar to the ACES can protect a crew member at an altitude of 78,000 feet and speeds of at least Mach 3 (~400 KEAS). More research is needed to close the survival gap. The only protection that is achievable is to ensure the integrity of the CM until the airspeed and altitude are within suit capability, which is currently not precisely determined.

5. **The final event with lethal potential was ground impact.** Existing shuttle CEE protects for ground impact with a parachute. However, the crew member must manually initiate the parachute opening sequence, or the parachute must be used in conjunction with the crew escape pole of the shuttle to initiate the parachute automatic opening sequence. Military and sport parachuting solutions exist for opening parachutes independent of crew action.

3.4.4 Synopsis of crew analysis

The crew was unaware of an impending survival situation prior to the LOC. At the time of LOC, the flight deck crew was probably troubleshooting the caution-and-warning messages that were associated with the FCS fault, left main landing gear talk-back, and tire pressure messages. One of the middeck crew members was likely attempting to become seated and restrained under the dynamic LOC conditions. Until the fore-body separated from the orbiter vehicle, the crew was conscious and had not suffered serious injuries. Cause of death was unprotected exposure to high-altitude conditions and blunt trauma.

[REDACTED.]

Chapter 4 – Investigative Methods and Processes

4 Investigative Methods and Processes

This chapter discusses the methods and processes that were used during the Spacecraft Crew Survival Integrated Investigation team (SCSIIT) investigation. The SCSIIT activity was a continuation of the Crew Survival Working Group (CSWG), which was formed during the *Columbia* Accident Investigation Board (CAIB) investigation. The SCSIIT structure, personnel, and investigative process and lessons learned from that process are presented. The remainder of the chapter documents the methods, processes, and tools used by the SCSIIT for various analyses.

The following is a summary of the findings, conclusions, and recommendations for this section. Some recommendations are targeted at improving future NASA spacecraft crew survival accident investigation processes. Some are general suggestions for other organizations that may be tasked with such investigations in the future. In some cases, certain findings and conclusions reflect existing NASA policies and practices that were considered particularly effective and are included for emphasis for future investigators.

Finding. NASA priorities put emphasis on Return to Flight recommendations, long-term recommendations, and observations, in that order. As a result, the SCSIIT effort suffered from low priority relative to other program recovery efforts. Team members had to divide their time between the investigation work and the work for their home organization. This led to delays in completing the SCSIIT work and, in some cases, significant decrease in availability or complete loss of members of the SCSIIT.

Finding. Formally trained NASA-designated accident investigation personnel were not available for inclusion on the SCSIIT due to the intensity of safety and mission assurance work related to Return to Flight activities. SCSIIT members were selected primarily based on their technical knowledge and experience as well as availability. Many SCSIIT members did not have formal accident investigation training. The team preparation training sessions did not include the lengthy accident investigation training that is normally provided to NASA-designated investigators.

> ***Recommendation A1.*** In the event of a future fatal human space flight mishap, NASA should place high priority on the crew survival aspects of the mishap both during the investigation as well as in its follow-up actions using dedicated individuals who are appropriately qualified in this specialized work.

> ***Recommendation A4.*** Due to the complexity of the operating environment, in addition to traditional accident investigation techniques, spacecraft accident investigators must evaluate multiple sources of information including ballistics, video analysis, aerodynamic trajectories, and thermal and material analyses.

Finding. It was not uncommon to find several versions of documents supporting CAIB and CSWG work.

> ***Recommendation A6.*** Standard templates for accident investigation data (document, presentation, data spreadsheet, etc.) should be used. All reports, presentations, spreadsheets, and other documents should include the following data on every page: title, date the file was created, date the file was updated, version (if applicable), person creating the file, and person editing the file (if different from author).

Finding. Concerns about public release of sensitive information relative to the crew creates obstacles to the performance of crew survival investigations.

Recommendation A2. Medically sensitive and personal debris and data should always be available to designated investigators but protected from release to preserve the privacy of the victims and their families.

Recommendation A3. Resolve issues and document policies surrounding public release of sensitive information relative to the crew during a NASA accident investigation to ensure that all levels of the agency understand how future crew survival investigations should be performed.

Finding. The unique nature of the event, closeness of investigators to the accident victims, lack of previous exposure to the results of such tragedies, and need to keep information confidential created stress on some members of the investigation team. Counseling was provided, but the follow-up could be improved.

Recommendation A9. Post-traumatic stress debriefings and other counseling services should be available to those experiencing ongoing stress as a result of participating in the debris recovery and investigation. Designated personnel should follow up on a regular basis to ensure that individual needs are being met.

Recommendation A7. To aid in configuration control and ensure data are properly documented, report generation must begin early in the investigation process.

Finding. CAIB/CSWG data were not cataloged. *Challenger* supporting data were mostly uncataloged and unorganized, limiting their usefulness for investigations. *Challenger* debris is unpreserved and inaccessible for analysis.

Conclusion A8-1. Spacecraft accidents are rare, and each event adds critical knowledge and understanding to the database of experience.

Recommendation A8. As was executed with *Columbia*, spacecraft accident investigation plans must include provisions for debris and data preservation and security. All debris and data should be cataloged, stored, and preserved so they will be available for future investigations or studies.

Finding. Brightening events were easier to correlate between videos than debris-shedding events.

Finding. Sun angle illumination impacted the visibility of debris in video recordings.

Finding. Not all videos segments within compilations were individually categorized. Not all videos were re-reviewed once a better understanding of events had been gained.

Recommendation A11. All video segments within a compilation should be categorized and summarized. All videos should be re-reviewed once the investigation has progressed to the point that a timeline has been established to verify that all relevant video data are being used.

Finding. The lack of a single, standard data format for latitude/longitude data and the potential ambiguity associated with the need to convert data of different formats resulted in possible data errors.

Recommendation A10. Global Positioning System receivers used for recording the latitude/longitude of recovered debris must all be calibrated the same way (i.e., using the same reference system), and the latitude/longitude data should be recorded in a standardized format.

4.1 Background

Crew Survival Working Group

The CSWG was formed to support the CAIB on February 21, 2003 by authorization of the Johnson Space Center (JSC) Director. The CSWG was co-chaired by directors of the Space Life Sciences Directorate and the Engineering Directorate. Membership included personnel from Space Life Sciences, Engineering, Mission Operations, and Flight Crew Operations. Dr. James Bagian, a former astronaut and crew survival investigator for the *Challenger* accident, was the flight surgeon advising the CAIB. Dr. Bagian, together with Lt. Col. Don White of the Air Force Safety Center (an expert in crew equipment investigations), were the primary liaisons to the CAIB for the CSWG.

The CSWG was tasked with a limited charter: first, to determine the cause of death of the crew, second, to determine the "survival gap" (what equipment or procedures might have kept the crew alive), and, third, to pass the results to the CAIB. The CSWG performed aerodynamic, thermal, and structural analyses on individual debris items and an intensive study of the crew helmets and seats. In the process, team members made several trips to Kennedy Space Center (KSC) to view the crew module (CM) and helmet and seat debris. The CSWG developed a timeline that is consistent with the official CAIB timeline to derive the sequence of crew survival events based on the data.

Following recovery in the field, the crew remains were transported to the Air Force Institute of Pathology (AFIP) at Dover Air Force Base, Maryland for forensic and deoxyribonucleic acid (DNA) identification analysis. The Office of the Armed Forces Medical Examiner is the department within the AFIP that was responsible for determining the cause and manner of death for the crew of *Columbia*. The AFIP issued a report to the CSWG on the findings of the autopsies in May 2003, near the end of the CAIB investigation. Tissue samples were sent to the Federal Bureau of Investigation (FBI) for additional medical forensic analysis. Due to the timing of the AFIP report to the CSWG, little medical information was available to be discussed at the CSWG team level, and only a preliminary effort to integrate the medical findings with the engineering findings was possible. The need to fully integrate these findings was recognized by the CSWG team at the time.

Four types of data – aerodynamics, orbiter, forensic hardware, and forensic medical – were collected by the CSWG. These data were submitted to Dr. Bagian and Lt. Col. White on June 24, 2003 to assist them in preparation of the CAIB Report. The CAIB Report, Volume I, which was published in August 2003, contained a short discussion concerning crew survival.[1] Somewhat more detail was released in Volume V, which was published in October 2003.[2]

Unlike the other elements of the CAIB investigation, there was no NASA process for the administrative and financial framework of the CSWG investigation. As a result, when the CAIB investigation concluded, there were no resources available to continue the CSWG work although it was clear more work remained. The CAIB did not make any formal recommendations (only observations) regarding crew survival. Due to the priority of the Return to Flight program, CSWG activities were discontinued in October 2003. No report was published by the CSWG. Efforts were made to locate new funding and to identify an organization that would manage the continuation of the investigation and publish a report.

The SCSIIT was formed in October 2004 to resume the work of the CSWG and perform a multidisciplinary analysis of the *Columbia* fatal mishap that focused on the crew, crew equipment, and CM. The specific products include: the establishment of a comprehensive, computer-searchable body of information, the virtual reconstruction of the mishap, and a comprehensive report that provides valuable understanding and information for the design of crewed space vehicles and crew safety equipment.

To learn how improvements to crew survival could be made in the future, the following questions needed to be answered:

[1] *Columbia* Accident Investigation Board Report, Volume I, Section 10.2, Crew Escape and Survival, August 2003.
[2] *Columbia* Accident Investigation Board Report, Volume V, Appendix G.12, Crew Survivability Report, October 2003.

- What events occurred that had lethal potential for the crew, even after the crew became deceased?
- How did the CM lose structural integrity?
- How did the crew equipment perform in the pressure, thermal, and acceleration environments that were experienced by the crew?
- What operational insight did the crew members have into the events that occurred, and was their training appropriate and adequate for the circumstances encountered?

4.2 Spacecraft Crew Survival Integrated Investigation Team Structure and Personnel

Because the SCSIIT investigation was mainly concerned with crew survival, the team had a "crew-centric" focus. Shuttle crew members are surrounded by layers of protection, with the crew equipment being the closest layer, the CM being the next layer, and the vehicle being the outermost layer (figure 4-1).

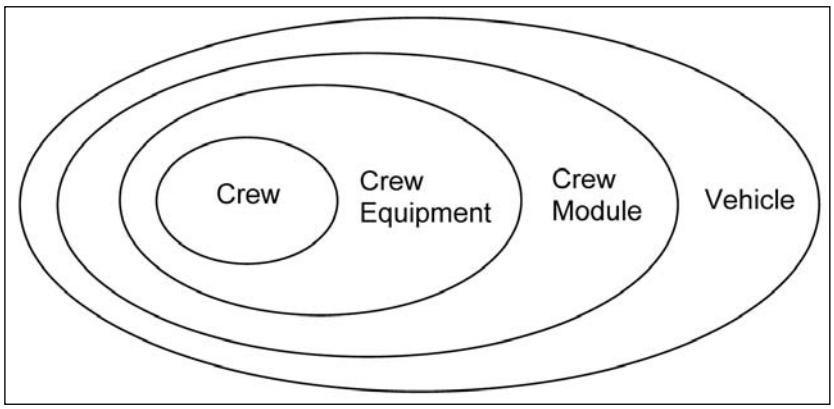


Figure 4-1. ***Concentric layers of protection for space shuttle crew members.***

After it was decided that the crew and the three "layers" would be used as functional areas of focus for the SCSIIT investigation, four teams were formed. These teams were highly interdependent in the development and sharing of information.

1. The Vehicle Team was responsible for determining the dynamics of the vehicle from loss of control (LOC) until the vehicle breakup to ascertain the dynamics that the crew members experienced. This team determined the vehicle breakup sequence and the motion of the intact orbiter and the free-flying forebody, and performed all ballistics analysis on debris. Additionally, this team performed most of the thermal analyses.

2. The Crew Module Team was responsible for determining the CM environments (acceleration, thermal, and pressure) until CM breakup to ascertain the environments that the crew members experienced. This team also determined the CM breakup sequence.[3]

3. The Crew Equipment Team was responsible for determining the performance of the crew equipment (crew worn equipment, seats, etc.) to ascertain how the equipment enhanced or worsened the crew survival probabilities. Results of analysis on crew equipment were used by the Crew Module and Crew Teams to aid their analyses.

[3]Prior to the completion of the investigation, the Crew Module Team lead had to return to his "home" organization. The Crew Module Team was essentially dissolved and its responsibilities were spread to the other three sub-teams.

4. The Crew Team was responsible for analyzing crew awareness during the mishap and the causes of deaths of the crew members, and identifying the threats to crew survival. Analysis of crew injuries was used by the vehicle and crew equipment teams to aid in developing their conclusions.

The SCSIIT management structure is shown in figure 4-2.

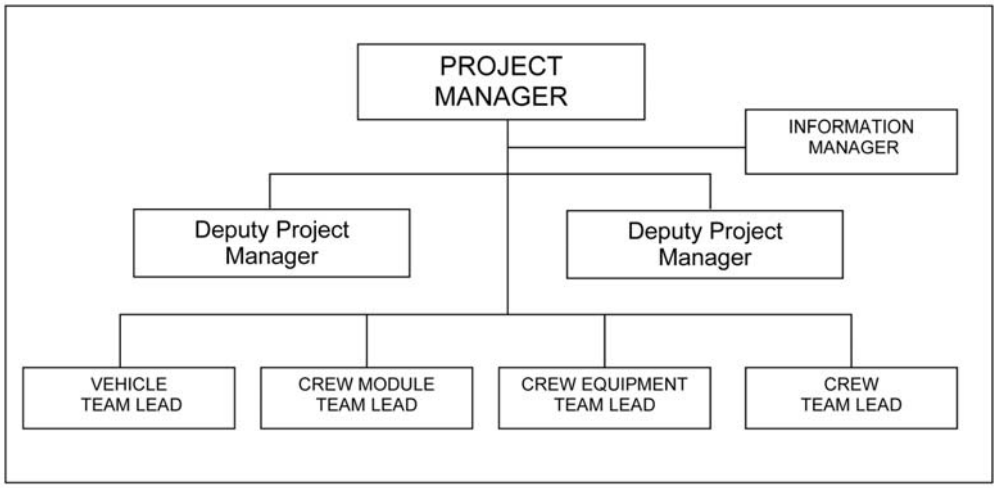


Figure 4-2. ***Management and leadership structure*****.**

4.2.1 Team membership

Potential SCSIIT members were identified based on previous association with the CAIB and CSWG and experience in the disciplines that were necessary to conduct this investigation. Selection was made considering an individual's area of knowledge, experience, interest, and availability. The team members were expected to divide their time between the SCSIIT project tasks and their "home" organization's tasks. At times, the home organization's work had deadlines requiring work to be prioritized over SCSIIT work. This led to delays in completing the SCSIIT work and, in some cases, significant decrease in availability or complete loss of members of the SCSIIT. It is recommended that personnel on future investigation teams be temporarily "released" from their "home" organizations to be free to work full time for the investigation organization.

Finding. NASA priorities put emphasis on Return to Flight recommendations, long-term recommendations, and observations, in that order. As a result, the SCSIIT effort suffered from low priority relative to other program recovery efforts. Team members had to divide their time between the investigation work and the work for their home organization. This led to delays in completing the SCSIIT work and, in some cases, significant decrease in availability or complete loss of members of the SCSIIT.

> ***Recommendation A1.*** In the event of a future fatal human space flight mishap, NASA should place high priority on the crew survival aspects of the mishap both during the investigation as well as in its follow-up actions using dedicated individuals who are appropriately qualified in this specialized work.

SCSIIT members were assigned to lead each of the four teams. The team lead was responsible for coordinating the analyses performed by that team and documenting the results. SCSIIT members were generally assigned to one team but often supported other teams when their expertise was called for. Although represented in figure 4-2 as a hierarchical team, the SCSIIT functioned as a highly integrated group with multiple interactions among the teams. This functional organization and the interactions among teams are shown in figure 4-3.

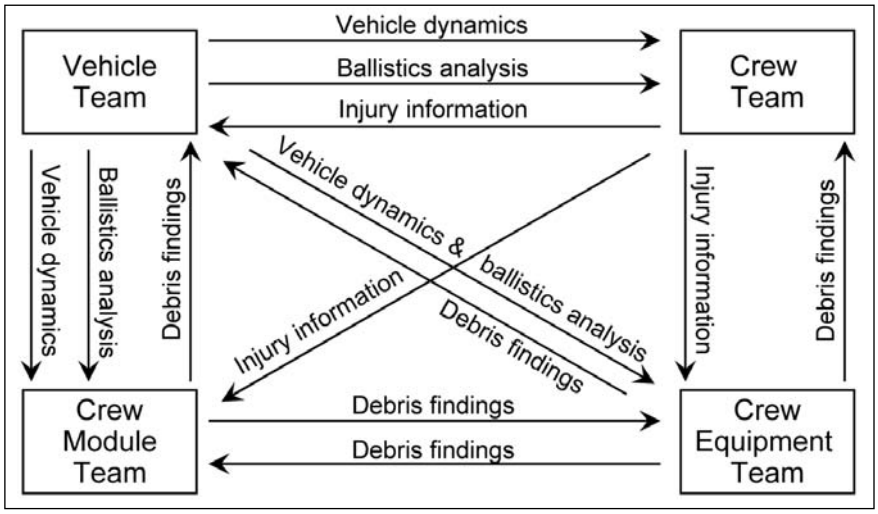


Figure 4-3. ***Functional organization of the Spacecraft Crew Survival Integration Investigation Team.***

The principal SCSIIT members and their respective home organizations, backgrounds, and responsibilities are listed in Table 4-1.

Table 4-1. ***Principal Spacecraft Crew Survival Integration Investigation Team Members***
(*Denotes SCSIIT members who were on the CSWG)

SCSIIT Member[4]	Organization	Background	Responsibilities
Dr. Gregory Hite	Shuttle and Exploration Division, Safety and Mission Assurance Directorate (retired 12/07)	Crew Survival	Project Manager
Dr. Nigel Packham	Safety and Mission Assurance Directorate	Safety and Life Sciences	Project Manager
Col. (ret) Pam Melroy, United States Air Force (USAF)*	Astronaut Office, Flight Crew Operations Directorate	Test Pilot, Shuttle Pilot Astronaut	Deputy Project Manager
Dr. Craig Fischer*	Space Life Sciences Directorate (retired 12/06)	Flight Surgeon and Pathology	Deputy Project Manager/ Crew Team Lead
Chrystal L. Hoelscher, Science Applications International Corporation (SAIC)	Shuttle and Exploration Division, Safety and Mission Assurance Directorate	Information/Knowledge Management and Database Administration	Information Management
Ellen Braden	Aeroscience and Flight Mechanics Division, Engineering Directorate	Aerodynamics and Flight Mechanics	Vehicle Team Lead
Mark Adams, United Space Alliance (USA)	Vehicle Integration Test Office, Flight Crew Operations Directorate	Shuttle Vehicle Integration Test Office	Crew Module Team Lead
David J. Pogue	EVA[5], Robotics and Crew Systems Operations Division, Mission Operations Directorate	Crew Systems and Crew Escape Equipment Operations Instructor and Flight Controller	Crew Equipment Team Lead
Eric Flagg, SAIC	Shuttle and Exploration Division, Safety and Mission Assurance Directorate	Military Pilot	Safety and Mission Assurance

[4]With the exception of Duke Tran from Boeing/Palmdale, California, all personnel were based at JSC in Texas.
[5]EVA = extravehicular activity.

Table 4-1. *Principal Spacecraft Crew Survival Integration Investigation Team Members* (Continued)
(*Denotes SCSIIT members who were on the CSWG)

SCSIIT Member[6]	Organization	Background	Responsibilities
Joe Hamilton	Habitability and Environmental Factors Division, Space Life Sciences Directorate	Military Pilot	Virtual Reconstruction / Concept Evaluation Laboratory (CEL)
Kandy Jarvis*, Lockheed Martin Space Operations/Mission Services (LMSO/LMMS)	Human Exploration Science Office, Astromaterials Research and Exploration Science Directorate	Orbital Debris and Planetary Astronomy	Video and Debris Analysis
Dennis Pate, SAIC	Shuttle and Exploration Division, Safety and Mission Assurance Directorate	Human Factors	Timeline and Crew Awareness
Duke Tran, Boeing	Boeing/Palmdale	Senior Orbiter Structures Designer	Orbiter Structural Analysis

The SCSIIT team principals called on other individuals for analysis and information. These individuals are listed in Table 4-2.

Table 4-2. *Individuals Supporting the Principal Spacecraft Crew Survival Integration Investigation Team*
(*Denotes SCSIIT members who were on the CSWG)

Name[7]	Organization
Ketan Chhipwadia*	Crew and Thermal Systems Division, Engineering Directorate
Katie Boyles Lee Bryant* Peter Cuthbert	Aeroscience and Flight Mechanics Division, Engineering Directorate
Lynda R. Estes Jeremy Jacobs* Kenneth Wong* Leslie Schaschl Brian Mayeaux	Structural Engineering Division, Engineering Directorate
Curtis Stephenson	Crew and Thermal Systems Division, Engineering Directorate
William Sarles, SAIC Paul Wilson, SAIC	Shuttle and Exploration Division, Safety and Mission Assurance Directorate
Robert Behrendsen, Barrios Technology (BAR) J. Lynn Coldiron, USA Adam Flagan, USA	EVA, Robotics, and Crew Systems Operations Division, Mission Operations Directorate
Laurie J. Bergman, Tietronix Software, Inc. Richard D. Delgado Jose Dobarco-Otero, Jacobs Technology (ESCG) William Rochelle,*[8] ESCG Ries Smith, ESCG	Space Life Sciences Directorate
Sudhakar Rajulu Kurt G. Clowers, Muniz Engineering (MEI) Sarah Margerum, LMSO Richard Morency	Habitability and Environmental Factors Division, Space Life Sciences Directorate

[6]With the exception of Duke Tran from Boeing/Palmdale, California, all personnel were based at JSC in Texas.
[7]With the exception of where noted, all personnel were based at JSC.
[8]Tragically, Mr. Rochelle passed away before this report was published.

Table 4-2. ***Individuals Supporting the Principal Spacecraft Crew Survival Integration Investigation Team*** **(Continued)**
(*Denotes SCSIIT members who were on the CSWG)

Name[9]	Organization
Rita Alaniz, MEI Rodney DeSoto, LMSO Chris Keller, BAR Mark Langford, LMSO Terry Mayes, USA Jeremy Reyna, Wyle Laboratories Chris Slovacek, LMSO Matt Soltis	Habitability and Environmental Factors Division/Concept Exploration Laboratory (CEL), Space Life Sciences Directorate
Danny Olivas	Astronaut Office, Flight Crew Operations Directorate (materials analysis)
Sharon Hecht, Tesseda (TES) Perry Jackson, TES Cindy Bush, TES	Information and Applications Systems Division, Information Resources Directorate
Stacey Nakamura	Safety and Mission Assurance Directorate
James Comer, USA Amy Mangiacapra, USA	*Columbia* Research and Preservation (CRP) Office, KSC
Steve McDanels M. Clara Wright	Failure Analysis and Materials Evaluation Branch, Materials Science Division, Engineering Directorate, KSC
Rick Russell	Orbiter Sustaining Engineering Office, KSC
Roy Christoffersen, SAIC	Astromaterials Research Office, Astromaterials Research and Exploration Science Directorate
Darren Cone	White Sands Test Facility
David Bretz*, LMSO Tracy Thumm*, LMSO Kathleen McBride*, LMSO Kim Willis*, LMSO	Human Exploration Science Office, Astromaterials Research and Exploration Science Directorate
Donna Shafer	Office of the Chief Counsel

Additionally, expert personnel who were external to NASA provided assistance to the SCSIIT. These persons are listed in Table 4-3.

Table 4-3. ***Experts External to NASA***

Name	Organization
Dr. Robert Banks	Biodynamic Research Corporation (BRC)
Dr. Jon Clark	National Space Biomedical Research Institute
Dr. Richard Harding	Biodynamic Research Corporation
Dr. Gregory Kovacs	Stanford University
Dr. Robert McMeekin	Previous Federal Air Surgeon
Dr. Thomas McNish	Biodynamic Research Corporation
Dr. Charles Ruehle	Federal Aviation Administration (FAA)
Dr. Glenn Sandberg	Armed Forces Institute of Pathology
Dr. Charles Stahl	Former Chairman of the Department of Forensic Sciences at the Armed Forces Institute of Pathology
Dr. Harry Smith	Biodynamic Research Corporation

It should be noted that extensive training on accident investigation processes and procedures was not provided to the team. Many of the team members had no previous accident investigation training. Because the SCSIIT members supported the investigation on a part-time basis, taking a lengthy course in accident investigation processes and procedures was not feasible. NASA has only a small group of formally trained accident investigators across the many field centers. These investigators also have other safety duties. None

[9]With the exception of where noted, all personnel were based at JSC.

were available to participate in the SCSIIT investigation due to higher priority activities occurring at the time, including Return to Flight preparation.

The SCSIIT leadership felt it was preferable for NASA space flight technical experts to learn accident investigation techniques rather than to have accident investigation experts become technical experts on the space shuttle. Therefore, personnel were selected for the SCSIIT based on experience with the CAIB and CSWG and/or who were technical experts in the disciplines that were necessary to conduct the investigation. Many SCSIIT team members did not have formal accident investigation training; advice from experienced investigators was sought at various times. Accident investigation experience would have been helpful to focus the team's efforts, especially early in the investigation.

Finding. Formally trained NASA-designated accident investigation personnel were not available for inclusion on the SCSIIT due to the intensity of safety and mission assurance work related to Return to Flight activities. SCSIIT members were selected primarily based on their technical knowledge and experience as well as availability. Many SCSIIT members did not have formal accident investigation training. The team preparation training sessions did not include the lengthy accident investigation training that is normally provided to NASA-designated investigators.

> ***Recommendation A1.*** In the event of a future fatal human space flight mishap, NASA should place high priority on the crew survival aspects of the mishap both during the investigation as well as in its follow-up actions using dedicated individuals who are appropriately qualified in this specialized work.

4.3 Investigative Process

The team traveled to KSC to perform first-hand inspection of the *Columbia* debris. Additionally, the results of previously performed analyses were reviewed. Additional analyses were then conducted (ballistics, materials, structural, aerodynamic, etc.).[10] Individually and in small groups, the team members assessed the results and formed conclusions. These conclusions were presented at Technical Interchange Meetings (TIMs) to the entire SCSIIT to compare and integrate results and findings. The team held four TIMs (March/April 2005, June 2005, August 2005, and March 2006). This was an iterative process, with each TIM resulting in more trips, analyses, and scenario revisions. In the process, integrated products were generated, with the most important being the timeline of key events. When possible, "no earlier than" and "no later than" times were identified for key events, and sequences were built.

In many regards, this investigation presented several challenges. Space flight is a relatively new and rare experience, and, fortunately, there have been only a few fatal mishaps. Consequently, there is no integrated or widely available body of information for how to analyze spacecraft accidents for crew survival. The physics of atmospheric entry and the environment in which a human spacecraft mishap occurs are unique when compared to aviation. The SCSIIT had to break new ground in how to conduct the investigation of a singular event in such a complex environment. The team had to determine how to modify existing models and tools, which are normally used for specific nominal situations in a predictive manner, to understand the mishap environment. Multiple tools and analyses were evaluated to provide multiple sources of information to develop scenarios.

> ***Recommendation A4.*** Due to the complexity of the operating environment, in addition to traditional accident investigation techniques, spacecraft accident investigators must evaluate multiple sources of information including ballistics, video analysis, aerodynamic trajectories, and thermal and material analyses.

[10] Debris analysis and other analysis methods and tools are described in Section 4.5.

The SCSIIT collected a large amount of data from the CAIB and CSWG activities. Also, a large volume of new data was generated during the investigation. The SCSIIT Information Manager was tasked with organizing and tracking the information in a SCSIIT Project database.

The initial SCSIIT activities included review of data produced by the CSWG in support of the CAIB. It was discovered that the existing data had not been centrally cataloged or organized, making access to specific items difficult. The data that were available were dispersed among several groups, as shown in Table 4-4.

Table 4-4. *Data Types and Data Sources*

Data Type	Data Source
Medical: X rays, autopsy reports, photographs, tissue samples	Armed Forces Institute of Pathology Biodynamic Research Corporation (BRC) Space Life Sciences Directorate, JSC
Debris: Seats, suits, CM, forward fuselage (FF), windows	*Columbia* Research and Preservation Team, KSC
Technical: Ballistics, thermal, structure, and materials analyses	Engineering Directorate, JSC
Video: Recovered On-board Video, Ground Based Video	Image Science and Analysis Group (ISAG) *Columbia* Video Archives Group Concept Exploration Laboratory
Operational: Vehicle Telemetry Data	Missions Operations Directorate

Because no final report was generated by the CSWG, it was not uncommon for several versions of specific analyses or presentations to exist as scenarios were iterated. Most of the files did not include the author, date, or version of the document, making it difficult to determine the final conclusions for the subject document. Additionally, without contact information, the team members were not always able to contact the original authors of these documents to discuss the contents or obtain additional information that may have aided the investigation. Report generation must begin very early in the investigation process, using a systematic approach with established procedures. At some point in the investigation, a transition from fact-finding must give way to documenting the findings, conclusions, and recommendations.

Finding. It was not uncommon to find several versions of documents supporting CAIB and CSWG work.

> ***Recommendation A6.*** Standard templates for accident investigation data (document, presentation, data spreadsheet, etc.) should be used. All reports, presentations, spreadsheets, and other documents should include the following data on every page: title, date the file was created, date the file was updated, version (if applicable), person creating the file, and person editing the file (if different from author).
>
> ***Recommendation A7.*** To aid in configuration control and ensure data are properly documented, report generation must begin early in the investigation process.

During the SCSIIT investigation, it was discovered that *Challenger* information was cataloged with keywords and descriptions that were more oriented toward the overall mishap investigation. This may be a result of the fact that the Rogers Commission did not specifically investigate crew survival. Additionally, the data were cataloged prior to current storage techniques and re-cataloged later by different personnel well after the accident investigation was complete. It was difficult to retrieve specific documents and analyses related to the CM and to crew survival. In many cases, *Challenger* information was obtained from individuals who were involved in the original investigation. Moreover, the *Challenger* debris items are unpreserved and inaccessible for analysis as they are stored in an abandoned underground missile silo with no access or climate control provisions. The lack of debris for comparison and methods of data preservation made the *Challenger* data essentially unavailable for this investigation. It is recommended that accident investigation plans include provisions for debris and data preservation and security. All debris and data should be cataloged, stored, and preserved so they will be available for future investigations or studies.

Finding. CAIB/CSWG data were not cataloged. *Challenger* supporting data were mostly uncataloged and unorganized, limiting their usefulness for investigations. *Challenger* debris is unpreserved and inaccessible for analysis.

Conclusion A8-1. Spacecraft accidents are rare, and each event adds critical knowledge and understanding to the database of experience.

Recommendation A8. As was executed with *Columbia*, spacecraft accident investigation plans must include provisions for debris and data preservation and security. All debris and data should be cataloged, stored, and preserved so they will be available for future investigations or studies.

4.3.1 Public release of information

For comparison and potential application to the SCSIIT investigation, the team researched how other government agencies conduct aircraft accident investigations. SCSIIT looked at processes that are used by the National Transportation Safety Board (NTSB), FAA, U.S. Air Force, and U. S. Navy to get an overview of the agencies' crew survival investigation processes. While the investigation process used by the SCSIIT was similar to those processes used in other agencies, there were some significant differences due to the uniqueness of the spacecraft operating environment. The most notable differences regarded public release of information and the use of personnel from the affected organization for the investigation.

The NTSB and FAA must conduct their affairs publicly, while Department of Defense (DoD) investigations are considered internal matters, and documents are released at the discretion of the DoD. At NASA, the public release of sensitive information is not specifically addressed in existing accident investigation plans. As a result, there was hesitation to investigate information that was relative to what happened to the crew out of the concern that the information would result in public release, and subsequent inappropriate speculation that would be painful to both the employees and the families. A more preferable situation would be to have a pre-determined plan for what crew-related information is appropriate to release to the public, and when the information should be released.

Finding. Concerns about public release of sensitive information relative to the crew creates obstacles to the performance of crew survival investigations.

Future spacecraft accidents may result in injuries and/or fatalities. To preserve the privacy of the *Columbia* crew members and their families, access to medically sensitive data, including the crew's personal items, was provided only to those personnel who had a need to know. This practice remains in place today and must remain in place in the future for *Columbia*-related information as well as for any future aerospace incident involving human casualties. This practice will preserve the privacy of the victims and their families. Almost as important, it will ensure that future investigations can be conducted without the concern of inappropriate release of sensitive information.

Recommendation A2. Medically sensitive and personal debris and data should always be available to designated investigators but protected from release to preserve the privacy of the victims and their families.

Recommendation A3. Resolve issues and document policies surrounding public release of sensitive information relative to the crew during a NASA accident investigation to ensure that all levels of the agency understand how future crew survival investigations should be performed.

4.3.2 Using members of affected organizations in the investigation

In commercial or military aviation accidents, members of the organization that is affected by the accident generally are not members of the investigation teams. For example, pilots in a specific squadron would not be members of a team investigating an accident involving one of their squadron-mates. Because space flight operations are highly specialized and there are no other "external" organizations with sufficient relevant experience, it is impractical to follow this investigative practice for NASA crewed spacecraft accidents. Initially, the CSWG did not include current astronauts or crew escape operations training personnel. Astronauts were added to the CSWG and provided operational experience to the group. Crew escape operations and training joined the team during the follow-on SCSIIT portion of the investigation.

A potential downside to using accident investigation personnel who are close to the victims of the accident is the psychological impacts of the investigation on the investigators. A crew survival investigation is an emotionally charged process, causing considerable stress in the people involved in the investigation. The psychological welfare of personnel who are involved in debris recovery must be protected as part of the accident investigation process. In the aftermath of the *Columbia* accident, there were no consistent post-event stress debriefings to assist with post-traumatic stress disorder syndrome in recovery and mishap investigation personnel until later investigation phases.

Finding. The unique nature of the event, closeness of investigators to the accident victims, lack of previous exposure to the results of such tragedies, and need to keep information confidential created stress on some members of the investigation team. Counseling was provided, but the follow-up could be improved.

> ***Recommendation A9.*** Post-traumatic stress debriefings and other counseling services should be available to those experiencing ongoing stress as a result of participating in the debris recovery and investigation. Designated personnel should follow up on a regular basis to ensure that individual needs are being met.

4.4 Medical Process Issues

When the *Columbia* accident first occurred, the highest priority task was rescuing the crew members. When it became apparent that they had not survived, the task transitioned to recovering the crew remains. The FBI was the agency that was in charge of recovering the crew member human remains. The Bureau was assisted by members of the Environmental Protection Agency, local and state law enforcement agencies, local coroners, and members of the NASA Astronaut Office. As searchers and citizens reported possible remains, teams were dispatched to document and recover the remains. In many cases, forensic experts in the field were able to make preliminary determinations of whether the remains were human or otherwise. Initially, trained recovery personnel were used to identify human remains in the field until all principal remains were recovered. Subsequently, recovery personnel were directed to collect all remains regardless of whether or not they could positively be identified as human or other. In most cases photographs of the remains were taken in the field prior to collection to document the "as-found" condition for use during the autopsies. The quality of the photographs and the information recorded varied greatly from site to site. In some instances where human remains may have been found near spacecraft hardware, there were no established procedures for documenting these important physical relationships.

After recovery, the remains were transported first to a local morgue facility. Intake photographs were taken to document the "as-received" condition of the remains for use during the autopsies. A forensic pathologist performed a review of the remains and was able to separate out many nonhuman remains that had been collected. The human remains were then prepared and transported to Barksdale Air Force Base, and then on to the AFIP at Dover Air Force Base.

The AFIP was the government agency that was tasked with positively identifying the remains, performing the autopsies, and preparing the remains for burial. Identification of the remains relied primarily on DNA testing and dental records.

NASA expected the AFIP service to include a complete cause of death analysis in similar fashion to what the Air Accident Investigation Team had historically done and had been done for the crew of STS-51L (*Challenger*). The AFIP performed very well in the receipt of human remains, gross examination, and definitive identification, and in providing death certificates and mortuary science support. However, the gross autopsy reports were incomplete and did not contain the level of detail necessary for a thorough accident investigation. X rays that were taken were of poor quality, and no interpretation of the X rays was transmitted to NASA. The microscopic examinations included in the final autopsy reports lacked many of the specific details that were required for the investigation. Although the AFIP continues to participate in routine aircraft accident investigations, in this instance it did not have the necessary resources to integrate all of the forensic findings into a comprehensive accident investigation report due to operational commitments. Subsequently, NASA has addressed these issues with process changes.

4.5 Analysis Methods, Processes, and Tools

Various methods, processes, and tools were used by the SCSIIT sub-teams to conduct their analyses. The following sections describe these methods, processes, and tools.

4.5.1 *Columbia* debris repository

Vehicle debris is one of the most useful sources of evidence in an accident. The debris can be used to determine failure modes, fracture dynamics, and thermal exposure, helping to develop vehicle breakup sequences. The debris also can be used to support physical and virtual vehicle reconstruction. *Columbia* debris is currently stored on the 16th floor of the Vehicle Assembly Building at KSC. The CRP Office is tasked with managing the debris and the database and providing access to the debris for research purposes. The investigation would not have been possible without the careful work done by the Reconstruction Team, particularly in the area of CM reconstruction. SCSIIT members made several trips to KSC to analyze the debris.

4.5.2 Physical reconstruction

Visual and microscopic inspection of individual debris items provided details of the thermal damage to components, the directionality of melted material deposits, and the characteristics of fracture surfaces. Evidence of mechanical damage or thermal exposure on fracture edges was used to help determine the timing of the breakup. For example, significant melting of fracture surfaces would indicate that the fracture occurred first, followed by thermal exposure. In addition to the study of individual debris items, multiple debris items were studied together as reconstructions of portions of the orbiter.[11] Reconstructions were performed for the flight deck and middeck floors, CM forward (X_{cm} 200) and aft (X_o 576) bulkheads, X_o 582 ring frame bulkhead, the airlock and tunnel adapter structure, CM avionics bays, crew seats, forward Reaction Control System pod components, and selected portions of the forward fuselage. These reconstructions were useful for helping to determine breakup sequences and mechanical and thermal damage patterns across large areas.

[11]Physical and time constraints prohibited a full reconstruction of *Columbia*.

4.5.3 Virtual reconstruction

Using modeling, simulation, and visualization tools, members of the Concept Exploration Laboratory (CEL) at JSC integrated data from multiple disciplines to create an interactive, dynamic, 3-dimensional simulation of the orbiter. This simulation included a virtual reconstruction of the CM interior using photographs of recovered debris items. The simulation also provided a dynamic visualization of the vehicle dynamics and breakup sequence, thereby creating a virtual reconstruction not only of the vehicle but also of the accident events timeline. The CEL team also developed unique capabilities and analysis techniques to support the investigation.

Trajectory and vehicle dynamics data for the intact orbiter were entered into the simulation, as were trajectories for selected debris items. Coordinates of the recovery location for debris items were loaded into the CEL simulation, and the debris items were grouped according to their origin on the vehicle (i.e., left wing, tail structure, FF, etc.).

The simulation was used to visualize the predicted vehicle dynamics and breakup sequence. Telemetry data were used where available to represent the flight path and vehicle orientation. The reference trajectory was used to represent translation. The aerodynamic simulation of the vehicle attitude (see Section 2.1) was used between LOC and the Catastrophic Event (CE). The aerodynamic simulation of the forebody was used to represent the orientation after the CE. Figure 4-4 shows frames from the dynamic visual simulation. The image on the left represents the orbiter just before the LOC, and is viewed from a point above, behind (west), and to the left (north) of the flight path. The image on the right represents the orbiter just after the CE, and is viewed from above and behind (west) of the flight path. The three main items in this view are the payload bay doors, the forebody, and the main portion of the orbiter. The left wing has already departed from the orbiter and is not visible in this view.

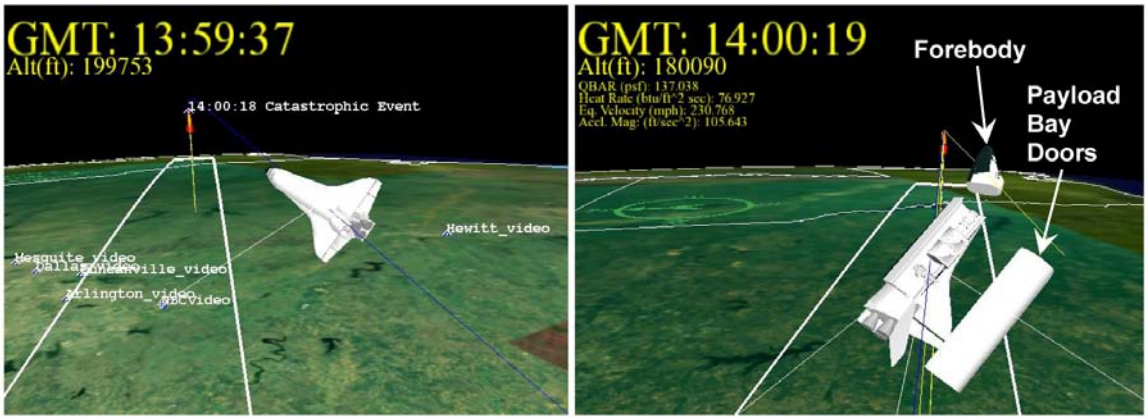


Figure 4-4. ***Visual representation of the breakup sequence, looking east.***

Figure 4-5 shows the trajectories of the intact orbiter (the blue line in the upper portion of the image) and multiple debris items (different color lines) as viewed from a point in space looking down toward the Earth in a westerly direction. State borders are shown in white, and selected cities are labeled in green. The white rectangle represents the main debris search area. Almost all of Texas is visible, with east Texas shown in the center of the image. Western Louisiana is at the bottom of the image.

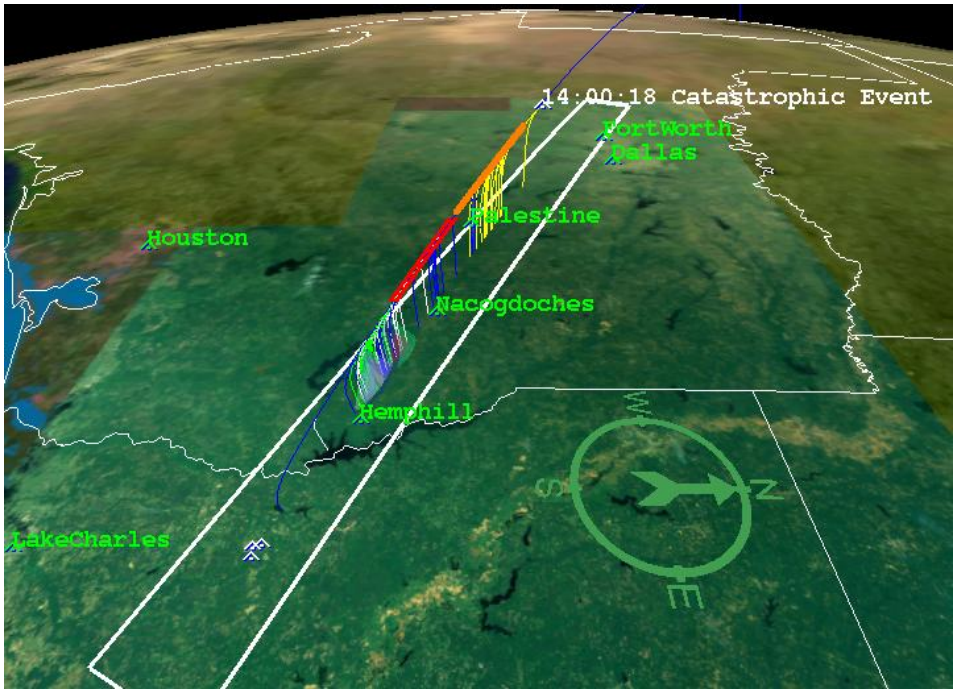


Figure 4-5. ***Orbiter and debris trajectories, looking down at Texas in a westerly direction.***

The simulation could be viewed from any vantage point, such as zooming in on the CM debris spread or any of the sites from which ground-based video was recorded. Figure 4-6 shows the trajectory and CM debris groupings as viewed from a point north of the trajectory, looking south-southwest. Debris item groupings are shown as triangles and are labeled ("Airlock", Middeck floor", etc.).

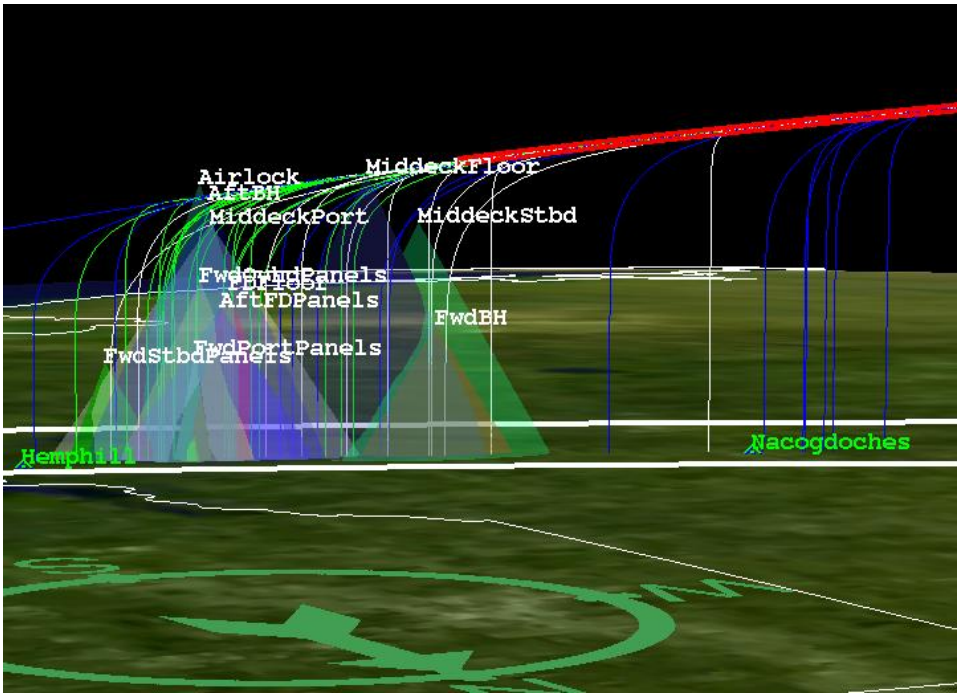


Figure 4-6. ***Crew module debris groups, looking south-southwest.***[12]

[12]"FwdBH" = Forward Bulkhead; "AftFDPanels" = Aft Flight Deck Panels.

Portions of the TIMs were conducted in the CEL where the SCSIIT could interact with a dynamic simulation of the LOC and breakup events while simultaneously reviewing video from the various ground sites that recorded the events. This was an invaluable asset in developing and refining breakup sequences.

The CEL developed a virtual reconstruction of the cockpit by mapping photographs of selected debris items to a computer model of the crew cabin. This virtual reconstruction was used in a variety of ways to augment direct examination of the debris. Figure 4-7 shows examples of this virtual reconstruction. The image on the left shows the aft flight deck and overhead windows (looking aft) with photographs of recovered panels and window frames included. The image on the right shows several starboard side middeck floor panels and the starboard-most crew seat (the blue item in the upper right corner). This image is from a vantage point looking port and slightly aft.

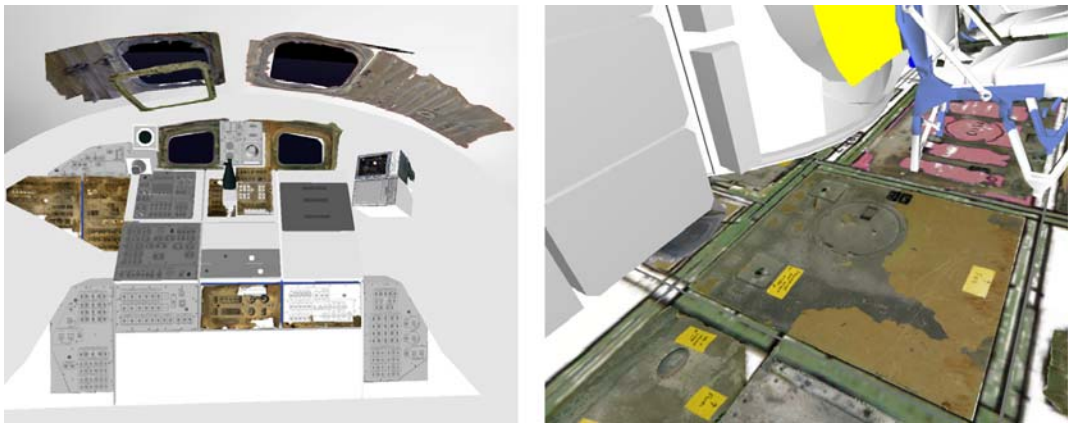


Figure 4-7. ***Virtual reconstruction of the aft flight deck*** **(left image)** ***and middeck floor*** **(right image).**

4.5.4 Motion analysis tools

The vehicle team was tasked with determining the behavior of the vehicle from LOC of the orbiter until forebody breakup. Analyses that were conducted include trajectory analysis, ballistics analysis, thermal analysis, forebody aerodynamic stability analysis, and CM survivability analysis. In addition to performing analysis on the vehicle and forebody, the Vehicle Team performed ballistics and thermal analyses on various items for the other SCSIIT sub-teams.

4.5.4.1 *Trajectory and attitude analyses*

Global Reference Atmospheric Model

The Global Reference Atmospheric Model was used to generate a representative flight day atmosphere, which is the U.S. Standard Atmosphere 1976 model. This model is a steady-state (year-round) model of the Earth's atmosphere at latitude 45N during moderate solar activity.

Intact Orbiter Simulation

The entry simulation[13] was developed to model the dynamics of the intact orbiter between loss of signal (LOS) at Greenwich Mean Time (GMT) 13:59:31 to the CE at GMT 14:00:18. This simulation provided the resulting modeled accelerations on the vehicle structure and crew during this timeframe. The simulation used the preflight predicted vehicle mass properties; downlinked general purpose computer (GPC) data including position, velocity, attitude, and any alarm/warning related data; and the Modular Auxiliary Data System/orbiter experiment recorder sensor data. The final 2-second period of reconstructed GPC data (RGPC-2) were used in an attempt to synchronize the simulation with the actual flight data. Full details about assumptions and models are described in EG-DIV-08-32, IEE Report, Appendix G – Post-LOS Analysis. The simulation used available data; therefore, there is a moderately high level of confidence in the representation of the motion and resulting accelerations.

[13]EG-DIV-08-32, Integrated Entry Team Report, Appendix G – Post-LOS Analysis.

Reference Trajectory

The reference trajectory is the path of the intact orbiter and forebody given assumptions about their aerodynamics properties and ballistic numbers. The trajectory provides a continuous trajectory from LOS to the Crew Module Catastrophic Event (CMCE) to main engine ground impact. The trajectory was used as a common reference for the thermal and debris ballistic analyses. The reference trajectory was generated using Simulation and Optimization of Rocket Trajectories (SORT). The reference trajectory is divided into four phases: the "Nominal Orbiter," with an average ballistic number of 108 pounds per square foot (psf); the "High-drag Orbiter," with an average ballistic number of 41.7 psf (lift generation occurs); the "No-lift Orbiter," with an average ballistic number of 41.7 psf (a 72-degree angle of attack assumed); and the "Forebody" vehicle, with an average ballistic number of 150 psf.

Simulation and Optimization of Rocket Trajectories

The SORT software program is a general-purpose, 3-degrees-of-freedom, computer-based simulation of the flight dynamics of aerospace vehicles. It was used to estimate the time of an object's release from the various configurations that the orbiter experienced during the accident. The aerodynamic forces experienced on the reference trajectories and the heating and atmospheric conditions were also generated. This program was selected because it was previously used to design the shuttle ascent trajectory and because of user familiarity.

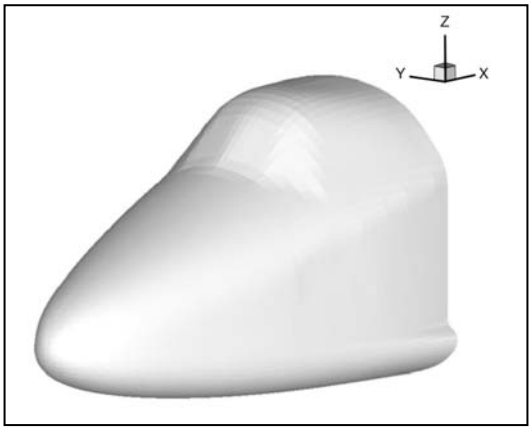
The aerodynamic coefficients, mass, area, and ground recovery location of the debris item were entered into SORT. SORT then calculated release times of the debris item from the reference orbiter trajectory until the location where the object was found matched the calculated ground location. The calculated release time had a ±5-second error bar due to unknowns in the reference trajectory.

Shuttle Engineering Simulator

In 2003, the Shuttle Engineering Simulator (SES) provided an engineering simulation flight reconstruction of the STS-107 entry. This simulation was based on flight data that had been recorded during the descent. The ascent/entry SES was supplied with a set of data files describing the atmospheric conditions during the descent. The original task was to model the changes to vehicle aerodynamics due to the left wing damage. The results of this work are documented in the guidance, navigation, and control portion of the CAIB Report. The visual simulation was also used as part of the "crew awareness" task that was assigned to the Crew Team. Additionally, video of the three primary SES cockpit displays covering the time from GMT 13:58:19 to GMT 13:59:39 (2 seconds after the LOC) were included in the CEL visual products.

Forebody Trim Analysis

The forebody configuration was idealized in that the mass properties were held constant from the CE to the CMCE and the aerodynamic geometry was symmetric. The configuration, which was an intact FF containing the CM, ended cleanly at the X_o 576 bulkhead. At the time, the team did not yet know that parts of the X_o 582 ring frame bulkhead stayed attached to the forebody. The forebody configuration is shown in figure 4-8.



The aerodynamic coefficients and surface pressure distributions for the forebody traveling at hypersonic speeds were predicted using Snewt, an implementation of the Newtonian engineering method. The aerodynamic properties were generated at orientations through a 0-degree to 180-degree sweeps in α (angle of attack) and β (sideslip angle). The aerodynamic moments were plotted in a contour analysis to evaluate stability and were used as inputs to the 6-degree-of-freedom kinematic motion simulation.

Figure 4-8. ***Forward fuselage/crew module (forebody) configuration.***

A sensitivity study was conducted to examine the effect of the forebody's damping moments on the motion. Because no damping moment data were available for the forebody configuration, Apollo capsule damping moments were used as the closest approximation. At the high Mach numbers that were relevant to the *Columbia* flight conditions, the effect of damping moments is low. It was found that the damping moments had little effect on the motion. As a result, subsequent analyses set the damping moments to zero.

The pitch stability of the forebody was examined by plotting the pitching moment coefficient vs. the angle of attack with the sideslip angle held at 0 degree. The roll and yaw stability was examined by plotting the rolling and yawing moment coefficients vs. the sideslip angle with the angle of attack being held constant at the pitch stable angle. This study showed that with any lateral center of gravity other than zero, neither the forebody nor the CM would achieve a stable trim attitude.

These aerodynamic data were then used as inputs to an MSC Visual Nastran simulation. The simulation, which began at approximately the CE, determined the forebody motion to approximately the CMCE. The inputs to the simulation include forebody geometry and mass properties, aerodynamic coefficients, and initial vehicle state conditions. The outputs include the motion parameters, (position, velocity, and acceleration), attitude, attitude rate, and G-loads at the seat locations. Animations of several cases (forebody, CM, initial rates, etc.) were also generated to give a better understanding of possible forebody motion. Unlike the intact orbiter analysis, no data were available for the modeled motion. Therefore, fewer conclusions can be drawn from these data. However, all analyses showed a failure to achieve a trim attitude, and video data supported this conclusion.

Snewt

Snewt is the name of a program that uses the modified Newtonian method to compute a surface pressure distribution and various aerodynamic coefficients. The program uses numerous inputs (reference area, reference length, moment reference center location, Mach number, angle of attack (α), sideslip angle (β), and the ratio of specific heat) and the vehicle outer mold line shape to compute the following coefficients (figure 4-9):

CA = axial force (force component in the body "x" direction)
CY = side force (force component in the body "y" direction)
CN = normal force (force component in the body "z" direction)
Cl = rolling moment (about the "X" body axis)
Cm = pitching moment (about the "Y" body axis)
Cn = yawing moment (about the "Z" body axis)
D = drag (force in the opposite direction of the velocity vector)
Cs = wind side force (force in the wind "y" direction)
L = lift (force in the vertical direction perpendicular to the velocity vector)

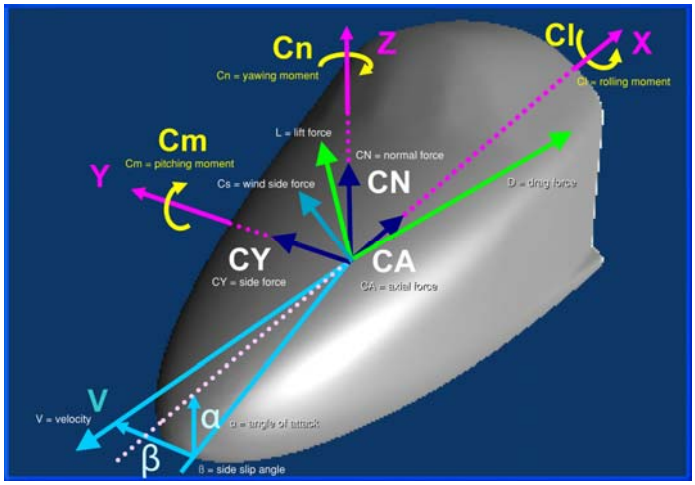


Figure 4-9. ***Coefficients that were used to define the vehicle outer mode line shape.***

The modified Newtonian method is considered a simple model that has several limitations restricting its accuracy. One of the most significant limitations is that the Mach number should be greater than approximately M=5. This was appropriate for the conditions in this accident. Another significant limitation of the method is that regions of a vehicle behind multiple shock waves will not be treated correctly. An example is the portion of a wing that lies inside the bow shock.

MSC Visual Nastran Motion Working Model
The MSC Visual Nastran Motion Working Model is a conceptual design software tool that analyzes mechanical systems. It was used by the SCSIIT to simulate the motion of the forebody (both translational and rotational motion) after the CE to gain understanding into what type of accelerations and forces the forebody and crew may have experienced. This tool was chosen because many of the equations that were needed for the analysis were already embedded in the source code, thus reducing the time needed to set up, initialize, and verify the outcome. Events or additional equations could be integrated into the simulation using the built-in formula language or linking to MATLAB or Excel routines.

Objects can be imported from computer-aided drafting drawings or created within MSC Visual Nastran. An object's mass properties and initial conditions – such as position, velocity, and rotation rates – can be specified. Simulation properties can be measured and displayed in digital or graphical formats and saved to Excel files for further analysis. The MSC Visual Nastran Working Model also provides audio visual interface files of the vehicle's motion, acceleration at different locations on the vehicle, and vehicle rotational rates that can be used for additional evaluation and analysis.

4.5.4.2 *Ballistic analysis*

Ballistic analysis of the debris determined the estimated release time of individual debris items from the orbiter using the reference trajectory and the debris item's recovery location (latitude/longitude), calculated ballistic number and aerodynamic drag coefficients.

Snewt was used to calculate the average ballistic number and aerodynamic drag coefficients. To make these calculations, Snewt requires an object's mass and geometry. If these data were not available from the KSC debris database, the debris item would be measured.

SORT was used to estimate the release time of the debris piece given the reference trajectory, the average ballistic number and aerodynamic drag coefficients from Snewt, and the location of the recovered debris piece. These calculations assume that the recovered debris piece has the same mass and geometry on the ground as it had when it left the orbiter. SORT calculated the release times of the debris item from the reference trajectory until the calculated ground location matched the location where the object was found.

4.5.4.3 *Thermal analysis*

Debris thermal analysis was conducted to determine the entry heating environment for specific debris items. This analysis used the estimated release time and state of the item, the average ballistic number, and the aerodynamic drag coefficients.

Object Reentry Survival Analysis Tool
The Object Reentry Survival Analysis Tool (ORSAT) uses a 3-degrees-of-freedom trajectory model, Detra-Kemp-Riddell aero heating equations, and a 1-dimensional, finite-difference thermal conduction model to predict thermal damage to items experiencing the entry environment. ORSAT was used to predict the temperatures that specific debris items would experience after release from the orbiter. These predicted temperatures were then compared to the observed thermal damage to determine whether entry heating was responsible for the observed thermal damage. In several cases (described in Section 2.1), the observed heat damage on the recovered debris items could not be explained adequately by entry heating alone. Literature searches revealed the phenomena of shock wave impingement and shock wave interference leading to increased heating rates.[14] These phenomena and their effects on spacecraft during the hypersonic flight regime are not well understood. Due to limited resources, the team was not able to advance the understanding of these phenomena.

SINDA/FLUINT
The Systems Improved Numerical Differencing Analyzer (SINDA)/FLUINT is a general-purpose numerical thermal/fluid solver. It was used to compare heat rates for some of the thermal analyses (see

[14]NASA TM X-1669 "Flight Experience with Shock Impingement and Interference Heating on the X-15-2 Research Airplane", October 1968.

Section 2.1), and is maintained by C&R Technologies (www.crtech.com). The user creates a thermal model that is represented by a nodal network of capacitors and conductors, applying the relevant initial and boundary conditions. The tool solves the thermal network and returns the desired parameters (i.e., temperature, heat flow) for the desired times in the simulation.

BLIMP-K

The Boundary Layer Integral Matrix Procedure-Kinetic (BLIMP-K) code is a FORTRAN-based code that was developed during the Apollo Program by Aerotherm for NASA for use in aeroheating analyses. The SCSIIT used it to compare to ORSAT and SINDA during some thermal evaluations (see Section 2.1). This kinetic version of BLIMP allows for a finite-rate, thermo-chemical model to be used in the analysis. The BLIMP-K model, which is described in the latest User's Guide,[15] incorporates temperature-dependent catalytic models and uses a maximum of 15 nodes across the boundary layer. Subsequent modifications allow the catalytic model to be updated at each station along the streamline, as well as allowing the ability to model up to 2,500 streamline stations. Macros and scripts have been used to speed up the pre-processing of the BLIMP-K input files for large parametric studies.

Computational Fluid Dynamics Thermal Analysis of Payload Bay Door Rollers

A computational fluid dynamics (CFD) thermal analysis of the payload bay door rollers was performed by the JSC Applied Aerosciences & CFD Branch. Various Mach numbers, from M = 7.5 to M = 15, were used to determine the flow field environment and temperature at the face of a roller for an orientation with the front of the roller facing directly into the direction of travel. Figure 4-10 shows the predicted heating rate and temperature distribution along the payload bay door roller at M = 10.5. The notations q_{cw} refers to "cold wall" and q_{hw} refers to "hot wall." In this figure, the CFD solution assumes a steady-state solution, meaning that given enough time, the flow will develop into the calculated results. This is a reasonable assumption for high-speed entry because the flow field will form very quickly. The wall temperature is calculated by assuming radiation equilibrium. This means that the amount of heat that is being absorbed by convective heating is the same amount of heat that is being expelled through re-radiation. Also seen in the upper right of figure 4-10 are two photographs of payload bay door rollers. The photograph for OV-105 shows an intact roller, and the photograph for OV-102 (*Columbia*) clearly shows the erosion of the exposed titanium surface on the front face of the roller.

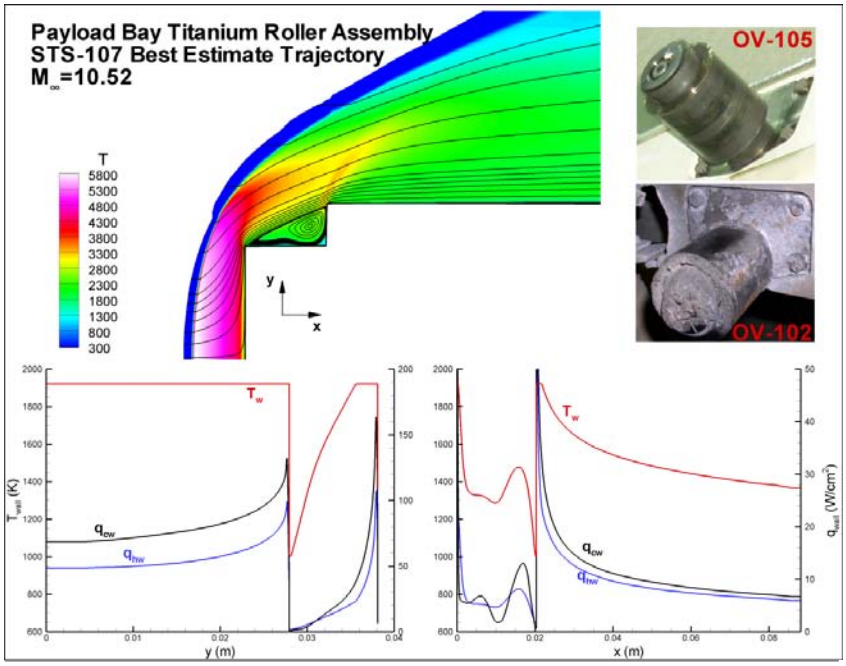


Figure 4-10. ***Computational fluid dynamics analysis of heating at the tip of the payload bay door roller for orthogonal geometry into the direction of travel at M = 10.5.***

[15]Murray, A. L., "Further Enhancements of the BLIMP Computer Code and User's Guide," AFWAL-TR-88-3010, June 30, 1988.

Radiation equilibrium is a reasonable assumption when used for analyzing the Thermal Protection System (TPS) that was designed to have a very low thermal conductivity; e.g., high-temperature reusable surface insulation tiles or reinforced carbon-carbon panels. However, for metals that will conduct the temperature inward, there is a time factor that needs to be accounted for and a temperature distribution through the object. Thus, the CFD analysis is very good at producing accurate heating rates, and the temperature calculation can give a reasonable upper bound.

Arc jet testing

It was proposed that the erosion of both the x-links and the payload bay door rollers appeared to have some element of material selectivity. Combustion was proposed as an explanation for why some materials eroded and other materials did not and also to explain how oxide formation, which requires extremely high temperatures, occurred.

A selection of enthalpy-pressure test points was chosen based on the predicted trajectory and ballistic number of *Columbia* and the free-flying forebody. A series of tests was conducted at the Boeing St. Louis Large Core Arc Tunnel plasma arc facility. The complete results of the arc-jet testing can be found in Olivas, J. D., Mayeaux, B. M., Melroy, P. A., and Cone, D. M., "Study of Ti Alloy Combustion Susceptibility in Simulated Entry Environments," *AIAA Journal*, 2008 (submitted for publication).

4.5.5 Video analysis

4.5.5.1 *Ground-based video analysis*

The CAIB Report contains information that is related to video analysis performed during that investigation.[16] The process followed in performing video analyses is highlighted below.

When the ISAG received a video, that video was "screened." The video was watched from beginning to end, identified as an ascent or descent video, briefly summarized, entered into the ISAG database, and cross-referenced to any photographic stills or other versions of the same video.

Following the initial screening, a "detail screening" was performed on videos, noting key timing events. Key timing events may include the start and end of the data tape, the first and last appearance of the orbiter, and any visual event that was significant enough to be potentially useful in time-synchronizing with other videos. The first appearance of an object is referred to in the ISAG database as the acquisition of signal (AOS), and the last appearance of an object is referred to as the LOS.

Events appearing in multiple videos could not only time-synchronize the videos but could also provide confirmation of the nature of those events (for example, whether it was a brightness change or a debris-shedding event). Visual events that may be noted in the videos include object-brightening events, changes in the trails of the objects seen, color changes in objects and/or trails, separation and/or breakup of objects, and first and last appearance of additional objects. Generally, brightening events were easier to correlate between videos than debris-shedding events.

Finding. Brightening events were easier to correlate between videos than debris-shedding events.

The AOS of an object is usually well identified. An object can have multiple AOSs and LOSs if it passes in and out of the field of view (FOV) due to tracking of the camera or changes in the magnification factor of the video. Visual separation of objects is impacted by the zoom factor of the video, the resolution of the recording device, focus, image stability, the amount of saturation of the pixels of the recording device due to the brightness of the objects in the FOV, and the viewing geometry of the viewer in relation to the

[16] *Columbia* Accident Investigation Board Report, Volume III, Appendix E.4 *Columbia* Early Sighting Assessment Team Final Report, June 13, 2003, details a combination of video, photograph, radar, and ground debris information and details data submission handling. *Columbia* Accident Investigation Board Report, Volume III, Appendix E.2, STS-107 Image Analysis Team Final Report, details the investigation of the imagery of the STS-107 entry (as well as imagery of the launch and suspect foam strike).

objects and the sun. The time of visual separation of objects in an FOV will always be later than the actual time of physical separation, with delays ranging from a few video frames (1 frame ≈ 1/30 second) to several seconds.

The LOS time for an object generally indicates that the object left the FOV rather than indicating the loss of a detectable visual signal. Typically, the videographers zoomed the FOV to concentrate on the largest or brightest object that was visible (typically the orbiter or the aftbody/engines). If an object left the FOV, it can be inferred that the object probably had a lower ballistic number (higher drag) than the object that the videographer was tracking. On rare occasions, LOS equated to an inability to track the object due to multiple pieces seen and/or a scintillating (flashing) visual signal. There were also cases when the visual signal dropped below the sensitivity of the camera or the object disintegrated. The cause of the LOS was rarely noted in the database.

The video key events times were used to develop event sequences for the CAIB. After the CAIB event sequence timeline was completed, a Late Re-entry Working Group (LRWG) was formed in support of the CSWG. The LRWG's primary tasks were to determine when the CM separated from the rest of the vehicle and when it broke up, any visible event that could relate to loss of cabin atmosphere, and any large deceleration event (i.e., an abrupt slowing of the CM). The LRWG results fed into some CSWG results and fed extensively into the SCSIIT work. A few videos provided good views of the breakup of the aftbody, but these were not studied in great detail since they were not relevant to the SCSIIT.

The SCSIIT used the time-synchronizing of the ISAG and the LRWG when possible. The "Apache" video source has an accurate GMT time since it is based upon the Global Positioning System (GPS). All other videos from the eastern timeline were synchronized from Apache video, if possible, and a few videos had adequate timing information to allow cross-checks of GMT. The times for video events have estimated errors ranging from ±0.3 second up to ±2 seconds, although generally they were less. Errors that were associated with video event times are impacted by the magnification of the FOV, resolution, and viewing geometry. For example, an image with a high magnification for the camera FOV will see the start of the CE sooner than an image from an FOV that was not zoomed. Resolution may delay timing for similar reasons. Viewing geometry can prevent an event from being seen until later, if it is seen at all. A combination of error sources can lead to an accuracy of ±1 second for defined events within a single video or between videos, although the actual error may be better or worse by up to an additional 1 second.

Although all 51 videos depicting the orbiter over Texas were examined and contributed to understanding the events, five videos were found to be key for the LRWG's investigation (see Table 4-5).

Table 4-5. ***Videos That Were Used***

Reference EOC No.	City in TX	Name	Latitude	Longitude
EOC2-4-0024	Arlington	Arlington	32.7	–97.1
EOC2-4-0209-B	Hewitt	Hewitt	31.4	–97.2
EOC2-4-0221-4	Mesquite	WFAA4/Mesquite	32.8	–96.6
EOC2-4-0221-3	Fairpark	WFAA3	32.8	–96.7
MIT-DVCAM	Fort Hood	Apache	31.2	–97.6
EOC2-4-0077	Burleson	NBC*	32.5	–97.3

*NBC = National Broadcasting Corporation

Late in the SCSIIT investigation, a previously unanalyzed video was discovered. Some videos received from television affiliates were compilations of video that had been collected by television camera operators and those that were submitted by the public. These compilations were not noted as "ascent" or "descent" videos if they were a mix of videos from both phases of the mission. Additionally, the original review of videos focused on identifying pre-CE videos, and this video was misclassified as recording events after the CE. It had been reviewed within the first 5 days of the video investigation related to the CAIB, when the importance of the video evidence was not fully realized. Because the video compilation did not have a "descent" annotation, both the LRWG and the SCSIIT were unaware of its existence. Additionally, because no team

had initially identified it as a critical video, the original was not requested in a timely fashion. A copy of part of the original video from the television station was obtained (in addition to a copy of the full video in the ISAG archives), but the original video no longer exists. The copy provides adequate resolution for data interpretation. This video, which is designated "NBC" (EOC2-4-0076-B), added several seconds of good-quality imagery to the eastern timeline.

Finding. Not all videos segments within compilations were individually categorized. Not all videos were re-reviewed once a better understanding of events had been gained.

> ***Recommendation A11.*** All video segments within a compilation should be categorized and summarized. All videos should be re-reviewed once the investigation has progressed to the point that a timeline has been established to verify that all relevant video data are being used.

It was originally assumed that the sun would have little impact on the brightness of the debris pieces. If a piece of debris was generating a trail, it was expected that the debris was self-illuminating due to the thermal effects of entry heating. Objects that were tumbling were not expected to vary in brightness solely due to the sun. However, review and comparison of videos showed that the sun did contribute to the variations in illumination of objects, even those generating trails. Illumination by the sun impacted the visibility of all debris.

Finding. Sun angle illumination impacted the visibility of debris in video recordings.

In establishing the sequence of events as seen from video and photographic imagery, there are brightening events and debris-shedding events. The appearance of these events was impacted by illumination angle, viewing angle, viewing geometry, and timestamp accuracy. Additionally, the accuracy of the timeline and related data was dependent on information from the videographers. A standard information sheet was used that helped to ensure that most, if not all, key information was available.

4.5.5.2 *Forebody triangulation*

The CEL created a custom application to evaluate the relative motion between the forebody and the aft portion of the vehicle. The application used ground-based videos that were recorded from multiple locations and triangulation to determine the relative distance between the forebody and the aftbody. The original intent of the analysis was to identify the initial conditions of forebody separation (forebody rotation rates, separation rates, etc.). The time between the CE (GMT 14:00:18) and the first point where the forebody was visible in two videos (~GMT 14:00:30) made it difficult to draw conclusions about the forebody separation conditions. However, as described in Section 2.1.5, this triangulation analysis yielded information regarding the forebody motion after the CE. The analysis led to the conclusion that the forebody was rotating in all three axes at approximately 0.1 rev/sec.

4.5.6 Debris mapping/plots

Debris field analysis was conducted to investigate where components of the orbiter impacted the ground compared to where the debris items originated from on the orbiter. This analysis involved plotting the recovery locations of debris items, identifying debris groups based on their location on the orbiter (i.e., left wing, payload bay, FF, etc.), and comparing the relative positions of the groups. The debris groups were referred to as "clusters." Cluster analysis is a technique that uses the assumption that when a large number of debris items from the same structural zone (e.g., tail, wings, payload bay, CM) are considered, different clusters will have a similar range of ballistic numbers. With a similar range of ballistic numbers, the centroids of the clusters can be evaluated *relative* to each other to approximate sequencing of key events.

Ballistic analysis for selected debris items within the various clusters provided calculated release times, and the release times were used to develop ballistics-generated breakup sequences. These sequences yielded similar sequences to the cluster analysis-generated sequences, providing confidence in the breakup sequences that are described throughout this report.

Latitude and longitude errors

Performing these analyses required accurate latitude and longitude data in a format that was usable by the plotting application. The majority of the items in the KSC debris database include latitude and longitude data of the recovery location. In some cases, no latitude/longitude information exists because debris were recovered and submitted by local residents. In other cases, the latitude/longitude data may have been recorded in the field but not imported into the KSC debris database. The data that do exist in the KSC debris database were assumed to be accurate unless there were obvious factors that indicated otherwise, such as the resultant plot indicating that an object was recovered beyond the reasonable bounds of the debris field.

Much of the data that are in the KSC debris database are recorded in degrees and decimal degrees (DD.ddddd), for example: 31.31063° N, 93.87701° W. However, the database contained several other formats, including 31.21.26.3; 31 21′ 26.3″; 31 21 26.3; 31 21.263; and 31.21.263.

The first three formats were assumed to indicate degrees, minutes, seconds, and decimal seconds (DD MM SS.sss), unless information indicated otherwise. The last two formats were assumed to indicate degrees, minutes, and decimal minutes (DD MM.mmm), unless information indicated otherwise.

To be usable by the plotting application, the data were converted into the degrees and decimal degrees format (DD.ddddd). In many cases, there were indications that the data would not need to be converted, despite the placement of decimal points or spaces. If the “minutes” or “seconds” numbers were greater than 59 in either the latitude or longitude, the data were assumed to represent degrees and decimal degrees, regardless of the placement of extra decimal points. For example, although “31.87.70.1” appears to be in the DD.MM.SS.s format, it cannot be because 87 minutes and 70.1 seconds are not valid coordinates. Therefore, this latitude, and its corresponding longitude, would be used as 31.87701° (DD.ddddd) without converting the numbers. In the absence of conflicting cues, the latitude/longitude data would be converted according to the assumptions described above.

The lack of a single, standard data format for latitude/longitude data and the potential ambiguity that is associated with the need to convert data of different formats resulted in possible data errors. One possible error type is due to the numbers being entered in the DD.ddddd format when they actually represent another format. This results in a *lack of conversion*. For example, 31° 23′ 41.410″ was entered in the database as 31.2341410°. The format implies that no conversion to DD.ddddd is necessary, but the original numbers actually were in the DD MM SS.sss format and should have been entered in that format (making the need for conversion obvious). Another error type is due to the numbers being entered in a format other than the DD.ddddd format when they actually represent the DD.ddddd numbers. This results in an *unnecessary conversion* being performed on the numbers. For example, 31.31063° was entered into the database as 31.31.06.3 and erroneously converted to 31.51842°.

Finding. The lack of a single, standard data format for latitude/longitude data and the potential ambiguity that is associated with the need to convert data of different formats resulted in possible data errors.

Figure 4-11 shows a plot for a pair of number strings (31 39439 and 94 53462) that is interpreted for three different latitude/longitude formats. Point A shows the numbers interpreted as the DD.ddddd format (31.39439° N, 94.53462° W). Point B shows the numbers as a DD MM SS.sss format (31° 39′ 43.9″ N, 94° 53′ 46.2″ W). Point C shows the numbers as a DD MM.mmm format (31° 39.439′ N, 94° 53.462′ W).

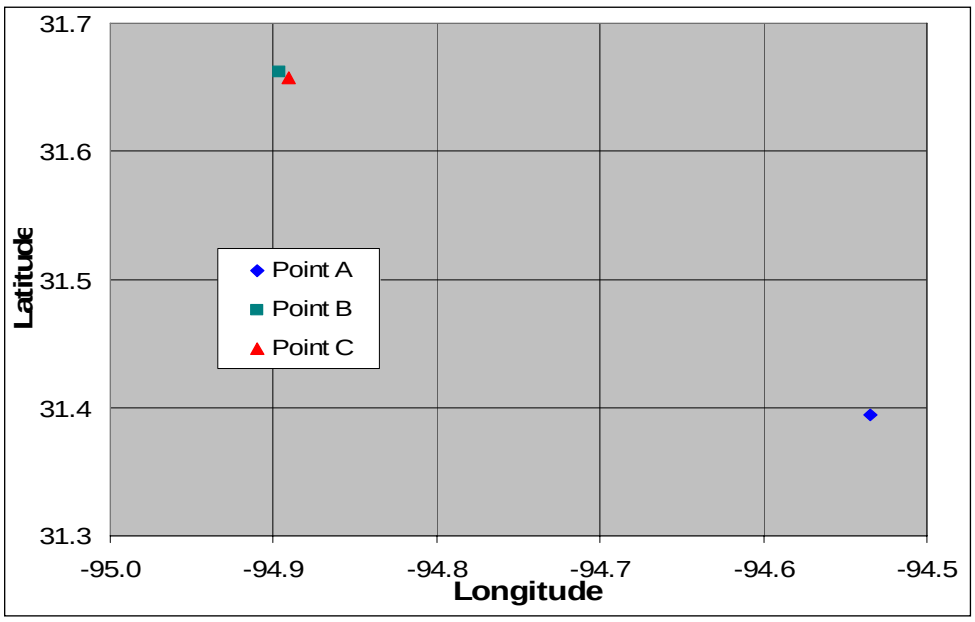


Figure 4-11. ***Plot for a pair of number strings that was interpreted for three different latitude/longitude formats*****.**

In this example, Point A is approximately 28 miles from Points B and C; while Points B and C are less than six-tenths of a mile apart. This example demonstrates the magnitude of error that can occur if the wrong data format is assumed and a necessary conversion does not occur (or an *un*necessary conversion is performed).

To avert these errors and increase the confidence in the debris-plot-related analyses results, all forebody debris of critical importance that had suspect latitude and longitude data were researched to confirm or correct the latitude and longitude data recorded in the KSC debris database (or to provide missing data). Correcting the data involved researching the original field data sheets and the field photographs taken with GPS receivers in view, and, in some cases, using the field descriptions of the area ("100 feet north of county road X, one-quarter of a mile east of the intersection of county road X and county road Y") and GPS/mapping software to establish corrected latitude and longitude data. This was a labor-intensive task, but the need for accurate debris recovery locations warranted the effort. Accurate debris recovery locations resulted in a higher degree of confidence in the debris plotting analyses and the ballistics-related analyses.

> ***Recommendation A10.*** Global Positioning System receivers used for recording the latitude/longitude of recovered debris must all be calibrated the same way (i.e., using the same reference system), and the latitude/longitude data should be recorded in a standardized format.[17]

4.5.7 Structural analysis

The recovered components of the CM and FF were studied in depth to provide information relating to failure mechanisms and timing. Areas of primary interest were the identified debris from the primary load bearing structures, windows, and hatches.

Figures were taken from drawings in the Orbiter Structures CATIA library, Version 4, which was developed by Boeing Engineer Daren Cokin.

[17]STS-107 *Columbia* Reconstruction Report, NSTS-60501, June 30, 2003, p. 142.

OV-102 structural loads were taken from the following series of volumes:

- SSD96D0095, Volume 5, Book 1, OV-102 Structural Analysis for Performance Enhancement, Crew Module-Shell Structure, October 1997.
- STS 89-0537, Volume 5, Book 1, OV-102 Structural Analysis for 6.0 Loads, Crew Module-Shell Structure, Addendum A, October 1995.
- STS 89-0537, Volume 4, Book 1, OV-102 Structural Analysis for 6.0 Loads, Forward Fuselage Upper/RCS, Addendum A, October 1995.
- STS 89-0537, Volume 3, Book 1, OV-102 Structural Analysis for 6.0 Loads, Forward Fuselage Lower, Addendum A, October 1995.
- STS 89-0537, Volume 15, Book 1, OV-102 Structural Analysis for 6.0 Loads, Mid Fuselage, December 1992.

Estimated loads from aerodynamic simulations were compared to the structural load limits from the volumes cited above. The results from video analysis, debris field analysis, and debris inspection/analysis were used to determine the sequence of breakup. Engineering judgment and knowledge of the CM structure were used to assess the breakup.

4.5.7.1 *Structural capability*

When the orbiter was designed, structural analysis was performed to ensure that the CM and/or vehicle would maintain integrity under nominal conditions and a few defined off-nominal cases such as a crash landing. However, little work was done at that time to characterize failure modes in an effort to understand what might happen in a catastrophic situation. During this investigation, analysis was performed to determine the structural capabilities of the orbiter and the CM, and to predict failure modes and failure locations assuming certain load cases derived from the accident scenario.

The primary tools that were used to perform the assessments were structural certification documents, NASTRAN finite element structural analysis software code (using detailed finite element models of local structure), and a medium-fidelity, global-finite element model of the orbiter. With these tools, investigations were completed to assess the vehicle's flight recorder loads, CM skin stress, CM attachment linkages and fittings forces, window thermal shock, window frame distortion, crew seat structural failures, and CM floor loading.

The objective of these assessments was to gain additional insight into CM structural performance and CM structural failures consistent with the accident debris. Load comparisons, conventional structural analysis approaches based on known conditions, and parametric analyses were the primary approaches that were used.

Conventional structural analyses were performed when primary inputs permitting this type of analysis were known, or when reasonable assumptions could be made regarding structural configuration, structure temperature, and loading. These analyses were performed using the flight data accelerations prior to LOS, while the vehicle was essentially intact.

Parametric assessments were required when less information was available. Since structural certification documents are based on nominal design conditions, the parametric assessments provided information about the structural response outside the design envelope. Exposure of the orbiter vehicle structure to high heating and changes in structural configuration required a parametric approach because, in many cases, the effects of heating shifted structural failure modes and failure locations away from areas that were documented as minimum capability zones in the certification reports. Parametric analyses were used to predict trends for structural failure with increasing temperature, stress contours for combined levels of pressure and inertial loading, and failure propagation as load paths changed.

4.5.7.2 *Crew seats*

In an attempt to determine individual crew load profiles, the team set out to identify seat debris items to specific seat locations. Configuration management records (tracking the serial numbers for seat components to the top-level seat assembly's serial number) were not accurately maintained, so identifying the locations of components by any surviving piece-part serial number was futile. The exceptions to this were components that were associated with the inertial reels, and all six recovered upper seatbacks were identified to specific seat locations.

Some seat debris items were identified to seat locations based on being attached to identifiable floor panels. One item was identified due to its unique application to one seat location. Most other seat components were identified to seat locations based on matching them to identified pieces (upper seatback components or pieces that were attached to floor panels). This process of matching pieces was very time-consuming and laborious. Eventually, slightly less than half of the recovered seat structure debris items were positively identified to specific seat locations. Had the individual seat components been permanently marked with serial numbers and those serial numbers tracked to the assembled seats using rigorous configuration management and control, reconstruction and identification would have been much easier and a higher percentage of pieces could have been identified to specific seats.

Seat structure debris items were studied to determine seat failure modes with the intent of determining the forces (and, therefore, CM dynamics) and the thermal profile. The analyses included gross and microscopic inspection of fracture surfaces, scanning electron microscopy of selected sectioned items, energy dispersive X-ray spectroscopy, and comparison to structural analysis of the seat design. These analyses provided information on the temperatures that were experienced by the seat components. Additionally, the identified failure modes revealed the loading conditions, directions, and dynamics.

Inertial reel mechanisms and straps were examined to obtain information on the seat restraint loading history (and, therefore, loads on the crew members), the thermal history, CM breach timeline, and crew separation timeline. Analyses included gross and microscopic inspection of the inertial reel mechanisms, straps, inertial reel housings, and upper seatbacks. Emphasis was placed on looking for evidence of loads (deformation or witness marks on the inertial reel components, failure modes and locations of the straps, deformation of inertial reel mounting hardware), and material deposits (metal deposits on the straps, melted strap material deposits on the inertial reel housings and seats).

4.5.8 Materials analysis

Materials analyses were performed by the JSC Materials and Processing Office, the JSC Astromaterials Research Office, the KSC Failure Analysis and Materials Evaluation Branch, and the White Sands Test Facility to characterize the deposition (char) on the window panes. Additionally, Langley Research Center performed materials testing on the seat fragments.

Analyses included performing microscopy of debris items to determine fracture dynamics and mechanisms, determining the compositions of materials deposited on the debris, testing materials to determine materials properties, and heating pristine materials samples in an attempt to match debris observations.

The following techniques were used for materials analysis:

- Optical microscopy including polarized light.
- Electron microscopy including scanning electron microscopy, transmission electron microscopy, and scanning tunneling electron microscopy.
- Energy dispersive X-ray spectroscopy.
- X-ray fluorescence spectroscopy.
- Fourier transform infrared spectroscopy.
- Focused ion beam milling.
- Selected area electron diffraction.

- X-ray diffraction.
- Backscattered Kikuchi diffraction pattern.
- Differential scanning calorimetry.
- Thermal gravimetric analysis.

4.5.8.1 *Crew seats*

Analyses were performed on the material deposited on the seat components to yield information relating to the CM breakup sequence and the thermal profile. These analyses included determining deposition patterns (portions of the seats containing deposits and directionality of the depositions) and analysis of the deposited material combined with investigation of vehicle materials to determine possible sources of the deposits.

Microscopy (optical and electron) and spectroscopy techniques were performed on the seat structures to determine failure mechanisms and thermal exposure.

4.5.8.2 *Boots materials thermal testing*

Thermal testing was performed on suit boot soles in an attempt to match the observed thermal damage. Boot soles of flight-like boots were heated in an oven to identify the range of thermal effects with varying thermal exposure. The test samples were exposed to 750°F (399°C), 1,000°F (538°C), or 1,250°F (677°C) at normal atmospheric pressure conditions (~14.7 pounds per square inch (psi), ~20% oxygen (O_2)) for 15, 30, 45, or 60 seconds. The materials showed no significant change in appearance until they combusted. This puzzled the team initially until it became clear that the presence of O_2 was affecting the results. The tests were repeated using new samples that were heated in a nitrogen (N_2) purge (<3% O_2). The samples were then compared to the recovered boot sole fragments. The results of the revised test protocol appeared similar to the recovered boot soles. The test samples that most closely matched the recovered debris items were those that were exposed to 1,000°F (538°C) for 30 to 45 seconds or 1,250°F (677°C) for 15 to 30 seconds. However, there is no credible scenario in which the *Columbia* boots would be exposed to these temperatures for the length of time indicated by the tests, so the test results could not be correlated directly to the debris observations. Because the test conditions (~14.7 psi, 97% to 99% N_2, 1% to 3% O_2) did not accurately approximate the entry environment conditions (low ambient pressure, monatomic oxygen, and possibly high dynamic pressure), they are a potential source of error in this analysis.

4.5.8.3 *Helmets*

Thermal gravimetric analysis was performed on helmet materials to determine the temperatures at which thermal decomposition (pyrolysis) begins. Determining these temperatures allowed the team to determine the thermal exposure of the helmets, which was used to predict helmet release times using ORSAT.

4.5.9 Medical processes

The SCSIIT Crew Team performed an extensive examination of all available medical evidence, performed additional studies including motion, thermal, and ballistic analyses, and correlated all of this information with the results of the other SCSIIT teams to develop a detailed crew event timeline sequence, cause(s) of death, threat matrix, and survival gap. NASA used an independent consulting firm specializing in injury analysis (BRC, San Antonio, Texas) to independently identify the threats with lethal potential that were faced by the STS-107 crew members and any mitigations that were in place at the time of the accident vs. those that are possible (i.e., the survival gap).

4.5.9.1 *Crew event timeline*

The crew event timeline development began with a review of the recorded air-to-ground (A/G) audio to identify any relevant information, such as timing of crew actions and the state of crew event awareness. The recovered videos, showing middeck and flight deck activities, were also reviewed and correlated with the A/G audio and recorded telemetry.

Forensic temporal markers were used to determine the sequence of observed injuries. The medical forensic evidence was then correlated with the audio and video evidence, recorded telemetry, and analysis of the recovered flight crew equipment. These correlations helped establish the configuration of the crew escape equipment at the time of the accident. All of this information was then used to develop the detailed crew event timeline.

4.5.9.2 *Cause of death determination*

The cause of death, blunt force trauma and hypoxia[18], was originally determined at the time of the autopsies, which were conducted by the AFIP.[19] During the CAIB investigation, a team that included members of the AFIP and FBI ensured definitive identification of the bodies, performed autopsies on the crew, provided photographic and X-ray documentation of the human remains, collected tissue specimens, provided sub-specialty interpretation of the microscopic slides, and performed toxicological analyses and spectral analysis of skin deposits. The data generated by the initial examining team were used by the SCSIIT as a starting point for determining what the crew experienced, the temporal sequence of events, and how these events related to the cause of death.

The standard AFIP autopsy protocol that was followed for determining the cause of death was subsequently found to be inadequate to address some of the unique aspects of a hypersonic, high-altitude-entry accident and did not include the collection of unique evidence that would have helped to better understand the sequence of events that the crew experienced and the unique injuries that were incurred from exposure to such extreme events and conditions. Although somewhat limited, the information that was derived from the autopsy reports was invaluable and was used to develop a matrix of the injuries inflicted on each crew member. This basic injury matrix was then expanded, based on additional studies and data reviews conducted by the SCSIIT Crew Team, and integrated with information derived from X rays, photographs, and additional reviews of the histology material. The expanded injury matrix was then used to identify any common injury patterns among the crew members. This also facilitated the identification of injury patterns based on crew location (i.e., flight deck vs. middeck, starboard vs. port).

All of the X rays and autopsy photographs were reviewed. A subset of the X rays and photographs, which was identified as having potentially useful information, was digitally enhanced and re-examined. This effort identified additional information that was not previously noted.

All of the histology slides were also re-examined to aid in confirming the cause of death and determine the temporal sequencing of injuries.

The Anthropometry Biomechanics Facility in the Habitability and Environmental Factors Division of the JSC Space Life Science Directorate was used to conduct medical analysis. A Vitus 3-dimensional scanner[20] was used to scan a SCSIIT member for use as a baseline in the suited and unsuited (i.e., minimally clothed) configuration. The anthropometry of the subject was recorded and compared to the STS-107 crew anthropometry data. The baseline subject's legs, torso, arm lengths, and shoulder and hip breadths were scaled using a percentage of the baseline subject's lengths to the STS-107 crew member's lengths to match each crew member's unique anthropometry, thus providing an anthropometrically correct model for each STS-107 crew member. The end results were scaled 3-dimensional (3-D) models representing the crew members. A simple 3-D skeleton model was also scaled to match the anthropometric measurements of each crew member. Each model was converted into a 3-D mesh and imported into a modeling tool called 3DStudioMax. Using the modeling software, the scaled suited mesh was attached to an underlying scaled skeleton that allowed all the linked scans to move in concert. After the individual crew member models were developed, they were incorporated into a 3-D computer model of the *Columbia* cockpit (developed by the CEL). Autopsy reports, dermatopathology data, and a review of the field intake and autopsy photographs were used.

[18] *Columbia* Accident Investigation Board Report, Volume I, August 2003, p. 77.
[19] The AFIP is the U.S. government agency that is authorized to conduct autopsies on astronauts flying on U.S. spacecraft who die in the line of duty.
[20] Vitus is a 3-dimensional scanner that records a digitized image of the body using a laser to capture a surface image of an object. This surface image, which is a volumetric representation of the subject, can then be saved digitally and used to take anthropometric data.

The resulting models were then reviewed by the Crew Team. Patterns were analyzed to determine sequencing and potential injury sources.

All of this information was then integrated with the results of other SCSIIT sub-teams to form an integrated medical scenario and generate augmented autopsy reports that included all updated information and a final determination of the injury sequence leading up to death.

4.5.10 Other methods and processes

4.5.10.1 *Hygiene/drink package depressurization testing*

Intact hygiene bottles and empty (used) drink bags were recovered in the *Columbia* debris field. The fact that these items were not ruptured may indicate a depressurization rate that the CM experienced. Therefore, rapid depressurization tests were conducted in an attempt to determine the maximum depressurization rate that these packages can sustain without rupturing, determining an upper bound for the *Columbia* cabin depressurization rate.

Each trial consisted of testing shuttle-type shampoo bottles and drink bags. Testing was conducted at approximately 14.5 psi/second, approximately 18 psi/second, and approximately 31.5 psi/second. No packages ruptured at any of these rates. Because none of the shampoo bottles or drink bags ruptured during the depressurization tests indicates that the items are capable of withstanding depressurization rates greater than approximately 31.5 psi/second (which would represent an instantaneous, explosive depressurization of the CM – a scenario that is not supported by debris evidence). Therefore, no conclusions could be made regarding the rate of depressurization of the *Columbia* CM based on these tests.

4.5.10.2 *Advanced crew escape suits and crew worn equipment*

The recovered components of the crew worn equipment (advanced crew escape suit (ACES)), survival gear, parachute harness, and parachute pack) were studied in depth to provide information relating to crew injuries, CM breakup sequence, crew separation sequence, and suit disruption mechanism and timing.

Almost all of the approximately 75 recovered non-fabric components were identified to specific crew members. This identification was possible because most of the suit components and subcomponents were marked with serial numbers, and the serial numbers were recorded and tracked to the suit assembly, which was tracked to a specific crew member. In many cases, the serial numbers were stamped or etched on the hardware items and, therefore, survived the entry environment. In some cases, damage to the exterior of the component precluded reading the serial number, so other means were used to identify the hardware items. For the search and rescue satellite aided tracking beacons and the Army/Navy personal radio communications (A/N PRC) 112 radios, the processing modules were extracted and interfaced with ground support equipment to “read” the unique identifier codes of the specific unit. For the Sea Water Activated Release System (SEAWARS), the devices were disassembled and unique markings on the electronics packages were discovered. The vendor was contacted and the team was able to trace the unique markings to specific serial numbers for the SEAWARS units.

The ability to ascribe recovered crew worn items to specific crew members was critical to being able to draw conclusions based on the crew worn equipment.

4.5.10.3 *Helmets*

Visual inspection was performed on the helmets to qualify the helmet structural damage. Several nondestructive evaluation techniques were used in an attempt to determine the extent of thermal and mechanical damage beyond what the visual inspections revealed, with an emphasis on sub-surface damage. Computer tomography scans, real-time X-ray, thermographic (pulse flash and through transmission), and ultrasound testing were used to determine the structural condition of the fiberglass shell. These techniques validated damage sites that were seen in visual inspections and revealed some additional damage areas, but did not produce significant additional findings.

4.5.11 Cabin depressurization modeling

The Killer Press Model is a Microsoft Excel spreadsheet that was developed and used by NASA flight controllers to calculate pressure equalization times for two or three volumes connected in series. The user inputs the starting volume, pressure, and temperature of the air in the volumes in question and parameters for the venting path between the volumes. In all cases, the model assumes a circular vent path between volumes. The model outputs pressure with respect to time for the volumes. The cabin depressurization curves in Section 2.3 were generated using version 2.06 of Killer Press. These plots show the differential pressures between three volumes: the flight deck and middeck volume of the CM; the lower equipment bay of the CM; and the ambient atmosphere.

The following parameters were used as inputs: flight deck/middeck volume = 2,163 $feet^3$, pressure = 14.7 psi, temperature = 70°F (21°C); lower equipment bay volume = 337 $feet^3$, pressure = 14.7 psi, temperature = 70°F (21°C); vent path between lower equipment bay and middeck is ~50 in^2 (8-in.-diameter hole); ambient atmosphere volume = 1×10^{20} $feet^3$, pressure = 0.022 psi, and temperature = –67°F (–55°C).

Future Work

Future Work

Chapters 2, 3, and 4 outline the details of the analyses and assessments that were performed to understand the events with lethal potential that occurred. Many findings, conclusions, and recommendations were documented. However, it is also important to discuss the work that was not performed.

The Spacecraft Crew Survival Integrated Investigation Team (SCSIIT) was intentionally established as a small team. This allowed the team to maintain subject confidentiality, which was very important prior to publication of the report. However, the small size of the team, the resources that were available given Return to Flight and other very important programs, and the fact that most of the team members also had to continue their "regular" job assignments resulted in limitations to the investigation. Some items of interest were not accomplished due to lack of resources and schedule. A selection of future projects that may be of interest to complete in the future is listed below.

Although the list of testing and analyses below did not have significant influence on the SCSIIT conclusions and recommendations, the team believes that the data gained from these test and analyses will increase the knowledge base of the accident dynamics, spacecraft hardware, and materials commonly used in space vehicles.

Spacecraft Accident Investigation Database. To develop a database for future investigators that will be cross-referenced across all spacecraft accidents to collect information in a single location.

Suit Destruction Testing. To determine the forces that are required to disrupt the suit (remove helmet shell from helmet neck ring, remove suit-side neck ring from the suit, remove wrist rings from the suit), and to perform an analysis to determine the aerodynamic loads/windblast (Qbar or knots equivalent air speed (KEAS)) that is required to attain the above-mentioned forces with the end goal of determining the suit's actual windblast capability. This evaluation should be done for visor down and visor up configurations.

Materials Analysis/Testing. To positively determine the sources of the materials that were found deposited on various debris items. Also, testing should be done on suit materials (Nomex, GORE-TEX®, nylon), seat straps, and boot materials in a low ambient pressure (near vacuum), high heat, and atomic oxygen environment to simulate the environment at the Crew Module Catastrophic Event (CMCE) to understand how nonmetallic materials react in the thermal and chemical environments of spacecraft entry.

Static and Dynamic Testing on Seat Restraints. To determine the forces that are required to fail the inertial reel strap with the strap fully extended and the strap partially retracted, and the forces that are required to fail the straps when at elevated temperature (200°F (93°C), 400°F (204°C), 600°F (316°C)).

Helmet Destruction Testing. To determine the impact forces that are required to cause physical damage to the helmet.

Emergency Oxygen System (EOS) Bottle Testing. Investigate using microscopic analysis, materials analysis, etc. to determine the failure mode of the EOS bottle fragments and analyze the fracture surfaces to see whether the bottles were pressurized when they were fragmented.

Window Glass Analysis. To evaluate which window glass fragments collected in the field are from *Columbia* and which are non-orbiter.

Shock Wave Analysis. To better understand the effects of shock waves on aerodynamic heating in the hypersonic flight regime.

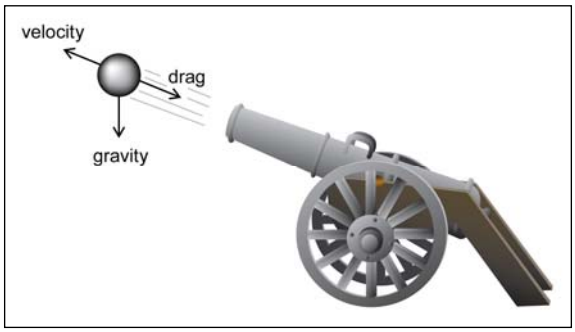
Titanium Combustion Analysis. To fully characterize the behavioral properties of titanium in an entry environment, including those that can lead to combustion.

Challenger *Analysis*. Complete an analysis on the *Challenger* debris to compare and contrast with the *Columbia* findings.

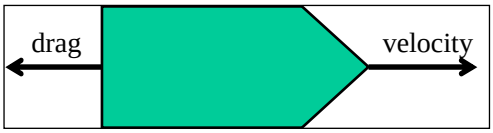
Appendix – Ballistic Tutorial

Appendix – Ballistic Tutorial

This is intended to give a reader a general understanding of ballistic number and ballistic trajectories. The objective is to give an intuitive understanding of the concepts rather than a rigorous derivation.



When the trajectory of an object is described as being ballistic, that means that the object has no control over its trajectory. As an example, think about a cannonball fired out of a cannon. Once the cannonball leaves the barrel of the cannon, the forces determining its trajectory are its own momentum (mass multiplied by velocity), gravity, drag, and winds.



Drag is the force that slows an object down as it travels through the atmosphere, and it is always opposite the velocity vector.

Drag is a function of thickness of the atmosphere, the velocity, reference area, and the coefficient of drag, C_D. The equation for the drag acceleration is

$$Drag\ \ acceleration = -\frac{1}{2} C_D \frac{Area}{Mass} \rho v^2$$

where C_D = coefficient of drag
Area = reference area
ρ = atmospheric density (thickness)
v = velocity

The minus sign indicates that drag acts opposite to the direction of travel. The coefficient of drag, C_D, is a function of the shape, orientation, and velocity of an object.

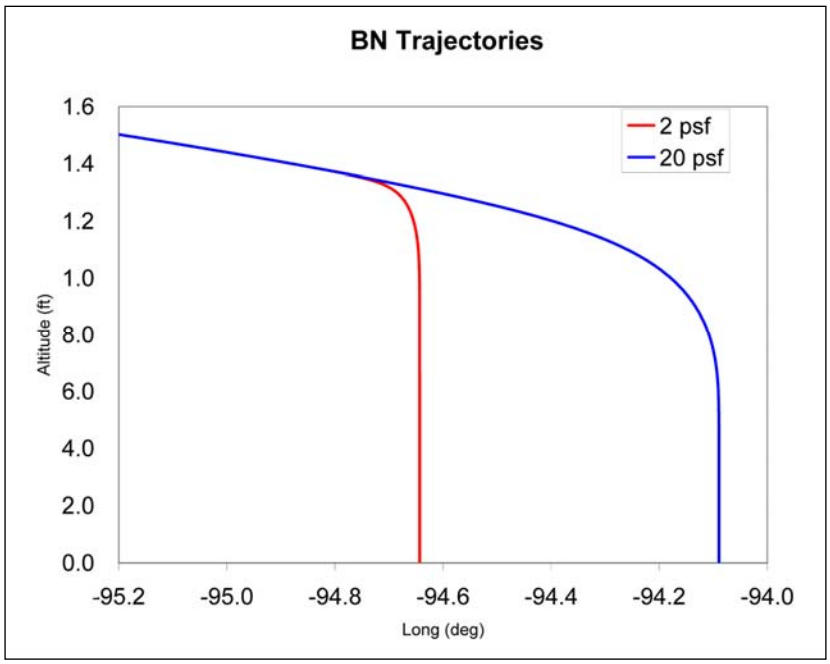
The ballistic number (BN) of an object can be thought as a measure of how far downrange the object will travel. The equation for the BN is

$$BN = \frac{W}{C_D * Area}$$

where W = weight

The units for BN are pounds per square foot (psf) (C_D does not have units associated with it).

Objects with large BNs will generally travel farther downrange than objects with smaller BNs. The plot below shows the trajectories for two objects with different BNs, 20 and 2 psf. The red line is the trajectory of the 2-psf objects and the blue line is the trajectory of the 20-psf object.

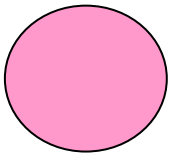


Let's look at how each of the parameters in the BN equation, weight, C_D, and area, affect the BN. First we'll look at weight.

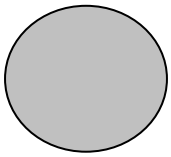
How Weight Affects BN

Two spheres, one is made of rubber and the other is made of aluminum. Both spheres have the same area and C_D.

Area = 0.087 ft^2 (a radius of 2 in.)
C_D = 1



Rubber
W = 1.3 lbs.
BN = 14.9 psf



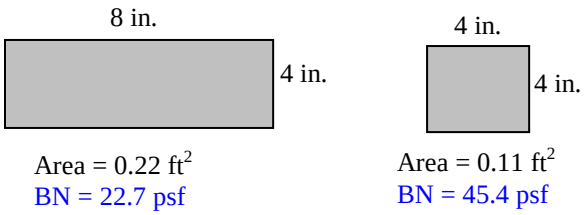
Aluminum
W = 3.3 lbs.
BN = 37.9 psf

The sphere with the larger weight, the aluminum sphere, will have a larger BN. If both spheres were released at the same altitude and velocity, the aluminum sphere would travel farther downrange than the rubber sphere because of its larger BN.

How Area Affects BN

For this example, there are two boxes, both made of aluminum. They have the same weight and C_D. The only difference is the area.

Weight = 5 lbs.
C_D = 1



Area = 0.22 ft^2
BN = 22.7 psf

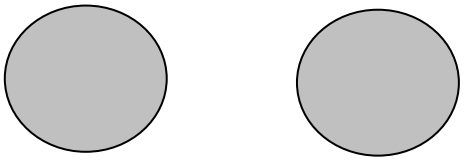
Area = 0.11 ft^2
BN = 45.4 psf

The box with the smaller area will have a larger BN. If both boxes were released at the same altitude and velocity, the box with the smaller area would travel farther downrange because of its larger BN.

How C_D Affects BN

Two spheres, both made of aluminum with the same weight and area. The only difference is that they are released at different velocities that give them different C_D's.

Area = 0.087 ft^2
Weight = 3.3 lbs.



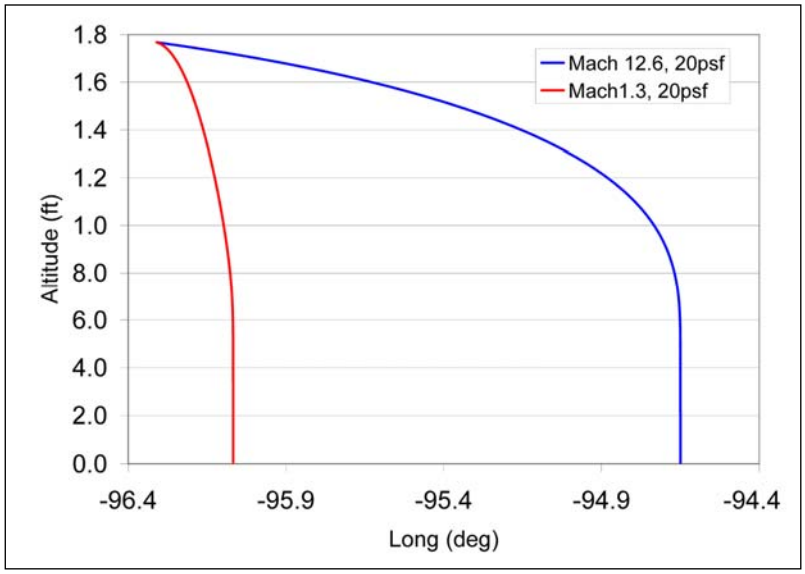
C_D = 1.16
BN = 32.7 psf

C_D = 1.71
BN = 22.2 psf

The sphere with the smaller C_D will have a larger BN. But in this particular example, the sphere with the larger BN may not have the larger downrange distance because the initial velocities were different. The next section looks at how the initial velocity affects a ballistic trajectory.

How Initial Velocity Affects Ballistic Trajectories

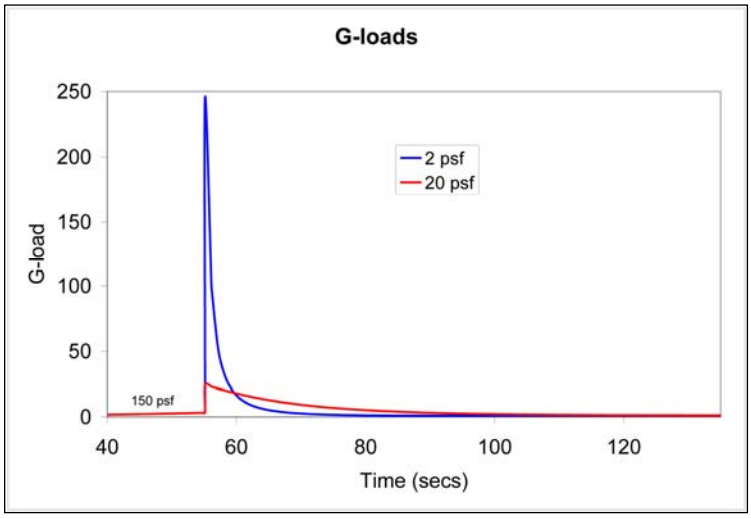
Two spheres have the same BN, which is equal to 20 psf, and are released at the same altitude. But they are released at different initial velocities, Mach 12.6 and Mach 1.3. The plot at the top of the following page shows the trajectories of each sphere. The red line shows the trajectory of the sphere that is released at Mach 1.3. The blue line shows the trajectory of the sphere that is released at Mach 12.6.



Even though both spheres have the same BN, the sphere that is released at the higher velocity will travel farther downrange. The initial conditions of the ballistic trajectory play a role in determining the downrange distance of an object.

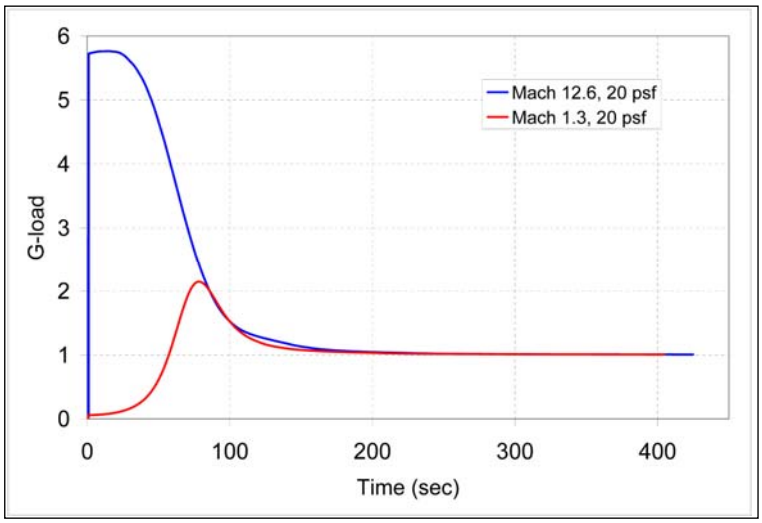
G-loads

Another topic that is related to BN is G-load. G-loads are the accelerations acting on an object divided by Earth's gravitational acceleration. When an object with a small BN is released from an object with a larger BN, the object with the smaller BN will experience a sudden change in the accelerations acting on it. This sudden change in the accelerations is called a G-load spike. As an example, let's go back to the 20 psf and 2 psf objects that we looked at in the beginning. Both of these objects are released from a larger 150-psf object. The plot below shows the sudden change in acceleration that each object experiences. The black line before the G-load spikes is the 150-psf object, the red line is the 20-psf object after release, and the blue line is the 2-psf object after release.



The 2-psf object experiences a much greater change in acceleration after being released than the 20-psf object. The greater the difference between the BNs of two objects, the greater the G-spike.

In a previous example, we examined the effect of initial velocity on two objects with the same BN. What would their G-loads look like? The G-loads for the Mach 12.6 and Mach 1.3 spheres are shown in the plot below. The red line is the Mach 1.3 object, and the blue line is the Mach 12.6 object.



The object with the higher release velocity had a larger G-load spike. The reason for this is found in the drag acceleration equation. It is a function of velocity squared. So, the higher-velocity object will experience a greater drag acceleration than the lower-velocity object.

$$Drag\ \ acceleration = -\frac{1}{2} C_D \frac{Area}{Mass} \rho v^2$$

But why did the G-load of the object at the lower initial velocity ramp up? That object's velocity actually increased after release, increasing the drag acceleration, which increased the G-load.

To sum it up, a ballistic object has no control over its trajectory. The BN is a general measure of how far downrange an object will travel. The larger the BN, the farther downrange it will travel. G-load is a measure of the accelerations acting on an object. The larger the difference in BN between two objects, the larger the initial G-load, or G-load spike, will be at release. Initial conditions are important in ballistic trajectories. Different initial velocities for the same BN will generate different downrange distances and G-load profiles.

AD ASTRA PER ASPERA • SEMPER EXPLORO

APOLLO 1 • CHALLENGER • COLUMBIA

COI03 1 • COI03 11

National Aeronautics and Space Administration

Lyndon B. Johnson Space Center
Houston, Texas 77058

www.nasa.gov