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MISSION SAFETY EVALUATION REPORT FOR STS-36

Postflight Edition: June 15, 1990

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Safety Division

Office of Safety and Mission Quality

National Aeronautics and Space Administration

Washington, DC 20546

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MISSION SAFETY EVALUATION

REPORT FOR STS-36

Postflight Edition: June 15, 1990

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EXECUTIVE SUMMARY

After 5 attempts, including 3 complete tanking operations, *Atlantis* was launched from Kennedy Space Center Launch Pad 39A at 2:50 a.m. Eastern Standard Time (EST), February 28, 1990. No technical problems occurred during the countdown. Rain showers moved into the area shortly after the opening of the launch window and remained in the area until shortly after 2 a.m. The ceiling over the Return-To-Launch Site runway began to dissipate at approximately 2:30 a.m., and the countdown clock, held at the T–9 minute mark, was restarted at 2:40 a.m. After a brief hold at T–5 minutes to double-check Return-To-Launch Site and TransAtlantic Abort Landing Site weather, *Atlantis* lifted off Pad 39A at 2:50 a.m. EST.

Atlantis touched down on Edwards Air Force Base lakebed runway 23 at approximately 10:09 a.m. Pacific Standard Time, March 4, 1990. Exterior review of *Atlantis* found it to be in excellent shape. Sixty-two debris hits on tiles were recorded, 14 greater than 1 inch. None will require tile replacement. Orbiter Auxiliary Power Unit #1 anomalies were the only problems of major concern.

FOREWORD

The Mission Safety Evaluation (MSE) is a National Aeronautics and Space Administration (NASA) Headquarters Safety Division, Code QS produced document that is prepared for use by the NASA Associate Administrator, Office of Safety and Mission Quality (OSMQ) and the Space Shuttle Program Manager prior to each Space Shuttle flight. The intent of the MSE is to document safety risk factors that represent a change, or potential change, to the risk baselined by the Program Requirements Control Board (PRCB) in the Space Shuttle Program Hazard Reports (HRs). Unresolved safety risk factors impacting STS-36 flight were also documented prior to the STS-36 Flight Readiness Review (FRR) (FRR Edition) and prior to the STS-36 Launch Minus Two Day Review (L–2 Edition). This final postflight edition evaluates performance against safety risk factors identified in previous MSE editions for this mission.

The MSE is published on a mission-by-mission basis for use in the FRR and is updated for the L–2 Review. For tracking and archival purposes, the MSE is issued in final report format after each Space Shuttle flight.

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SECTION 1

INTRODUCTION

1.1 Purpose

The Mission Safety Evaluation (MSE) provides the Associate Administrator, Office of Safety and Mission Quality (OSMQ) and the Space Shuttle Program Manager with the NASA Headquarters Safety Division position on changes, or potential changes, to the Program safety risk baseline approved in the formal Failure Modes and Effects Analysis/Critical Items List (FMEA/CIL) and Hazard Analysis process. While some changes to the baseline since the previous flight are included to highlight their significance in risk level change, the primary purpose is to ensure that changes which were too late to include in formal changes through the FMEA/CIL and Hazard Analysis process are documented along with the safety position, which includes the acceptance rationale.

1.2 Scope

This report addresses STS-36 safety risk factors that represent a change from previous flights, factors from previous flights that have impact on this flight, and factors that are unique to this flight.

Factors listed in the MSE are essentially limited to items that affect, or have the potential to affect, Space Shuttle safety risk factors and have been elevated to Level I for discussion or approval. These changes are derived from a variety of sources such as issues, concerns, problems, and anomalies. It is not the intent to attempt to scour lower level files for items dispositioned and closed at those levels and report them here; it is assumed that their significance is such that Level I discussion or approval is not appropriate for them. Items against which there is clearly no safety impact or potential concern will not be reported here, although items that were evaluated at some length and found not to be a concern will be reported as such. NASA Safety Reporting System (NSRS) issues are considered along with the other factors, but may not be specifically identified as such.

Data gathering is a continuous process. However, collating and focusing of MSE data for a specific mission begins prior to the mission Launch Site Flow Review (LSFR) and continues through the flight and return of the Orbiter to Kennedy Space Center (KSC). For archival purposes, the MSE will be updated subsequent to the mission to add items identified too late for inclusion in the prelaunch report and to document performance of the anomalous systems for possible future use in safety evaluations.

1.3 Organization

The MSE is presented in eight sections as follows:

Section 1 - Provides brief introductory remarks, including purpose, scope, and organization.

Section 2 - Provides a summary mission description, including launch data, crew size, mission duration, launch and landing sites, and other mission-related information.

Section 3 - Contains a list of safety risk factors/issues, considered resolved or not a safety concern prior to STS-36 launch, that were impacted or repeated by anomalies reported for the STS-36 flight.

Section 4 - Contains a list of safety risk factors that are considered resolved for STS-36.

Section 5 - Contains a list of Inflight Anomalies (IFAs) that developed during the STS-32 mission.

Section 6 - Contains a list of IFAs that developed during the STS-34 mission.

Section 7 - Contains a list of IFAs that developed during the STS-36 mission. Those STS-36 IFAs which are considered to represent safety risks will be addressed in the MSE for the next Space Shuttle flight.

Section 8 - Contains background and historical data on the issues, problems, concerns, and anomalies addressed in Sections 3 through 7. This section is not normally provided as part of the MSE, but is available upon request. It contains (in notebook format) presentation data, white papers, and other documentation. These data were used to support the resolution rationale or retention of open status for each item discussed in the MSE.

Appendix A - Provides a list of acronyms used in this report.

SECTION 2

STS-36 MISSION SUMMARY

2.1 Summary Description of STS-36 Mission

After 5 attempts, including 3 complete tanking operations, *Atlantis* was launched from Pad 39A at 2:50 a.m. Eastern Standard Time (EST), February 28, 1990. The first two opportunities were scrubbed due to poor weather conditions and a minor upper respiratory infection that kept Commander Creighton from being cleared for flight. Conditions improved, and a third attempt was made on February 25. Tanking and launch preparations went relatively smooth encountering only minor problems. Shortly after resuming the countdown at T–5 minutes, the Superintendent for Range Safety announced that Cyber B, one of two Range Safety computers, was down. The countdown clock was halted at the T–31 second mark in an effort to allow time for Cyber B to return to operational status. Several minutes into the hold, the Space Shuttle Main Engine Liquid Oxygen temperature rose above the Launch Commit Criteria (LCC) limit. Launch Director, Bob Sieck, determined that a recycle attempt could not be accomplished prior to the end of the launch window and scrubbed the launch.

The fourth attempt to launch *Atlantis* was made on February 26, 1990. Tanking and launch preparations again went smoothly; however, surface winds at Pad 39A were periodically gusting above the 24-knot LCC limit. The countdown clock was held at the T–9 minute mark in the hope that the high wind condition would subside. During this hold, a cloud deck beneath the 5,000-foot LCC limit moved in and was predicted to remain over Kennedy Space Center for the remainder of the launch window. At approximately 2:30 a.m., the Launch Director scrubbed the launch and ordered a 48-hour launch turnaround.

February 28, 1990, between 12 midnight and 4 a.m. was the fifth target window for the launch of *Atlantis*. No technical problems occurred during the countdown. Rain showers moved into the area shortly after the opening of the launch window and remained in the area until shortly after 2 a.m. The ceiling over the Return-To-Launch Site runway began to dissipate at approximately 2:30 a.m., and the countdown clock, held at the T–9 minute mark, was restarted at 2:40 a.m. After a brief hold at T–5 minutes to double check Return-To-Launch Site and Transatlantic Abort Landing Site weather, *Atlantis* lifted off Pad 39A at 2:50 a.m. EST.

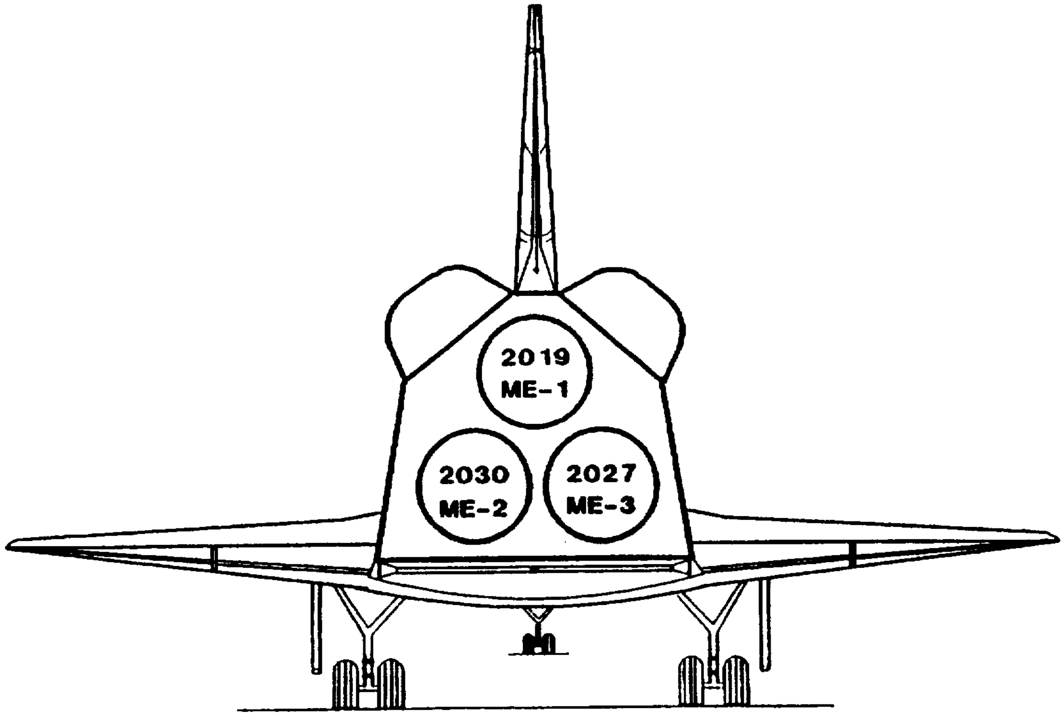
Ascent, through Solid Rocket Booster separation, Main Engine Cutoff, and External Tank separation went well. Nothing unusual was seen during the review of the ascent films. An anomaly with hydraulic system #1 pressure was the only item of concern. For the first time in 4 launches, there was no report of holddown post stud hangup or broaching. The A-286 "weak link" modification to the Debris Containment System held at least 96% of the debris on 7 of 8 holddown posts; approximately 64% was contained in the eighth.

Approximately 2 hours into the flight, the *Atlantis* crew was given the go ahead to continue their classified Department of Defense mission. The crew and vehicle were reported to be healthy throughout the remainder of the mission. Two Reaction Control System thrusters failed off and were deselected by Redundancy Management. One to 2 cups of water were found during on-orbit inspection of the humidity separators. Water escape from the separators was not as bad as that experienced on STS-32/OV-102; however, further investigation of the water escape problem is underway.

Atlantis landed at approximately 10:09 a.m. Pacific Standard Time (PST) on Edwards Air Force Base Runway 23 (lakebed), on March 4, 1990. Exterior review of *Atlantis* found it to be in excellent shape. Sixty-two debris hits on tiles were recorded, 14 greater than 1 inch. None will require tile replacement. Hydraulic fluid was found leaking from the engine #1 heat shield. In addition to recorded pressure anomalies during ascent, hydraulic system #1 pressure was erratic during descent. Closer inspection found that the entire aft compartment had been sprayed with hydraulic fluid. Investigation into the source of the leak found a 1/2-inch cut in the stainless steel braid of a flexible feedline. This feedline runs between the Auxiliary Power Unit #1 pump and filter package. Initial indications were that the cut was the result of a line burst versus an external cut. The flex hose was removed for further analysis. Hydraulic System #1 fluid reservoir was found dry, indicating that nearly 5 gallons of hydraulic fluid was lost. Removal of hydraulic fluid from the aft compartment has been completed. Failure analysis determined that this was not a generic flex hose problem; it was isolated to a single lot of flex hose teflon liner.

2.2 Flight/Vehicle Data

- Launch Date: February 28, 1990
- Launch Time: 2:50 a.m. EST
- Launch Site: KSC Pad 39A
- RTLS: Kennedy Space Center, Runway 33
- TAL Site: Classified
- Alternate TAL Site: Classified
- Landing Site: Edwards AFB, CA, Lakebed
- Landing Date: March 4, 1990
- Landing Time: 10:09 a.m. PST
- Crew Size: 5
- Inclination: Classified
- Altitude: Classified
- Orbiter: OV-104 (6) *Atlantis*
- SSMEs: (1) #2019, (2) #2030, (3) #2027
- ET: ET-33
- SRBs: BI-036
- SRMs: RSRM Flight Set #9
- MLP: MLP #1



ENGINE	#2019	#2030	#2027
POWERHEAD	#2020	#2107	#4004
MCC*	#2023	#2026	#2024
NOZZLE	#2024	#4013	#2027
CONTROLLER	F4	F17	F23
FASCOS*	#23	#26	#20
HPFTP*	#6008	#6003	#4010
LPFTP*	#2022R1	#2027	#4005
HPOTP*	#4406R3	#2324R5	#2225R2
LPOTP*	#2025R1	#2216	#4302

* Acronyms can be found in Appendix A.

2.3 Payload Data

The payload is classified.

NASA Headquarters Office of Safety and Mission Quality (OSMQ) did not participate in the Safety Reviews for the DoD payloads. NASA Headquarters Safety Division, Code QS, did participate in review of the Integrated Cargo Hazard Report (ICHR) by the System Safety Review Panel (SSRP).

SECTION 3

SAFETY RISK FACTORS/ISSUES IMPACTED BY STS-36 ANOMALIES

This section lists safety risk factors/issues, considered resolved (or not a safety concern) for STS-36 prior to launch (see Sections 4, 5, 6, and 7), that were repeated or related to anomalies that occurred during the STS-36 flight. The list indicates the section of this Mission Safety Evaluation (MSE) Report in which the item is addressed, the item designation (Element/Number) within that section, a description of the item, and brief comments concerning the anomalous condition that was reported.

ITEM		COMMENT
Section 4: Resolved Safety Risk Factors		
Orbiter 14	Reaction Control System (RCS) thruster weld issue.	Two RCS thrusters failed during STS-36. Both anomalies (IFA No. STS-36-04 and STS-36-12) were attributed to contamination of the oxidizer poppet valve that formed as a direct result of water intrusion prior to launch. The thruster chamber-to-injector weld concern did not a play role in either anomaly. Efforts are in progress to expedite the procurement of the universal throat plug that is designed to eliminate the potential for water intrusion into RCS thrusters.
SSME 5	Bluing found on nozzle of engine #2107, STS-33.	Postflight inspection of engine #2027, nozzle #2027, found similar bluing (IFA No. STS-36-I-01). Hardness tests found no annealing in the area of discoloration on this nozzle. Both instances of bluing have been attributed to steep reentry profiles.
Section 5: STS-34 Inflight Anomalies		
Orbiter 4 Orbiter 5	Humidity separator "B" (and "A") water bypass. (IFA No. STS-32-07A and STS-32-07B)	The crew reported finding approximately 2 cups of free water outside of humidity separator "A" (IFA No. STS-36-11). Postflight vendor analysis found contamination in the humidity separator pitot tube inlet, partially blocking it. Contamination had not been experienced in this location in any humidity separator to date.

ITEM		COMMENT
Section 6: STS-34 Inflight Anomalies.		
Orbiter 2	Auxiliary Power Unit (APU) #1 fault to high speed. (IFA No. STS-34-04)	There were no APU controller failures during STS-36. There was, however, an APU #1 flex hose leak (IFA No. STS-36-08) and a hydraulic system depressurization anomaly (IFA No. STS-36-17) on STS-36. There is no indication that these anomalies are related.

SECTION 4

RESOLVED STS-36 SAFETY RISK FACTORS

This section contains a list of the safety risk factors that were considered resolved for STS-36. These items have been reviewed by the NASA safety community. A description and information regarding problem resolution are provided for each safety risk factor. The safety position with respect to resolution was based on findings resulting from System Safety Review Panel (SSRP) and Program Requirements Control Board (PRCB) reviews (or other special panel findings, etc.). It represents the safety assessment arrived at in accordance with actions taken, efforts conducted, and tests/retests and inspections performed to resolve each specific problem.

SECTION 4 INDEX

RESOLVED STS-36 SAFETY RISK FACTORS

SECTION 4 INDEX - CONTINUED

RESOLVED STS-36 SAFETY RISK FACTORS

SECTION 4 INDEX - CONTINUED

RESOLVED STS-36 SAFETY RISK FACTORS

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Unitrode diode, Part Number (P/N) IN4148, Lot Date Code (LDC) 8339, failures. ***There were no anomalies on STS-36 attributed to Unitrode diode failures.***	During Range Safety Distributor (RSD) Serial Number (S/N) 118 recertification, the controller card failed thermal testing. This failure was isolated to a Unitrode diode, P/N IN4148, LDC 8339. There was a previous alert against P/N IN4148, LDC 8339. P/N IN4148 diodes are used in almost every RSD and Integrated Electronic Assembly (IEA), as well as other Solid Rocket Booster (SRB) components. LDC 8339 diodes were installed in the RSD on STS-36 on both legs of the redundant paths. Only 2 other failures of this diode were recorded in the Allied Signal program history; 1 against LDC 250 and this problem with LDC 339. The diode was taken to the laboratory for failure investigation. The Orbiter Project also reviewed the use of Unitrode diodes, P/N IN4148, LDC 8339. A list was compiled of Line Replacement Units (LRUs) that use P/N IN4148 diodes; however, the list did not indicate if Unitrode-manufactured diodes were used. Applications of IN4148 diodes in Orbiter LRUs that could result in a criticality 1 failure are considered highly remote. There have been no reports of Unitrode P/N IN4148 diode failures in the Orbiter Project history. There were 7 diode failures of undetermined manufacturers which could have been Unitrode. The Orbiter Project found no indication of generic IN4148 diode failure history, and retrofit was not recommend. Rationale for STS-36 flight with Unitrode diodes, P/N IN4148, LDC 8339, was: • For SRB applications, there is a very large usage with a small failure history (2 failures in 1,000 applications). • For Orbiter applications, there is no failure history, and failure modes for criticality 1 applications are considered remote. ***This risk factor was resolved for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1	Old bolts reinstalled instead of new improved bolts during Gaseous Hydrogen (GH_2) and Gaseous Oxygen (GO_2) disconnect modification kit installation. HR No. INTG-035 ***There were no disconnect anomalies reported on STS-36.***	During modification kit installation at Kennedy Space Center (KSC), old bolts were reinstalled on all vehicles instead of the new high-strength bolts from the modification kits. Investigation determined that the modification kit Time Compliance Technical Instruction (TCTI) provided the required instruction for the new bolt, but the Shuttle Processing Contractor (SPC) document did not contain an adequate description. Action was taken to reduce workmanship errors. New bolts were installed on OV-104. Other vehicles will have bolts replaced after flight. Old bolts have adequate safety margin and were approved for flight. ***This risk factor was resolved for STS-36.***
2	Forward Power Control Assembly (FPCA) #1 Remote Power Controller (RPC) #1 fails voltage drop test during vehicle checkout. HR No. ORBI-292 ***There were no further FPCA anomalies reported during STS-36.***	During vehicle checkout, the FPCA #1 RPC #1 failed the voltage drop test (dropped 1 volt). FPCA #1 RPC #1 was removed, replaced, and successfully retested. The failed RPC was shipped to the vendor for failure analysis. ***This risk factor was resolved for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3	Safety did not concur with approved exceptions to Airframe and Mid-Fuselage Wing Carry-Through inspections. HR No. ORBI-277 ORBI-278 ***There were no structural-related anomalies reported on STS-36. Structural inspections will be performed during the STS-38 flow.***	Periodic structural inspections of the Orbiter Airframe and Mid-Fuselage Wing Carry-Through are required. Due in part to schedule constraints, the requirement to inspect each Orbiter has not been met. Exceptions were approved for each Orbiter during the STS-33, STS-32, and STS-36 flows that allowed deviations from inspection requirements. In each case, Johnson Space Center (JSC) structural engineers and safety engineers voiced opposition to Program Requirements Control Board (PRCB) approval of the exceptions. Failure to perform structural and zonal inspections on the required schedule is, in itself, not a safety-of-flight issue for the near term. However, not performing these inspections impacts detection of potential structural and wiring anomalies, leading to possible problems in the long term that might be preventable. Safety's position was that there was a lack of technical data presented as rationale to support approval of the deviation. However, since performance of this inspection is not a safety-of-flight issue for the immediate mission, and because the inspection could only be adequately performed in the Orbiter Processing Facility (OPF), the decision was made not to interrupt the STS-36 flow schedule. KSC representatives who requested the exception were instructed to schedule the inspection during the next OV-104 flow. KSC was also instructed not to bring deviations and waivers, such as this, to the PRCB so late in the flow that either the schedule would be severely impacted or nothing else could be done but approve the waiver/deviation. Safety plans to track inspection requirements more closely in the future and to identify problems to the Space Shuttle Program as early as possible. ***This risk factor was acceptable for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Philadelphia Insulated Wire Company alert. HR No. INTG-178 ***There were no wiring anomalies reported on STS-36.***	Goddard Space Flight Center (GSFC) identified a problem with silver-coated, 20-gauge wire procured from the Philadelphia Insulated Wire Company. It was found that several of the 19 copper strands which comprised the 20-gauge, MIL-W-22759/11 wire, LDC 170-8-402, had either damaged silver coating or no coating at all. The silver plating is used to prevent corrosion. This wire type is used throughout the aerospace industry. Rockwell International (RI) identified that none of the suspect wire was in the Vehicle wiring. However, MIL-W-22759/11 wire is used by many vendors manufacturing Orbiter LRUs. These vendors could have used wire from this LDC. Application of this wire type is used within Teflon insulation with environmentally-protected connectors. Additionally, the LRUs are sealed with little or no likelihood of exposure to a corrosive atmosphere. There is no loss of effectiveness without the silver plating, only an accelerated aging process. ***This risk factor was acceptable for STS-36.***
5	Night landing crosswind limit revised without Safety concurrence. HR No. ORBI-179 ORBI-211A ***STS-36 landing was made during daylight.***	A proposal was made to increase the night landing crosswind limit from 10 to 12 knots. This modified Flight Rule #4-64F. A recent change increased the daytime landing crosswind limit to 15 knots. Safety did not concur with the change to the night landing limit, because supporting flight data was not available. The Orbiter has demonstrated some instability during landings in other than ideal conditions. Increasing the adverse landing condition limits increased the risk of damage to the Orbiter. The Level II PRCB approved this change and accepted the increased risk. ***This risk factor was acceptable for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6	OV-104 Right-Hand (RH) inboard elevon actuator. HR No. ORBI-003 *There were no further elevon actuator anomalies experienced during STS-36.*	The RH inboard elevon actuator channel #2 secondary delta P was found out-of-tolerance during individual servovalve null testing. Testing was used to ensure that the servovalve spool returns to null following single-servo operation. The possible causes for the null shift includes a sticking servovalve, partial blockage of orifices, torque motor electrical problem, or the delta P transducer. The actuator was replaced, and the failed unit was returned to the vendor for failure investigation. The replacement actuator was functionally verified. Servovalves are redundant within each actuator. *This risk factor was resolved for STS-36.*
7	OV-104 Engine Interface Unit (EIU) Power-On Reset (POR) anomaly. HR No. INTG-165 ORBI-066 *There were no further POR anomalies reported on STS-36.*	During EIU testing, position 3, S/N 20, the 60-Kilobit (Kbit) data path dropped out for 1.1 second (sec), and the subsequent read of the Bite Status Register (BSR) indicated POR. The anomaly occurred twice on December 11, 1989, and recurred once on December 18, 1989. The concern was that a simultaneous POR-A and POR-B in the last 30 sec prior to Main Engine Cutoff (MECO) would result in the General Purpose Computer (GPC) closing prevalves on running engines resulting in a catastrophic shutdown. (This is the worst-case failure scenario.) Failure of GPC command of the engine requires manual shutdown. Flight rules/crew procedures exist for this condition. System management alert and Main Engine (ME) status (amber) light on the panel (F-7) would alert the crew. Crew reaction is required to manually shut down MEs prior to prevalve closure. The subject Power-On Reset was characteristic of 12 previous occurrences (6 units, 10 vehicle flows since January 1983). POR was transient and self clearing, and troubleshooting did not reproduced the problem.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
7 (Continued)		The rationale for STS-36 flight was: • Each occurrence has been a single POR (POR-A or POR-B). • No single POR will fail more than one data command/data channel. • POR has not occurred in flight. • A single POR prior to T–0 will result in a launch hold or launch abort (post Space Shuttle Main Engine (SSME) start). • Simultaneous reset of both channels has never been experienced and requires 2 failures. • The last 30 sec prior to MECO is a very short time window in which the crew cannot react, and flight rules and procedures exist for this condition. • OV-104 EIU #3 (S/N 20) was removed and replaced (unit had most frequent POR occurrences) prior to STS-36. ***This risk factor was resolved for STS-36.***
8	Elevon tile improperly bonded. HR No. ORBI-249A ***There were no lost tiles, elevon or other, reported on STS-36.***	STS-34/OV-104 postflight tile inspection found 130 loose/improperly bonded tiles. Some of the tiles were loose enough to be removed by hand. OV-103 was also inspected, and similar conditions were found. OV-102 elevon tile wiggle tests were performed; no tile problems were found. The Lockheed Space Operations Company (LSOC) evaluation of the OV-104 elevon trailing edge bond anomalies revealed that all suspect tiles were bonded by the same technician with the same lot of Room-Temperature Vulcanizate (RTV). The LSOC investigation showed that

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8 (Continued)		the tiles were probably bonded at the low side of the pressure range (1 pound per square inch (psi)). Examination of the removed tile bond line also indicated that the Strain Isolator Pad (SIP) did not made full contact with RTV in the cavities. Other tiles were found to have "fuzzy" type bonds. It should be noted that, even though RTV penetration into the SIP in the "nominal" bonds was adequate for structural integrity, the bond line characteristics still indicated marginal RTV penetration into the SIP. All OV-104 improperly bonded tiles were replaced, and tile wiggle tests were performed. No additional tile problems were found. ***This risk factor was resolved for STS-36.***
9	Large voids found across Fuel Cell (FC) flowmeter braze joints. HR No. ORBI-286 ***No fuel cell flowmeter anomalies were reported on STS-36.***	During rebuild of an improved spare fuel cell at IFC (vendor), x-ray of the FC Oxygen (O_2) flowmeter braze joint showed voids exceeding specifications. Each FC contains one Hydrogen (H_2) and one O_2 flowmeter. The O_2 system operating pressure is 1050 pounds per square inch absolute (psia) maximum; H_2 system operating pressure is 335 psia maximum. Leak rate (4×10^{-6} cubic centimeters per second (ccs)) met RI specification requirements but failed tighter supplier (IFC) requirements (no indicated leakage allowed). X-rays showed braze void areas exceeding 15% allowable, with connecting voids providing a leak path. MIL-B-7783 specification requires braze coverage of 85% or greater and no voids greater than 15% of the joint length. Review with the flowmeter manufacturer (Rosemont) determined that the original x-rays of this and other flowmeter braze joints failed to meet the 15% maximum void area requirement.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
9 (Continued)		OV-104 fuel flowmeter braze joint x-rays were reviewed. Of the 33 braze joints involved: 13 met standards, 17 did not meet standards, and 3 x-rays were not located. Of the 17 that did not meet standards, the worst case contained 15 to 20% braze coverage. Braze joints for which x-rays were missing are in OV-104 FC #2 H_2 flowmeter. OV-104 braze joints were not accessible for inspection at the pad. Structural analysis of the braze areas indicated that all braze joints were stronger than the tube itself. (Worst-case braze coverage results in a braze joint Factor of Safety (FOS) of 1.27.) All braze joints passed a minimum 2 x maximum operating pressure proof test: • O_2 proof 2100 psia (factor of 2) • H_2 proof 2100 psia (factor of 6) All OV-104 flowmeters passed numerous leak checks, including pressure decay checks conducted during each OV-104 turnaround. All OV-104 flowmeters had flown 3 flights without problems and were otherwise undisturbed. Based upon the history of these FC flowmeters, including proof, leak, and flight experience, and the fact that only 1 extremely small O_2 leak developed during their entire history, OV-104 flowmeters are acceptable for continued service. A waiver for these braze joints was approved for all vehicles. ***This risk factor was resolved for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
10	OV-104 Auxiliary Power Unit (APU) #1 turbine wheel limited-life issue. HR No. ORBI-031 ***APU #1 was started 3 times during the STS-36 mission evolution, with no turbine wheel anomalies reported.***	A determination was made that APU #1 S/N 307 would violate turbine wheel limited-life requirements during this flight. The turbine wheel was originally installed on APU S/N 312 and accumulated 6 starts and 11 minutes (min) of high-speed time. It was later installed on APU S/N 307. Records indicated that this turbine wheel was installed on S/N 307 as new. The accumulated time on S/N 312 was inadvertently dropped. Since the installation on S/N 307, the turbine wheel in question accumulated 13 starts and 37 min of high-speed time. This brought the total accumulated life to 19 starts and 48 min of high-speed time, well above the current limited-life inspection requirements. Under nominal conditions, this turbine wheel would experience 2 additional starts during STS-36, with no high-speed run time. APU turbine wheels are life limited due to blade cracks induced by High-Cycle Fatigue (HCF). Operational Maintenance Requirements and Specifications Document (OMRSD) inspection requirements were established based on the number of APU starts and high-speed run time. Since the establishment of these requirements, additional high-speed testing was performed. This data was evaluated by RI, Sundstrand, and Southwest Research Institute (a JSC contractor). The consensus, based on results of recent analyses and tests, was that the original life limit established for new turbine wheels was conservative. Accordingly, the Level II PRCB approved an increase in the APU #1 S/N 307 turbine wheel limit to 24 starts or 69 min of high-speed operation for this flight only. The confidence level for 0.999 reliability at 69 min of high-speed operation for a new wheel is 95%. There is no planned high-speed APU operation during a nominal mission. If a contingency requires high-speed operation in excess of the life limit, the reliability of this wheel is 0.995 with 95% confidence. ***This risk factor was acceptable for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11	OV-104 has 4 Multiplexer-Demultiplexers (MDMs) that may contain Erie capacitors which are prone to failure. HR No. ORBI-038 ***There were no MDM failures or anomalies reported on STS-36.***	In 1981, Erie capacitors were found to be prone to failure due to a low-resistance short. The Orbiter Program Office (OPO) directed that the capacitors be purged from increment III MDM builds and directed that replacement of these capacitors on other MDMs be made by attrition or when MDMs were returned to the vendor. Presentations made at the STS-28 Flight Readiness Review (FRR) Action Item Closeout Meeting indicated that OV-104 had 4 MDMs installed with Erie capacitors. The configuration of OV-104 MDMs is as follows: S/N 27 PL2 Payload MDM #2 (16 Erie Capacitors) S/N 67 OF3 Operational Instrumentation - Forward MDM #3 (272 Erie Capacitors) S/N 71 OF4 Operational Instrumentation - Forward MDM #4 (160 Erie Capacitors) S/N 74 OA3 Operational Instrumentation - Aft MDM #3 (256 Erie Capacitors) Failure of PL2 would result in a single path for communications and control of the payload through PL1. This MDM is one of a redundant set that would be required to operate only through payload separation from the Orbiter. There were no signs of incipient failure. OF1, OF2, and OF3 are criticality 1/1 units. Failure of OF3, coupled with a FC heater relay turn-on, cannot be detected. Additionally, failure of OF3 in combination with a subsequent undetected failure in a FC stack due to H_2/O_2 crossover could result in possible loss of crew and vehicle. Failure of OF3, therefore, results in reentry at the next Primary Landing Site (PLS), per Flight Rule 9-23.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)		Three of the 272 Erie capacitors in OF3 S/N 67 are in critical locations in the MDM. They are on Input/Output (I/O) module card #1, channels 27, 28, and 29. These channels carry critical Cell Performance Monitor (CPM) FC reactant (H_2 and O_2) crossover data. Automated ground monitoring of the Orbiter would not detect failure of OF3 during operation. Nominal CPM substack delta voltage output is 0 to 150 millivolts (mV). Exceeding the 150-mV redline would trigger an alert at the Electrical Power System (EPS) console at the Mission Control Center (MCC). Failure of the capacitor would skew the output downward by a factor of approximately 2 to 20. Therefore, the 150-mV redline could be exceeded by a large amount without instrumentation triggering an alarm. The CPM outputs a 50-mV reference voltage every 7.5 min for a period of 2.3 sec. This reference voltage may be manually read by the EPS console operator, but is not automatically annunciated. When operating correctly, a 50-mV reference voltage would be displayed on the console. Failure to read 50 mV could be an indication that the capacitor has failed. Rationale for flight was based on the low probability of occurrence and the fact that there had been no Erie capacitor flight failures to date. JSC Safety, Reliability, and Quality Assurance (SR&QA) recommended that the MCC EPS console operator be alerted to this condition and directed to pay particular attention to the CPM self-test reference voltage for MDM OF3. Additionally, it was recommended to the OPO that the OF3 MDM S/N 67 be removed and replaced prior to the next OV-104 flight, STS-38. ***This risk factor was acceptable for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
12	Fabrication weld defects in Orbiter aft structure. HR No. ORBI-278 ***There were no anomalies attributed to structural weld defects on STS-36.***	An Orbiter weld quality investigation team was established to investigate fabrication weld defects in the Orbiter aft structure. A concern developed as a spinoff of the SSME High-Pressure Fuel (HPF) duct failure investigation. (See also Section 4, SSME 5.) The original weld concerns were thought to be limited to the RI Palmdale B1-B weld facility. However, the investigation expanded to include the El Segundo facility based on Rocketdyne findings of similar problems in the SSME hardware. The plan of action was to clear the STS-36 hardware through a review of the OV-104 hardware produced by the B1-B at the LA/Palmdale facilities. The review team examined x-rays of the suspect hardware. The Orbiter aft structure was identified as the only area with suspect welds. The weld manufacturing process and weld test coupons were reviewed; no problems were identified. X-ray review found some minor defects; however, all were within allowable specification. A visual inspection of all accessible aft structure components was made prior to flight; no problems were discovered. ***This risk factor was resolved for OV-104/STS-36.***
13	OV-104/STS-36 right Orbital Maneuvering System (OMS) engine gimbal profile anomaly. ***Data indicated that the right OMS engine gimbaled correctly when commanded during the STS-36 mission.***	During S0008 testing of the right OMS in the Vehicle Assembly Building (VAB), all actuator drive rates (pitch and yaw in both active and standby) were too slow. Interference with a new heat shield seal was suspected. The new heat shield seal was installed after proper Thrust Vector Control (TVC) actuator operation was verified in the OPF. The heat shield was removed, and the S0008 OMS TVC gimbal tests were performed. Gimbal profile rates were recorded to be within specification. Inspection of the heat shield and seal found no abnormalities. Both the heat shield and the seal were reinstalled.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
13 (Continued)		Plans called for repeated runs of the gimbal profile tests at the Pad. Frictional resistance caused by seal interference diminished with repeated gimbaling of the TVC, and the gimbal profile was within specification. Initial gimbaling resulted in "scraping" and "chattering" noises. The scraping was attributed to heat shield fastener torquing. The torque requirement is 16 inch-pounds (in-lb); however, this was found to be too loose to bind the seal in place. The fastener torque was increased to further secure the seal and to eliminate the scraping. Retest after torquing resulted in a good gimbal profile and no unusual noise. Investigation into the gimbal profile test issue uncovered the fact that 1 of the 2 heat shield seals on the right OMS engine was not properly heat-cleaned or baked-out. Visual inspection of the seals found that 1 seal was white in color, instead of a gray color. The seals comprise a ceramic fiber sleeve (Fiberfrax) over an Inconel spring that surrounds a ceramic fiber core (rope). After seal assembly at Carborundum (vendor), the seal was supposed to be heat-cleaned to remove an organic material used to facilitate the weaving process. Heat-cleaning reduces outgassing of volatiles in the organic material. The white fiber sleeve turns gray during this process. Through quality control "escapes" at both Carborundum and McDonnell Douglas Corporation, 6 heat shield seals were delivered to KSC without undergoing the heat-cleaning process. One of these 6 was on the right OMS engine heat shield on OV-104/STS-36. Based on discovery of the white seal on the right OMS engine, an investigation was undertaken to determine if these non-heat cleaned seals were acceptable for flight. It was determined that the white seals were acceptable for flight. The white seals will outgas the volatiles when first exposed to heat. There are no material incompatibilities, and outgassing is not a problem to the crew or OMS Pod due to seal location. The fabric in the seal sleeve would not sustain a flame if the volatiles ignite. In addition, sealing capability is not affected by lack of heat cleaning. The

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
13 (Continued)		Material Review Board (MRB) reviewed the use of the white seal on OV-104/STS-36 and approved the seal for unrestricted use. Note that exposure of the seal to engine heat basically heat cleans it. The MRB determined that the white seal met form, fit, and function of a heat-cleaned seal. RI initiated an action to send the remaining 5 white seals from KSC to McDonnell Douglas/ Carborundum for required heat-cleaning. ***This risk factor was resolved for STS-36.***
14	Reaction Control System (RCS) thruster weld issue. HR No. ORBI-119 ***There were 2 RCS thruster anomalies reported on STS-36 (IFA No. STS-36-04 and STS-36-12). These failures were attributed to contamination resulting from water intrusion at the Pad.***	During instability testing at the White Sands Test Facility (WSTF), RCS thruster S/N 111 failed. Thruster chamber flange temperature rose to approximately 1600°F, causing the wire wrap to short to the flange and resulting in thruster shutdown. Subsequent inspection revealed cracks along the thruster chamber-to-injector weld location. Three cracks were found, 1-1/2" to 2" in length, 90° apart. No fatigue striations were observed. S/N 111 had flown 7 flights prior to its return to Marquardt for the wire wrap modification. The failure occurred during the required post-modification instability testing. Upon sectioning at Marquardt, a lack of fusion was found in the weld; only 40% to 60% weld penetration was found around the circumference. This lack of penetration was due to a misalignment of the weld beam. Review of the 6 original x-rays found no indication of lack of fusion. The cracks found during failure analysis were probably caused by the high temperature experienced during instability testing. At the STS-36 FRR, the OPO stated that the thrusters had a 33-mission life. This was based on analysis previously performed at Marquardt using FLAGRO, a NASA flaw growth analysis tool which models hoop stress on radially symmetric models.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
14 (Continued)		A joint NASA-RI investigation at Marquardt revealed that FLAGRO was improperly used in making this life-limit determination. Initial FLAGRO conditions entered by Marquardt included specifying a weld penetration of only 10%; this initial condition was based on use of a proof test fixture. The NASA-RI investigation discovered that the stresses induced at a weld penetration depth of 15% would be greater than yield stress. For this reason, the previous analysis using 10% penetration was invalidated. Review of the proof test fixture demonstrated that axial stresses induced are reacted in the fixture, rather than the thruster nozzle. This reassessment concluded that 10% weld fusion could not be assured using the proof test data, thus invalidating the use of proof test data as a pass/fail criteria. RI use of FLAGRO, along with other stress analyses, found that a 30% weld penetration would result in a thruster life of 68 missions. During the recent investigation, the loading environment was defined. Through analysis, the maximum bending stress experienced at the flange-to-chamber joint was determined to be 26,480 psi. This induced stress is the result of radial expansion of the joint during thruster firing. Pressure, coupled with a 300°F temperature differential between the flange and the chamber, provides the radial expansion. An Acceptance Test Procedure (ATP) comprises 5 steady state firings and 335 pulse firings of the thruster. Applied thermal loads and pressures during ascent are not as severe as those experienced by a thruster during ATP. During ATP, firing tests produce an equivalent of 19 to 20 "applications" of flange-to-chamber maximum bending stresses (26,480 psi). An "application" is one full set of flight loads from a reference mission, described in the next paragraph. A one-mission load environment is divided into 3 conditions: ascent; long-duration thruster firing; and short, pulse thruster firing. Three-sigma ascent loads provide

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)		approximately 357 cycles of bending stress at ± 4,760 psi maximum, through random vibration. This level of stress occurs in the long thruster of the forward RCS. Shorter thrusters receive less stress and will have a longer fatigue life. A long-duration thruster firing results in a maximum bending stress of +26,480 psi. There are nominally 2 long-duration thruster firings during a mission, again by the forward RCS long thrusters. Short, pulse thruster firings, approximately 200 per mission, result in bending stresses of +4,901 psi to −3,189 psi. The investigation also determined the minimum weld penetration which could result, based on a review of the welding process. The welding process comprises 5 steps: • Preweld test - Determines the appropriate power setting and verifies the weld penetration. • Tack weld - The first pass around the joint where 4 tack welds are made. These welds are normally 0.5" length by 0.046" width by 0.084" in diameter. Weld width tolerances are ±0.003". • Seal pass - After removal of the chamber tack fixture, a seal pass is made over the joint circumference and the previous tack welds. • Penetration pass - Immediately after the seal weld, the power adjustment is turned up and a penetration weld is made over the entire circumference. The weld width increases to 0.084" and the depth to 0.150", 100% of the joint thickness. • Cosmetic pass - Covers the weld area with a less-focused welding beam to smooth the surface.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)		It was determined that tack welds are probably centered on the joint within 0.010". Tacks assure beam centering within 0.023" (half the nominal weld width), because the chamber would fall off when the tack-weld fixture is removed if the tack weld beam was offset more. Misalignments during the seal pass would be detectable. If the beam missed by greater than 0.023", the operator would be able to see the unconsumed joint because the tack welds would be unconsumed and highly visible. No mechanical beam adjustments are made between the penetration pass and the seal pass so that the penetration weld is coincident with the seal pass. The power settings are increased between the seal pass and the penetration pass to double the weld width and complete the weld penetration. It is very unlikely that the operator would overlook an unconsumed joint following the seal pass. A geometric analysis of a worst reasonable case, misaligned weld yields less than 0.023" offset. Therefore, a 30% weld penetration was determined to be the worst case. Fracture analysis was performed to determine the failure mode of the flange-to-chamber joint. Using a flat plate model, 0.150" thick by 12" wide, and FLAGRO, the minimum weld penetration of 30% was used to define the initial flaw. ATP and mission axial and bending stresses experienced were added. The result demonstrated that the welded joint would not fail, or burst, but would eventually leak through flaws or cracks that would propagate from the initial flaw. The investigation team also reassessed the use of the wire wrap modification. Experience has demonstrated that the wire wrap assures the capability to safely shut down a failed thruster. A review of the wire wrap certification found the following results. Eight demonstration tests were performed, none of which included instability. All tests were performed with a hole drilled in the thruster chamber wall critical area to simulate burnthrough. Test holes ranged from 0.125" to 0.500" in diameter. Thrusters in all tests were shut down by the wire wrap, ranging from

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)		80 to 350 milliseconds (msec) during steady-state firings, and 1 to 4 pulses of 80 msec each during pulsed firings. In all tests, no external damage to the thruster was found, and the thrusters did not burst. Two additional thrusters tests, involving forced instability, were performed on S/N 140 and S/N 012 during certification . Both tests resulted in chamber cracks, and both were shut down by the wire wrap. Chamber cracks initiated approximately 28 sec into the firing of S/N 140. This thruster was shut down at 34.7 sec. The resulting fatigue crack was determined to be caused by the induced instability and was not located at a weld joint. During the test with S/N 012, a chamber crack initiated approximately 18 sec into firing and was shut down by the wire wrap at 25.5 sec. Failure analysis of this crack revealed that thermocouple wires inadvertently left next to the chamber wall melted, causing a liquid metal embrittlement phenomenon. In both cases, there was no resulting external damage to the thruster. Two failures of thrusters have occurred during post-modification instability testing at WSTF; S/N 111, which led to this investigation, and S/N 128. Chamber burnthrough of S/N 128 resulted after 46.5 sec of unstable firing. Here too, the wire wrap shut down the thruster. No external damage resulted in either case. Rationale for STS-36 flight was: • The worst-case thruster for STS-36 was F2D, a long thruster in the forward RCS, which had flown 10 missions. Analysis showed a 68-mission projected life, assuming a worst reasonable case 30% weld penetration. A short thruster in the aft RCS had flown 12 missions, but in a more benign stress environment.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
14 (Continued)		• It was established that the only reasonable failure consequence of a 30% or greater weld penetration weld is a leak, not a total failure or burst. The wire wrap would shut down a thruster leaking through one of these welds in sufficient time to prevent further damage to the thruster or adjacent thrusters. The wire wrap modification was installed on all OV-104/STS-36 RCS thrusters. • RCS thrusters are redundant. ***This risk factor was resolved for STS-36.***
15	Small spark observed during prelaunch flight crew suit operations.	During flight crew suit operations, a small white spark was observed by the JSC suit technician while attaching an additional mirror for the pilot to panel F7. The mirror tether was being attached to the switch wicket for S1, the Main Propulsion System Helium Meter Selector Switch. No shock was felt by the technician when the spark was observed. The technician was wearing standard anti-static apparel and used approved procedures for spark mitigation and mirror tether installation. Humidity in the crew cabin was between 20.4 and 21.8%; lower than the normal 35-to-50%. Sparks have been experienced during other suit technician activities. Sparks were previously observed during switch guard installation or removal activities. These occurrences were attributed to a "flint" type action. This type of spark was attributed to material incompatibility between the switch wicket and the switch guard. All previous occurrences were accompanied with an odor; no odor was witnessed during this occurrence on STS-36/OV-104.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
15 (Continued)		Attempts to recreate the spark immediately after this occurrence were unsuccessful. Bonding checks of the panel verified that no damage would occur to electrical components from static electricity. The most probable cause of this spark was a static discharge between the technician/mirror tether clip and the Helium Meter Selector Switch wicket on panel F7. ***This risk factor was resolved for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	High-Pressure Oxidizer Turbopump (HPOTP) S/N 2225R2 on engine #2027 was the first to be installed on a flight engine without teardown and refurbishment. HR No. ME-C1 (All Phases) ***There were no HPOTP anomalies reported on STS-36.***	HPOTPs that were previously flown are nominally torn down and refurbished prior to being flown again. The primary reason for postflight teardown is to inspect the bearing cage for cracks. At the STS-36 Orbiter Rollout Review, Rocketdyne and Marshall Space Flight Center (MSFC) reported that HPOTP S/N 2225R2, installed on engine #2027, was the first to be installed on a flight engine without being refurbished. S/N 2225R2 flew on STS-34 and was returned to Stennis Space Center (SSC) after the mission. At SSC, S/N 2225R2 was instrumented for detection of bearing cage harmonics and green run for 80 sec. No vibration was witnessed or recorded during this 80-sec run. MSFC and Rocketdyne have found through experience that not witnessing/recording vibration with harmonics instrumentation during a HPOTP test run is a strong indication that bearing cage cracks do not exist. Because this was the first flight test of an unrefurbished HPOTP, it was considered an increase in overall risk. Results of postflight inspection of HPOTP S/N 2225R2 will determine the long-term HPOTP turnaround procedure. ***This risk factor was acceptable for STS-36.***
2	Nozzle protrusions on engine #2027 were greater than the established limit. HR No. ME-B7 (All Phases) ***There were no SSME anomalies attributed to nozzle protrusions on engine #2027 during STS-36.***	Nozzle #2027, installed on engine #2027, was measured and found to have nozzle tube protrusions of 0.082". These tube protrusions were due to having the tube stack shifted aft and the nozzle lip were machined. The tube protrusion limit established by the SSME Project is 0.078" for 2770 sec of operation with the Flow Restriction Inhibiter (FRI) modification. This limit was determined by measuring all engine nozzle protrusions and by extended testing of engine #0209 that has the greatest tube protrusions measured to date. When previously installed on engines #2031 and #0213, nozzle #2027 tube protrusions were measured at 0.065" and 0.077", respectively. The MSFC and Rocketdyne position was that this condition

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
2 (Continued)		was acceptable for 1 flight only. This was based on a demonstrated safety factor of 2 for total, worst-case flight operation time determined through test engine operation experience of 5447 sec (5447/2 is approximately 2770) with the FRI modification installed. The FRI modification was installed on engine #2027. Additionally, the Main Combustion Chamber (MCC) on engine #2027 did not have a machined lip, thus reducing the potential for hot gas recirculation. Subsequent testing on engine #0213, with 0.102" effective tube protrusions, showed a safety factor greater than 2 for a 1 flight abort duration. ***This risk factor was acceptable for STS-36.***
3	High-Pressure Fuel Turbopump (HPFTP) #6008, engine #2019, exhibited turbine-end liftoff seal leakage. HR No. ME-A1A ME-A1D ***There were no HPFTP anomalies on STS-36.***	HPFTP #6008 had a leak in the turbine-end secondary seal. The leak, measured at 54 standard cubic inches per minute (scim) after the green run of the pump at SSC, was above the specified 50-scim limit at 25 ±5 pounds per square inch gage (psig). Initial disposition at SSC was to repeat the leak check following pump torque and travel measurements. The liftoff leak check revealed a 38.5-scim leak. The pump was then sent to KSC for installation on engine #2019 for STS-36. At KSC, technicians conducted leak checks and found HPFTP #6008 to have a 51.27-scim leak. A decision was made to incrementally increase the leak check pressure and measure the leak. The following leak data was recorded: 38.7-scim leakage at 20 psig 51.3-scim leakage at 25 psig 64.4-scim leakage at 30 psig

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
3 (Continued)		Rationale for flight of HPFTP #6008 on engine #2019 was: • Other liftoff seal leak checks and pump performance were nominal. – The nose and pump-end secondary seal leak checks were passed. – Acceptance test drain line parameters were nominal and well within HPFTP green run specifications. – The turbine-end secondary seal leakage was most likely due to a minor imperfection in the surface. • Secondary seal leakage was determined to be a criticality 3 condition based on analysis. – Worst-case effect is a partial closure of the liftoff seal. This requires complete failure of the secondary seal. Resulting reduction in coolant flow is still sufficient to cool turbine-end bearings. – Flight experience base established with 5 HPFTPs with greater than specified turbine-end secondary seal leakage; as high as 138 psig. ***This risk factor was acceptable for STS-36.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
4	SSME HPF (multiple HPFTPs) duct weld issue. HR No. ME-D3 (All Phases) ***There were no SSME anomalies attributed to weld defects on STS-36.***	HPF ducts used on the SSMEs are manufactured by RI and proof-pressure tested upon delivery to Rocketdyne, Canoga Park. One of these ducts recently failed the proof-pressure test by rupturing at a weld. Failure investigation found O_2 contamination in the failed weld. A complete review was performed of the fabrication pedigree of the HPF ducts on OV-104. The HPF duct on engine #2030, STS-36 engine #2, had a documented case of incorrect weld filler in 1987. That section of the duct was removed and replaced at that time. An x-ray review of the welds on the STS-36/OV-104 engines found defects on the inlet on engine #2027, HPFTP #4008. Discussion of the weld defect found during the x-ray review led to belief that the defect had been repaired. This could not be readily determined because neither repair records nor post-repair x-rays of HPFTP #4008 could be located. Because of this issue, the OPO reviewed the history and acceptability of structural welds. See Section 4, Orbiter 12 for further details. Several options for resolving this problem were considered. First, because it was believed that the weld defect on HPFTP #4008 had been repaired, the decision was made to remove HPFTP #4008 and x-ray the weld defect location at the Pad. Two x-rays of the weld area were inconclusive, because the weld could not be readily seen. A third x-ray found that the weld defect was not repaired as thought. This disqualified HPFTP #4008 from further consideration. Consequently, program management directed that this pump be removed and replaced prior to STS-36 flight. A second option was to pull HPFTP #4010 from engine #2022, used on STS-32 (OV-102). A review of HPFTP #4010 x-rays revealed no weld defects. However, a potential, undocumented weld repair was discovered. Review of KSC x-rays, El Segundo x-rays, and a borescope inspection of HPFTP #4010 was performed.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
4 (Continued)		X-ray reviews identified minor weld anomalies, including variations in weld widths. The borescope inspection uncovered an unconsumed weld wire at the intersection of welds 6, 7, and 9 in the flow stream. Concerns were raised relative to structural integrity of the weld, flow restrictions, and the possibility that the weld wire would dislodge and become debris. A structural concern was raised because the borescope technician reported that the weld was yellow in color, indicative of too much O_2. Anything darker than a light wheat color is considered to have higher than desired O_2 concentrations. A review of the borescope inspection by Rocketdyne engineers found that the color was in the acceptable range. The concern was determined to be small, because the unconsumed weld wire was seen as a small spur. The risk of this wire becoming debris was considered unlikely because this HPFTP was flown on STS-32, and no detrimental effects were noted. With alleviation of these concerns, HPFTP #4010 was cleared for installation on OV-104/STS-36 to replace HPFTP #4008. HPFTP #4010 installation was successfully performed. Engine Flight Readiness Tests were successfully completed on February 19, 1990. ***This risk factor was resolved for STS-36.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
5	Bluing found on nozzle of STS-33 engine #2107. HR No. ME-B7 (All Phases) ***Postflight inspection of engine #2027, nozzle #2027, found similar bluing (IFA No. STS-36-I-01). Hardness tests found no annealing in the area of discoloration on this nozzle. Both instances of bluing have been attributed to steep reentry profiles.***	Postflight inspection of STS-33 engine #2107 found discoloration of the nozzle at the aft manifold closure. The discoloration, or bluing, was near the lower centerline on the aft facing surface. Bluing of the aft manifold closure had not been seen previously. This nozzle was new and flew for the first time on STS-33. The nozzle on engine #2031 was also new; however, no similar discoloration was found. Rocketdyne analysis found, through hardness tests, that there was no annealing in the area of the discoloration. They approved the nozzle for further flight use. The cause of the bluing could not be readily determined. Causes resulting from contamination, ascent or decent heating, improper material properties, and flight profile were all ruled out. This anomaly was closed as unexplained. Nozzles on STS-36 engines were inspected for similar bluing; however, there was no basis to predict that discoloration in this location would recur. ***This risk factor was resolved for STS-36.***
6	Heat-damaged nozzle steerhorn insulators on engines #2022 and #2028 indicated potential for cold wall leaks. HR No. ME-A2 (All Phases) ***There was no heat damage witnessed in the area of the STS-36 nozzle steerhorns during postflight inspection.***	Postflight inspection of STS-32 engines found external signs of heat-damaged insulators on the nozzle steerhorns of engines #2022 and #2028. Insulators from these engines were compared to insulators with known degraded or melted batting. This comparison verified that the engine #2028 insulator was damaged. The concern with signs of heat damage was based on the potential for coolant leaks and fire. Inspection of engines #2022 and #2028 steerhorns found no signs of melting or heating in the internal diameter of the insulators. Dye penetration inspection found no anomalies. No leaks were detected through leak and sniff checks. Etch and hardness tests verified proper steerhorn material properties. Adjacent coolant tube leak checks showed no leaks in the area of the damaged insulators. However,

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
6 (Continued)		2 tube leaks were noted in the area away from the anomaly. One was documented and repaired prior to flight; the other had not been previously found. Both are considered within the experience base found during postflight engine inspections. Inspection was performed of the steerhorn area of the engines on OV-104/STS-36. Visual inspection of steerhorn insulators found no anomaly; no signs of heating on the internal diameter of the steerhorn and no abnormal heating were noted. Two steerhorns on engine #2030 and all of the steerhorns on engine #2019 were not insulated; they were not exposed to reentry heating. The engine #2027 nozzle was installed after engine acceptance testing; therefore, it had not been exposed to engine hot fire or flight. Rationale for STS-36 flight was: • Insulator damage on engine #2028 was not the result of coolant leakage at the steerhorn. • OV-104/STS-36 engines were cleared through inspection, lack of insulator requirements, and new insulation. ***This risk factor was resolved for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
7	Eccentric ring fold-over found on STS-32 engine #2028. HR No. ME-C1 (All Phases) ***Postflight inspection of STS-36 engines found no similar fold-over.***	Postflight inspection of STS-32 engine #2028 revealed that the eccentric ring was folded over during installation of the HPOTP. The concern was that crimped metal could cause a decrease in capability of pressure-assisted seals. This could ultimately lead to a hot-gas path from the oxidizer preburner to the hot-gas manifold. This condition results in loss of turbine power, opening of the Oxidizer Preburner Oxidizer Valve (OPOV), and exceedance of the HPOTP turbine discharge temperature redline. There was also concern relative to hydrogen strut coolant loss (warming), resulting in leaks to the hot-gas manifold and possible turbine failure. There were no on-vehicle inspection techniques available to determine if this condition existed on the STS-36 engines. Past incidents have not resulted in anomalous operation; seals expanded to take up the additional distance left by the fold-over. Based on flight and test experience, little or no leakage would be expected. ***This risk factor was resolved for STS-36.***
8	Broken Chamber Coolant Valve (CCV) thrust bearing retainers found on engines #2206 and #0209. HR No. ME-B2M ME-B2S ME-B6M ***There were no CCV-related anomalies recorded on STS-36 engines.***	Three CCVs were returned to SSC for various anomalous conditions. All were found to have broken thrust bearing retainers. The first was returned for exceeding the torque requirements during installation on engine #2206. A review of SSC planning indicated that the gate assembly had slipped due to improper slider replacement. Improper stack-up caused the thrust bearing retainer to be crushed, which prevented the CCV from meeting torque requirements. The second CCV, also from engine #2206, was received at SSC with a different number of shims in place than when it left the CCV room. This indicated that the slider was replaced and the coupling assembly was reshimmed outside the

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
8 (Continued)		valve room. During disassembly of the CCV, the thrust bearing was found to be broken. The third retainer problem was found when the CCV was not installed on engine #0209 because it was missing the entire coupling assembly. Investigation found that the thrust bearing retainer was also broken. A record search determined that the slider was indeed replaced and that the coupling assembly was reshimmed outside the CCV room. Because the coupling assembly was missing, a count of the number of shims was impossible. The following failure scenario was also postulated. When the CCV is removed from the CCV duct, the internal parts stack-up is maintained by the lower coupling. The lower coupling is loosened during slider replacement while the CCV is removed from the CCV duct. This allows the gate assembly to slip, permitting the thrust bearing retainer to move out of position in a radial direction. When the CCV is reinstalled into the CCV duct, the out-of-position retainer is crushed. As a result of the first incident on engine #2206, SSC planning was changed to prevent slider replacement when the CCV is removed from the CCV duct. A drawing change was approved to add an assembly caution note to the CCV assembly process. This will drive the same planning changes at other facilities where slider replacement could be performed. A review of KSC and SSC planning was performed to determine if adjustment of lower couplings was necessary during installation of CCVs on OV-104/STS-36 engines; no adjustments were required or performed. ***This risk factor was resolved for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
9	Preburner purge pressure redline exceedance during engine #0213 testing at SSC. HR No. ME-B2 (All Phases)	During engine #0213 ground testing at SSC, the engine was automatically shut down 0.36 sec into the test. It was determined that the shutdown was due to an exceedance of the 50-psia preburner purge pressure redline. Abnormal fuel preburner purge pressure was found to be caused by an internal helium leak from the pressure actuated valve. Post-test leak checks determined that the internal helium leak was 16.1 standard cubic feet per minute (scfm); maximum leak rate is 20 scim. Worst-case failure mode for a similar pressure activated valve leak on a flight engine would be immediate engine shutdown and abort on the pad. The pressure activated valve that failed had experienced 90 tests and 24,239 sec of operation. The pressure activated valve with the most operation on STS-36 SSMEs had experienced 11 tests and 3,743 sec of operation. Preflight pneumatic control assembly internal leak checks of STS-36 SSMEs verified that there was no pressure activated vent port leakage. Review of prior STS-36 engine operation data demonstrated normal purge pressure levels, which also indicated no outlet port leakage. ***This risk factor was resolved for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Left SRB Rate Gyro Assembly (RGA) #1 yaw output failure. HR No. B-50-18 Rev. C ***There were no RGA failures on STS-36.***	During S0008 testing in the VAB, the left SRB RGA #1 S/N 105 yaw output exceeded the null tolerance of 0 ±0.22 degree/second (°/sec). At power-up, RGA #1 yaw rate read 0.26°/sec and slowly increased to 0.50°/sec during the next 10 min of operation. S/N 105 was removed and replaced with S/N 117. Repeat of the associated S0008 steps was successful. ***This risk factor was resolved for STS-36.***
2	Thermal curtain bracket installation. HR No. B-60-50 Rev. C ***Postflight SRB inspection did not find and thermal curtain problems on STS-36.***	Postflight inspection of STS-32 found 32 helicoils on the LH SRB and 2 helicoils on the RH SRB backed out of the threaded holes. Most were backed out 1-4 turns; some were stretched out. This condition was also noted on STS-33 SRBs, where approximately 80 helicoils were backed out. The concern was that a gap created by differing thicknesses of thermal curtain material causes the helicoils to partially pull out of the nozzle compliance ring during installation. This gap ranged from 0.043" to 0.153" along the upper bolt row. Structural analysis showed that the thermal curtain bracket is fail-safe. Thermal analysis showed no detrimental effects of backed-out helicoils. If the thermal curtain fails (independent failure mode), TVC components would be vulnerable. There was no potential debris problem, because the top-row bolts are safety wired to the bottom row. ***This risk factor was acceptable for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
3	Elimination of SRB Holddown Post (HDP) Debris Containment System (DCS) frangible link. HR No. B-60-12 Rev. B ***There were no HDP anomalies reported on STS-36. Nearly all debris was captured in the DCS. Additionally, there was no report of HDP stud hangup on STS-36.***	For the first time since STS-30, a large amount (approximately 7 pounds (lb)) of debris escaped from the HDP DCS during STS-32 liftoff. This was directly attributed to removal of the frangible link from the DCS. For the first 3 missions since reflight, debris escaped from the DCS in large quantities. Concern was raised relative to the possibility of this debris striking the Space Shuttle Vehicle (SSV). The frangible link modification was installed just prior to STS-28 and was successful in containing nearly 100% of the debris formed. Beginning with STS-34, HDP stud hangups and broaching were experienced. This problem was again experienced on STS-33. The concern was that adverse loads were introduced into the aft skirt and SSV due to multiple HDP stud hangup. For this reason, it was recommended that the frangible link modification should be removed from the STS-32 HDP DCS. While HDP stud hangup was not a problem on STS-32, HDP debris escape was. Options for STS-36 were presented at the February 8, 1990, PRCB. These options included: • Install the A-286 frangible link used from STS-26 through STS-30 (allows minor debris escape). • Install the NP35N frangible link used on STS-28 through STS-33 (captured over 99% of debris formed). • Do not install either frangible link. The Level II PRCB decided to install the A-286 frangible link used from STS-26 through STS-30. While the A-286 link has increased debris risk compared to the NP35N link, no stud hangups were reported when it was used.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
3 (Continued)		Efforts are underway to optimize frangible link design to reduce the potential for stud hangup. ***This risk factor was acceptable for STS-36.***
4	TVC fuel filter bowl drain cap hazard upgrade. HR No. A-20-24 B-20-04 ***There were no Quick Disconnect (QD) anomalies reported on STS-36.***	A Change Request (CR) was submitted to the Level II PRCB to reclassify Hazard Report (HR) A-20-24, Hydrazine Leakage (during final countdown), and B-20-04, Hydrazine Source in the Presence of an Ignition Source (during boost phase), from "Controlled" to "Accepted Risk." This classification change was related to the hazard of hydrazine leakage past the Hydraulic Power Unit (HPU)/APU fuel pump filter bowl drain QD and its cap that forms a bore seal on the QD. The cap is the secondary seal in this case. The concern for leakage was based upon the absence of torquing/lockwiring to secure the HPU/APU pump filter bowl drain QD cap and the possibility of the cap backing off as a result of vibration during flight. The issue that surfaced during the PRCB was lack of data to confirm that the QD cap would not back off in a flight vibration environment while under pressure. The QD cap had not been subjected to a flight vibration environment while under pressure. MSFC was tasked to investigate this issue and reported to the PRCB that combined vibration and operating (leakage) pressure are not detrimental to caps torqued to 20-25 in-lb. Consideration was given to the additional provision of adding a safety wire to the QD caps. The subject HRs retained the hazard-level classification of "Controlled."

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u> 4 (Continued)		Rationale for STS-36 flight was: • The QD primary and secondary seals were each designed for 4000 psi, an FOS 2 times that necessary to meet the operating pressure of 1500 psi. • The cap and poppet were proof tested to 2250 psi individually and together. • A Problem Reporting And Corrective Action (PRACA) system review showed no history of caps backing off and no previous record of leaks or QD failure in the area of the HPU/APU fuel system. • An engineering CR was generated to add torque specifications to the QD caps. All 5 STS-36 APU QD caps were torqued to 25 in-lb. ***This risk factor was resolved for STS-36.***
5	Potential criticality 1 failure mode related to Solid Rocket Beacon Tracking System (SRBTS). HR No. SRBTS-01 ***The STS-36 SRBTS performed with no anomalies.***	Hazard analysis identified a potential criticality 1 failure mode related to the SRBTS antenna based on the following event sequence: • Material deficiency in the quartz-to-baseplate epoxy bond or deficient workmanship that results in bond failure. • The quartz becomes disengaged from the baseplate and is a debris source. In either case, the resultant debris has a high probability of impacting the Orbiter. The epoxy used (Tra-Duct 2902) is a conductive silver epoxy that contributes to antenna operation. The antenna ground plane is provided through the epoxy conductive path from the antenna baseplate to the quartz.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
5 (Continued)		The antenna used in the SRBTS was qualified by White Sands Missile Range (WSMR) to meet or exceed the severity of SRB flight environments. During qualification and production, SRBTS processing, inspections, controls, and tests remained the same. To date, no discrepancies were noted relative to debonds or quartz disengagement. The Voltage Standing Wave Ratio (VSWR) tests do not verify a proper bond, as specified earlier in SRB HR No. SRBTS-01. SRBTS-01 was updated to delete the VSWR test as a control under the heading of material defect. Fifteen antennas were pull tested without a failure; this included 1 antenna that had experienced splashdown and ascent loads. The pull test load level corresponds to 2x the maximum ascent loads, a plus 0.103 margin. The antennas on STS-36 were not tested because they were not accessible. Additionally, the SRB Project performed a test on the bond and found that the quartz would yield (rupture) before the epoxy bond. There is no history of bond failure on any qualification or production units. Bond line design provides a safety factor greater than 30. ***This risk factor was acceptable for STS-36.***
6	Instrument and Electronics Assembly (IEA) battery temperature sensor anomaly. HR No. A-00-02 Rev. B-PDCN1 B-00-01 Rev. B-PDCN1 ***There were no IEA anomalies reported on STS-36.***	IEA recovery battery temperatures are monitored during SRB pad validation testing. On STS-36, the RH recovery battery temperature, monitored through IEA S/N 012, was recorded to be 17.4°F; this temperature should be 60°F. The low-temperature reading was indicative of an open circuit. Troubleshooting found no problem with the IEA signal conditioning, recovery battery thermostat, or interconnecting cabling. The anomalous temperature reading could not be duplicated, and the recovery battery temperature was within the expected range. A similar anomaly occurred during STS-28 prelaunch validation with IEA S/N 020 in a battery temperature sensor circuit. This failure was duplicated using a second signal conditioning card. Troubleshooting continues on the STS-36 IEA.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
6 (Continued)		The most probable cause of this anomaly was an intermittent contact between the IEA signal conditioner card and the card connector. This was based on a similar problem encountered with IEA S/N 012 at the vendor, Allied Signal, in April 1989. The vendor anomaly related to the External Tank (ET) Range Safety System (RSS) battery temperature measurement. This problem was closed by replacing the socket portion of the conditioner card connector. Card connectors of this type are only used with criticality 3 signal conditioner cards. Rationale for STS-36 flight was: • The temperature circuit in question is a criticality 3 function. • Low recovery battery temperature is a violation of a Launch Commit Criteria (LCC) and results in a contingency plan to monitor voltage and current to determine the health of the battery. ***This risk factor was resolved for STS-36.***
7	STS-36 SRB Fuel Supply Module (FSM) pressure drop. HR No. B-20-04 Rev. B-DCN1 ***Gaseous Nitrogen (GN_2) tank pressure was above the LCC at STS-36 launch. No further anomalies were reported.***	During post hot-fire leak checks of the SRBs, a 1 psi/day GN_2 blanket pressure decay was noted on the LH SRB tilt FSM. Troubleshooting, including repressurization and additional leak checks, was performed. Upon removal of the foam insulation, a small hydrazine leak, less than 1 part per million (ppm), was noted at the B-nut-to-dynatube fitting on the FSM horizontal drain line. Further troubleshooting found a leak at a "T" fitting in the GN_2 pressure and purge line. KSC discussion with United Space Boosters, Inc. (USBI) and MSFC led to retorquing the B-nuts to 173 in-lb. There was no indication of excessive yielding during the torquing (173 in-lb is 3% above the OMRSD requirement of 168 in-lb). At 173 in-lb torquing, leakage of less than 0.5 ppm with no bubbles was observed

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
7 (Continued)		on one side of the "T" fitting and 0.5 ppm on the other side with bubble indication; this was well within specification. Other options included repressurizing the GN_2 tank to 400 psi. Given the leak rate, GN_2 tank pressure of 400 psi provided sufficient GN_2 to support the late February launch. ***This risk factor was resolved for STS-36.***
8	Improper helicoils used in SRB floor plate. HR No. INTG-081A ***The SRB floor plates remained in place during STS-36 first stage through recovery. There was no indication of loose fasteners/helicoils during postflight inspection.***	The SRB Project Manager reported that the wrong size helicoils were used in a floor plate near the aft skirt on both SRBs. It was determined that 1/4", 28-thread (fine) helicoils were placed in the same bin as 1/4", 20-thread (course) helicoils and were subsequently installed in the floor plate in 20-thread tapped holes. Three helicoils secure the splice plate in a low-load environment (417 pounds force (lbf) total). Indications were that these improper helicoils were also used for the same application on STS-33 and STS-32 SRBs. Initial tests found that the 28-thread helicoils were not as tight for the first few turns, but torqued down well. Two of the 3 helicoils are covered by cabling, and the splice plate should also remain in place under the same cabling. Tests of the helicoil fastener assembly indicated that it will carry greater than 2-1/2 times the design load. Design FOS is 2.0. Fastener preload was determined to be sufficient to sustain flight environments. ***This risk factor was acceptable for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Standard-weight, forward cylinder S/N 37 thin spot. HR No. BC-09 Rev. B ***No anomalies were attributed to the reported thin spot on STS-36.***	It was determined that the standard-weight, forward cylinder of the forward casting segment S/N 37 had a localized thin spot on the inner surface of the membrane wall. This was the result of segment grit blasting performed in 1984. The local wall thickness at the suspect location could be as thin as 0.426"; the minimum allowable thickness is 0.477". The wall thickness was determined with the use of ultrasonic measuring techniques. Because the ultrasonic equipment is accurate to only ±0.010", the actual thickness of S/N 37 was not totally known; therefore, the structural margin of safety was in question. Additionally, the ultrasonic equipment used (Nova gage) was not qualified for this application. The thin spot was documented in a Discrepancy Report (DR) after the grit blasting, and the segment was dispositioned for use as is. This was based on a two-dimensional axial-symmetric analysis performed in 1984 which predicted a margin of safety of 0.012 at the Maximum Expected Operating Pressure (MEOP). Hydroproof testing of the segment found, using biaxial strain gages at the thin spot, that no yielding occurred on the outer segment surface. This segment was hydroproof tested a total of 7 times; 3 tests were conducted after 1984. S/N 37 had flown 3 times, the last after the thin spot was originally dispositioned. Magnetic particle inspection found no cracks at the thin spot. Three-dimensional, elastic-plastic analysis of S/N 37 was performed. This analysis included true stress-strain curves of S/N 37 based on recent segment material testing. The minimum yield strength of the segment is 187.5 ksi based on tensile tests of segment trim material. Ultimate material tensile strength is 200.0 ksi. The model used to predict failure modes used a wall thickness of 0.492" for the membrane surrounding the thin spot, and the measured thickness of 0.426" was used for the center of the thin spot. With maximum liftoff loads/pressures, the model verified the hydroproof test strain gauge results. Four failure modes were predicted using this model; radial deflection, ultimate strength using the biaxial improvement

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)		factor, general membrane yielding, and ductile rupture. Of these, the biaxial improvement method produced the lowest margin of safety: 0.013. This was considered a very conservative method for determining margin of safety. The January 26, 1990, Level I PRCB accepted the 0.013 margin of safety for STS-36/Redesigned Solid Rocket Motor (RSRM) Flight Set #9. ***This risk factor was acceptable for STS-36.***
2	Nozzle joint #5 removal with a new torque tool. HR No. BC-04 Rev. B ***No nozzle joint problems were reported on STS-36 SRMs.***	Defective Stat-O-Seals in nozzle internal joint #5 (fixed housing/bearing) required removal and replacement. The concern was the integrity of joint #5 with the use of a new torque tool. The STS-36 nozzles were removed from the STS-33 aft segments after discovery that joint #5 Stat-O-Seals were improperly fabricated. (The vendor failed to heat treat the Stat-O-Seals.) A torque tool was designed and used to remove and replace bolts, one at a time, to avoid disassembly of the snubbers and the forward exit cone. Rationale for STS-36 flight was: • Metrology calibrated the joint #5 torque tool using normal process torque wrench and torque tool calibration standards. • Laboratory and breakaway tests showed no statistically detectable variance between the normal process torque wrench and joint #5 torque tool.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2 (Continued)		• The joint #5 torque tool was used and successfully demonstrated on TEM-4 (Test and Evaluation Motor) joint #5. • The bolts are visible, and replacement of all Stat-O-Seals was verified by temporarily marking the bolts. ***This risk factor was resolved for STS-36.***
3	STS-31 nozzle internal joint leak test investigation. HR No. BN-03 Rev. B ***No nozzle joint problems were reported on STS-36 SRMs. Recent analysis of this problem at Thiokol found grease in the nozzle joint #3 leak check port. An investigation is underway to determine the source of the grease.***	An engineering review of STS-31/RSRM Flight Set #10B nozzle joint leak test results determined that the leak test on nozzle joint #3 was not performed properly. Leak checks of nozzle joint seals are performed by pressurizing the volume between the two O-ring seals and recording pressure leak rates. Two pressure tests are performed on each joint; high and low pressure. The high-pressure test is employed to check for contamination and major seal defects. Low-pressure tests identify possible minor seal/surface damage, damage incurred during assembly, and minor contamination. Using Boyle's Law, calculations are made to determine the joint volume, based on the known volume of the test equipment and the recorded initial pressure of the test equipment and the final pressure of the test equipment and pressurized joint. In the case of RSRM Flight Set #10B nozzle joint #3, calculations indicated that the joint was not pressurized. Under both high- and low-pressure tests, the final volume of joint #3 was calculated to be near 0; 0.1 and –0.07 cubic inch (in^3), respectively. Average low-pressure volume of joint #3 is 0.87 in^3, and the average high-pressure volume is 1.45 in^3. Without tearing apart the joint, no definite cause was found to explain the low joint volume. For this reason, Space Shuttle Program Management directed that the STS-31/RSRM Flight Set #10B aft segment/nozzle be destacked and replaced.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
3 (Continued)		Reevaluation was made of STS-36/RSRM Flight Sets #9A and #9B joint leak test results. Comparison of "as built" and the "as performed" leak tests determined that all joints, including nozzle joint #3, were properly tested. It was impossible to compare resulting joints volumes with the average low- and high-pressure volumes because RSRM Flight Set #9 leak tests were performed with alternate test equipment. However, recorded data and calculations demonstrated that joints were pressurized and resulting leak rates were within the allowable limit. This reevaluation cleared STS-36/RSRM Flight Set #9A and #9B for flight. ***This risk factor was resolved for STS-36.***
4	Voids in the SRM nozzle radial bonds between the phenolic rings. HR No. BN-04 Rev. B ***No STS-36/SRM anomalies were attributed to the presence of these voids.***	Review of STS-36 RSRM Flight Set #9 nozzle x-rays found small, intermittent adhesive voids in the nozzle nose cap-to-forward nose phenolic ring radial gap. The design objective for adhesive in the nozzle radial gaps is to provide a thermal barrier to protect nozzle metal parts and O-rings. The worst-case thermal environment seen by the metal components and O-rings, assuming no adhesive in the radial bond, would be 130°F; this is well below the design requirement for aluminum parts (250°F) and steel components (500°F). Given the observed voids, analysis indicated that the metal components remain at ambient temperature throughout the SRM operation. The adhesive is needed, however. Total absence of adhesive could result in circumferential flow concerns in the nozzle. Process controls and visual inspection at assembly confirm the presence of adhesive. There is no structural requirement for the adhesive; therefore, no degradation in the structural FOS results from these voids. Adequacy of the nozzle radial gap adhesive system and flight hardware is based on critical process controls and post-fire flight hardware evaluation of over 85 SRM nozzles.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
4 (Continued)		Rationale for flight of STS-36/RSRM Flight Set #9 nozzles with known adhesive voids was based on the fact that presence of these voids in the radial bonds did not compromise the required thermal or structural safety factors. ***This risk factor was acceptable for STS-36.***
5	SRM Ignition Initiator (SII) manufacturing anomalies. HR No. BI-05 Rev. B ***STS-36 SIIs performed nominally.***	The SII is a modified NASA Standard Initiator (NSI); a second seal is added to the SII because it experiences motor heat and pressure for the duration of motor firing. As a result, seal redundancy is needed. Per discussion by the System Safety Review Panel (SSRP) on February 15 and 16, 1990, other NSIs experience the heat and pressure only briefly; therefore, they do not require redundant seals. Manufacturing anomalies were found in the SII sealing areas. Threads on the SII were found to extend into the primary sealing surface. Deformation of the secondary O-ring sealing washer was noted. There was degradation of bearing load on the SII port as a result of weld deformations that were allowed. The concern was that the sealing surface of the SII would not meet the sealing specifications (O-ring squeeze requirements) and that redundancy would be violated. Thiokol took action to separate out all SIIs that did not meet the new inspection criteria. Revision to the NASA SII drawing resulted in removal of the 0.004" secondary seal flatness requirements (allows sealing washer to deform to uneven surface). This condition was documented at Thiokol both during SII acceptance and postflight inspection. These SIIs were used in flight hardware since the acceptance inspection was for damage or contamination and not manufacturing defects.

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
5 (Continued)		A preliminary Alert, G6-A-88-04, was issued by Thiokol in February 1989. On February 9, 1990, SII sealing area anomalies were identified as a problem in Significant Problem Report DR4-5/188. (A hazard notification on this issue is being prepared.) SIIs are supplied to Thiokol by NASA as Government Furnished Equipment (GFE). Two SIIs are located 180° apart on each Safe & Arm (S&A) device (a total of 4 SIIs per flight set). SIIs are connected to the SRB firing lines, receive the necessary voltage, then fire into the Pyrotechnic Basket Assembly that fires into the igniter initiator, which ignite the igniters and finally the SRM propellant. The rationale for STS-36 flight was: • Both primary and secondary seals were static seals. • A damaged primary O-ring was never found on a dissembled SII. • 25 fired SIIs per lot passed a high-pressure between-seals leak test (3640 psi) at the supplier before delivery to Thiokol. • SII seals never leaked. (This primary seal configuration was test-fired on hundreds of S&As. SII secondary seals were tested on Transient Pressure Test Article (TPTA)-1.3 and TPTA-2.2.)

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
5 (Continued)		• All flight SII seals have passed the standard 50-psi between-seals leak test as installed in the barrier-booster. • Since these are static seals, the leak tests do verify seal performance in the flight condition. ***This risk factor was resolved for STS-36***
6	RSRM igniter joint deflections. HR No. BC-03 Rev. B BI-02 Rev. B ***No problems were reported relative to joint deflections on STS-36. For subsequent flights, the LCC minimum joint temperature has been raised to 100°F.***	JSC Safety personnel, along with MSFC Safety counterparts, identified a concern regarding RSRM igniter joint deflections that could result in the need to raise the LCC minimum allowable temperature. Concerns derived from an MSFC engineering assessment of the Thiokol analysis for determining the minimum acceptable joint temperature for launch placed doubt on the validity of the inner joint hot-fire test data and subsequent LCC temperature derivation. Concerns included: instrumentation accuracy not compatible with magnitude of deflections being measured; non-consideration of data from one deflection gauge that apparently experienced less temperature effects than the gauge considered valid; knockdown factor applied to the LCC equation that resulted in the LCC being applicable for only one lot (lot 38) of igniters; and the application of a "seal factor" that could not be assessed for validity due to lack of explanation. As a result, the LCC temperature, that was based on a 0.0031" maximum expected gap (derived from the hot-fire data), was thought to be inadequate to ensure a 1.4 safety factor for the inner Gask-O-Seal.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
6 (Continued)		Thiokol reevaluated the hot-fire data. As a result, a new deflection of 0.0036" was indicated vice the previous 0.0031". Thiokol performed long-term compression tests of the elastomer used in the Gask-O-Seal to determine resiliency of the material versus time of compression. Preliminary results indicated that the elastomer is more resilient after long-term compression than previously assumed. This added resiliency provides sufficient margin to ensure the 1.4 safety factor. The issue of LCC temperature would become a factor only if both of the redundant heaters for this joint fail and the ambient temperature are low enough to drive the joint temperature into the questionable region. To date, no igniter joint heater problems were reported, nor are there any anticipated. ***This risk factor was resolved for STS-36.***

RESOLVED STS-36 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ET</u>		
1	First Flight of reworked ET SRB fittings. HR No. S.11 ***There were no problems attributed to these reworked fittings on STS-36.***	The October 1986 review of ET acceptance data indicated that installed SRB fittings were improperly heat treated and susceptible to stress corrosion cracking on 5 delivered ETs (STS-36/ET-33, STS-40/ET-39, STS-38/ET-40, STS-39/ET-42, and ET-55 intertank). It was determined that reheat treatment was required for the STS-36/ET-33-Y SRB forward fitting. The fitting was removed, reworked, and returned to the required specification by reheat treatment. The fitting has a structural FOS of 2.08. Stress analyses of the intertank mainframe splices indicated an FOS of 9.90. The intertank mainframe outer chord cut web tang FOS is 2.76. All electrical and mechanical hardware affected during rework successfully passed acceptance retesting. The reworked configuration met all flight requirements. ***This risk factor was resolved for STS-36.***

SECTION 5

STS-32 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the OV-102/STS-32 mission. Each anomaly is briefly described, and risk acceptance information and rationale are provided.

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STS-32 INFLIGHT ANOMALIES

SECTION 5 INDEX - CONTINUED

STS-32 INFLIGHT ANOMALIES

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Auxiliary Power Unit (APU) #3 lubrication oil outlet pressure high. IFA No. STS-32-02 HR No. ORBI-036 ***APU lubrication oil outlet pressures were in the nominal range during STS-36 operations.***	Higher than expected lube oil outlet pressure was observed on OV-102/STS-32 APU #3. The lube oil pressure was 75 to 95 pounds per square inch absolute (psia) for the first 10 minutes (min), then returned to normal (55-60 psia) at full operating temperature. The pressure was nominal for ascent and landing. This anomaly was similar to the APU #1 anomaly on STS-33 (IFA No. STS-33-01). It was determined that the high pressure was caused by the presence of hydrazine in the gearbox. Hydrazine reacts with the lube oil to form pentaerythritol and hydrazides, which liquefy at approximately 200°F. This contamination/wax buildup collects on the lube oil filter and partially blocks the filter. Heating of the oil causes the contaminant to liquify and clears the filter. Continued operation disperses the contaminant throughout the gear box. It is believed that the seepage of hydrazine past the seals into the gearbox is due to the postlanding gearbox pressure being below the seal cavity pressure. The worst-case criticality is 1R/2 for gross hydrazine leakage into the gearbox and loss of the APU. The Launch Commit Criteria (LCC) allows for lube oil outlet pressures up to 110 psia. The Flight Rules do not address lube oil pressure, and the temperatures were normal; therefore, this anomaly did not result in a violation of the Flight Rules. The most probable effect of wax buildup is an increase in lube oil outlet pressure for a 10-min period, which does not affect the operation of the APU. The solution to this problem is postlanding pressurization of the gearbox above the pressure of the seal cavity. Hot oil flushing techniques were developed, which allow the wax to melt and the system to be flushed. A Requirements Change Notice (RCN) was issued to change the Operational Maintenance Requirements and

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)		Specifications Document (OMRSD); it includes the new procedures for postlanding pressurization of the gearbox to no less than 5 psia greater than the seal cavity pressure. APU #3 was removed; the system was flushed, sampled, and reserviced. Rationale for STS-36 flight was: • OV-104 APUs flew 3 flights with no lubrication oil contamination. • Oil outlet pressures and gearbox pressures were nominal. • Gearbox lube oil samples were taken during the STS-36 turnaround and verified to be within specification. • New lube oil and filters were installed during turnaround processing. ***This anomaly was not a safety concern for STS-36.***
2	Right-Hand (RH) Orbital Maneuvering System (OMS) "no-back" device moved during ascent. IFA No. STS-32-04 ***There were no similar anomalies on STS-36.***	The RH OMS yaw actuator, Serial Number (S/N) 117, drifted 0.112° during the first 50 seconds (sec) after the launch of STS-32. The "no-back" device is designed to prevent back-drive or movement of the actuator during powered operations. After the initial movement, the "no-back" device held the actuator as required for the remainder of ascent. During entry, movement of 0.048° was recorded. Failure of the "no-back" device could result in positioning the OMS engine nozzle into the air flow. The air flow dynamic pressure will cause a nozzle to deform.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)		Review of previous flights using this yaw actuator was completed. On STS-28, there was indication of 0.082° movement. The previous mission employing S/N 117, STS-61C, recorded movement of 0.098°. Movement in each of the 3 missions reviewed occurred during ascent when the highest vibration environment is experienced. Because the occurrences of movement during ascent are relatively consistent, there was no indication of degradation. Movement similar to that witnessed on STS-32/OV-102 was not considered detrimental to actuator function since no significant problem exists until 1.5° of movement occurs during launch. If movement occurs during major modes 102 and 103 of OPS 1 (ascent software), the software will automatically power up the OMS controller primary channel if the gimballed angle reaches 0.7° and commands the actuator to the stow position. In either software mode, an alarm will sound alerting the crew to manually select the backup channel. An RCN changed the acceptable limit for actuator movement to 0.2°. No troubleshooting of this anomaly was anticipated. ***This anomaly was not a safety concern for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Gaseous Oxygen (GOX) Flow Control Valve (FCV) #2 opened sluggishly. IFA No. STS-32-06 HR No. INTG-150A ORBI-248A ***All GOX FCVs performed nominally on STS-36.***	The GOX FCV #2 exhibited sluggish operation when first cycled open at T+61 sec. This FCV took approximately 0.75 sec to open versus the nominal operation of 0.2 to 0.4 sec. Sluggish operation of this FCV was only experienced during the first open cycle; all other closed-open/open-closed cycles were nominal. Investigation of previously experienced sluggish GOX FCV operation found contamination as the cause of the slowness. This was the first instance of sluggish GOX FCVs since STS-29. All GOX FCVs on OV-102 were replaced with new reshimmed valves during the STS-35/OV-102 turnaround process. Poppet installation was completed, leak and functional tests were completed, and signature testing was completed. FCVs are scheduled for replacement with fixed orifices in the fall of 1990. Rationale for STS-36 flight was: • FCVs are redundant; sluggish action of 1 FCV would not affect External Tank (ET) ullage pressure. • Most likely cause of sluggish behavior was transient contamination. • No FCV has failed open or closed to date. • FCVs on OV-104/STS-36 passed all turnaround tests and leak checks. ***This anomaly was not a safety concern for STS-36.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Humidity separator "B" water bypass. IFA No. STS-32-07A HR No. ORBI-051 ORBI-254 ORBI-321A *The crew reported finding approximately 2 cups of free water outside of humidity separator "A" (IFA No. STS-36-11). Postflight vendor analysis found contamination in the humidity separator pitot tube inlet, partially blocking it.*	During changeout of the Lithium Hydroxide (LiOH) canister, the crew discovered uncontained water. They indicated that the LiOH canister was wet, and water was coming from the humidity separator "B" exit port. At the time, humidity in the cabin was nominal, and there was no indication of a humidity separator problem. The crew switched from humidity separator "B" to "A" and initiated free fluid cleanup procedures. Approximately 2 gallons of water was collected. The separator was removed and sent to the vendor for failure analysis. Note that humidity separator anomalies occurred on OV-103 (STS-26) and OV-104 (STS-27). The cause of the failures was believed to be due to an accumulation of debris on the condensing heat exchanger and high humidity. There was no recurrence of the problem on subsequent flights of OV-103 or OV-104 since the refurbishment of their humidity separators. The units used on STS-36 passed borescope inspection. The OV-104/STS-36 system was flushed of accumulated contamination. Crew procedures were available should a problem occur during STS-36 flight, and the separators are redundant. *This anomaly was not a safety concern for STS-36.*

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Humidity separator "A" water bypass. IFA No. STS-32-07B HR No. ORBI-051 ORBI-254 ORBI-321A ***The crew reported finding approximately 2 cups of free water outside of humidity separator "A" (IFA No. STS-36-11). Postflight vendor analysis found contamination in the humidity separator pitot tube inlet, partially blocking it.***	On Flight Day (FD) 6, the crew found approximately 8 ounces (oz) of water around humidity separator "A". Inflight maintenance procedures (towel and bag) were initiated to collect additional water that might escape from humidity separator "A". On FD 7, the crew reported more water coming from humidity separator "A". Cleanup procedures were again initiated, and approximately 2 cups of water were collected. It was postulated that this water escape resulted from the high level of crew activity. Separator "A" leakage was most likely due to carryover of water from humidity separator "B". Water was not collected for the remainder of the flight. The humidity separator was removed and sent to the vendor for failure analysis. OV-104 humidity separators did not demonstrated similar anomalies during previous missions. No additional testing of OV-104 humidity separators was accomplished prior to STS-36. A near-term procedure was developed to have the crew inspect the humidity separators for water prior to and after sleep periods. This procedure was implemented on STS-36. ***This anomaly was not a safety concern for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	Flash Evaporator System (FES) topping duct "B" string heater failure. IFA No. STS-32-14 HR No. ORBI-276B ***There were no failures of the FES heaters on STS-36.***	After activation of the FES topping duct heater "B", the aft duct temperature did not increase at the correct rate. This occurred during inflight checkout of redundant heater strings on FD 7. FES topping duct heater "A" was selected and operated nominally for the remainder of the mission. Worst-case effects occur with loss of all 3 redundant heater strings, resulting in loss of the FES. Flight Rule 9-71B states that, if the FES topping duct cannot be maintained at 100°F or more, the FES is considered lost and results in a minimum duration mission. Troubleshooting determined that Remote Power Controller (RPC) #34 was defective. The unit was removed, replaced, and retested. Rationale for STS-36 flight was based on: • FES redundancy. • Heater redundancy. ***This anomaly was not a safety concern for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	Inertial Measurement Unit (IMU) #1 was deselected by Redundancy Management (RM) due to Y-axis transients. IFA No. STS-32-15 HR No. ORBI-051 ***There were no IMU anomalies on STS-36.***	IMU #1 S/N 24 was deselected by RM on FD 6 due to 7 occurrences of erratic Y-axis accelerometer transients; however, IMU #1 continued to track the redundant IMU set after deselection. The crew was able to reselect IMU #1 prior to IMU alignment. After IMU alignment, all 3 IMUs were operating nominally. IMU #1 was deselected prior to crew sleep periods to avoid waking the crew should an alarm occur. No further problems were reported with this IMU for the remainder of the mission. The crew continued to deselect IMU #1 prior to their sleep periods as a precaution. Playback data indicated that the failure was caused by multiple Y-axis velocity transients. No previous problems associated with this IMU were reported. Additionally, there was no indication that this is a generic problem. IMU #1 was replaced. The removed S/N 24 unit was sent to Johnson Space Center (JSC) for failure analysis. The IMU system has triple redundancy. ***This anomaly was not a safety concern for STS-36.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8	Hydraulic systems #1 and #2 unloader valves exhibited anomalous operation. IFA No. STS-32-16 HR No. ORBI-052 ***There were no similar anomalies on STS-36.***	Approximately 1 hour (hr) prior to circulation pump #2 deactivation during the launch scrub turnaround, there was a significant increase in unloader valve #2 cycling. Approximately 45 min after deactivation of circulation pump #2, all bootstrap fluid pressure was lost. Because of a similar anomaly on STS-28 (IFA No. STS-28-23), the decision was made to replace the hydraulic system #2 unloader valve prior to the STS-32 launch. The hydraulic system #1 unloader valve leaked excessively once the accumulator pressure fell below 2300 psia. This was an internal hydraulic leak with hydraulic fluid on the high-pressure accumulator side leaking to the low-pressure return side. Prior to this flight, it was known that all 3 OV-102 unloader valves were experiencing out-of-specification leakage (see STS-32 Mission Safety Evaluation (MSE), Section 4, Orbiter 9). This condition was waived (WK 1547) for 1 flight only, STS-32, with the understanding that hydraulic accumulator pressures would be closely monitored during prelaunch activities. It was believed that contamination in the unloader valve pilot area caused the leakage. System #1 unloader valve failed during testing and was removed, replaced, and successfully retested. Systems #2 and #3 data were acceptable. There were no constraints to STS-36 launch. There was no indication of anomalous unloader valve operation on OV-104. Bootstrap pressure can be maintained with increased circulation pump operation as was done on STS-32. Additionally, loss of one hydraulic system is acceptable. ***This anomaly was resolved for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	Water Spray Boilers (WSBs) #2 and #3 regulator pressure decaying slowly. IFA No. STS-32-17 HR No. INTG-072 INTG-117 ***There were no similar anomalies on STS-36.***	WSB systems #2 and #3 pressure decay rates were 0.11 pounds per square inch per hour (psi/hr); the allowable decay rate is 0.06 psi/hr. Probable cause of the pressure decay was determined to be either a water or Gaseous Nitrogen (GN_2) leak. Monitoring of the pressure decay rate throughout the mission found that the decay rate approached 0 by the end of the mission. There is a tank in which GN_2 is stored for use in pressurizing the WSB water storage tank. GN_2 is routed from this tank through a regulator and relief valve prior to entering the water storage tank. Pressure loss is measured between the regulator/relief valve and the water tank. GN_2 pressure loss was experienced previously and was attributed to GN_2 leakage overboard through an improperly reseated relief valve. The relief valve "burped" on ascent and reseated when pressure is greater than 28 pounds per square inch (psi). Other possible GN_2 leaks could occur at the pressure transducer port or at the GN_2 vent Quick Disconnect (QD); however, both failure modes are very unlikely. The other possible cause for the observed loss of pressure was loss of water in the storage tank. If water loss was the problem, a reduced mission duration would result. At the originally recorded decay rate, only 7.5 min of APU operation would be available due to APU bearing temperature constraints. It is believed that the pressure decays were due to the GN_2 relief valves not being fully seated and not due to water leaks. The poppets in the relief valves were removed and replaced. GN_2 24-hr decay check on system #2 indicated no leakage; decay check on system #3 indicated leakage of 0.06 psi/hr, which was within OMRSD limits. ***This anomaly was resolved for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
10	Avionics bay #3A smoke detector transient alarm and associated lights. IFA No. STS-32-19 HR NO. ORBI-259A ***There were no smoke detector/sensor anomalies on STS-36.***	On FD 8, the crew reported a smoke alarm with siren from avionics bay #3A, sensor 3A. The alarm cleared itself in approximately 6 sec. Playback data indicated that there was no increase in the smoke concentration readings. A successful fire/smoke detection test was subsequently performed indicating that there was no problem with the detection system. It was concluded that an intermittent fault in the smoke detection electronics most likely caused the alarm. Sensor 3A operated nominally for the remainder of the day after the initial alarm. There were several previous instances of smoke detection failures where the cause could not be found. Sensor 3A annunciated several additional times during the crew sleep period on FDs 9 and 10. Each time the crew checked for increase smoke concentrations; no increase was noted. A decision was made to pull the sensor circuit breaker to avoid nuisance alarms during the crew sleep period. Continued nuisance alarms prompted a decision to open the circuit breaker during reentry. This resulted in the loss of smoke detection redundancy in avionics bays #1 and #3. Safety concurred with this plan based on the fire/smoke detection/suppression test for avionics bays #1, #2, and #3 and after evaluating the risk of loss of redundancy against possible crew distraction during reentry and landing. Sensor 3A was removed and replaced. The defective unit was sent to the vendor for failure analysis. The replacement unit was successfully tested.

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
10 (Continued)		Rationale for STS-36 flight was: • There are alternate means to verify real or false alarms. • If false alarms recur, a circuit breaker can be pulled to eliminate the "nuisance" factor. ***This anomaly was resolved for STS-36.***
11	Waste water dump line/nozzle blockage. IFA No. STS-32-21 HR No. ORBI-254 ***There were no similar anomalies on STS-36. The water collection wand worked correctly while gathering free water from humidity separator "A".***	During free fluid disposal on FD 10, the crew did not get any suction through the collection wand. A later attempt at dumping the waste tank was also unsuccessful. Troubleshooting and visual inspection of the dump nozzles using the Remote Manipulator System (RMS) determined that there was no icing in the dump nozzles. It was suspected that a restriction formed in the waste dump line. Inspection at Dryden found charred material around the urine dump nozzle face. This was experienced previously; however, there was more than usual. A sample of the material was taken for analysis. A sample taken from the orifice indicated that some potassium was among the charred material; everything else was normal. "Mucky junk" was flushed from the dump line. Troubleshooting confirmed that the dump line was clogged. The line was removed, and the replacement line/nozzle installation was completed. Leak checks and heater/insulation installation were completed. ***This anomaly was not a safety concern for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
12	WSB #3 controller "A" overcooling. IFA No. STS-32-23 HR No. ORBI-170 ***There were no WSB controller failures on STS-36.***	WSB #3 went into the heat exchanger mode early and dumped excessive water while operating on controller "A". The crew switched to controller "B", and operation continued nominally. There were no previous WSB failures of this type. Troubleshooting confirmed the failure of controller "A". The defective controller was removed, replaced, and successfully retested. A redundant controller is also available. In addition, redundant hydraulic systems are available if both controllers in one system malfunction. ***This anomaly was not a safety concern for STS-36.***
13	Main Propulsion System (MPS) Liquid Hydrogen (LH_2) outboard fill and drain relief valve leak. IFA No. STS-32-25 HR No. ORBI-306 ***There were no similar anomalies on STS-36.***	The MPS outboard fill and drain valve PV11 was found with a blowing leak during OV-102/STS-32 postflight inspection. The leak was heard and felt at the 6:30-o'clock valve position. Helium tank decrease confirmed the leak. Investigation indicated some contamination in the system. The PV11 valve is redundant, and a second failure (PV12 or PV13) would be required to cause a hazardous condition during main stage. PV11 was removed and replaced. Postflight MPS leak checks did not identify any problems on STS-36. ***This risk factor was resolved for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14	RH stop bolt was found slightly deformed on the STS-32 centering ring of the forward ET attach/separation assembly. IFA No. STS-32-26 HR No. INTG-051A ***Postflight inspection found no bent stop bolts on the STS-36 ET attach/separation assembly.***	Postflight inspection of STS-32/OV-102 found the RH stop bolt slightly deformed (not bent condition) on the centering ring of the forward ET attach/separation assembly. This anomaly was similar, but not nearly as bad as that seen previously on STS-34. Deformations or flat spots, similar to that seen on STS-32, were found on other flight and qualification bolts. A bent stop bolt was first found on STS-34/OV-104. STS-34 postflight inspection at Dryden found the RH stop bolt to be bent, forward and inboard. This bolt, located on the centering ring of the forward ET attach/separation assembly, was found compressed into the centering mechanism. It is used to restrict side motion at the attach/separation assembly between the ET and Orbiter and is not considered a structural bolt. Indications were that the assembly sustained a side load. The moment required to bend this bolt is in excess of 10,000 inch-pounds (in-lb). The force required to obtain this moment is 900 pound (lb). A side load of this magnitude could lead to early uncontrolled separation of the Orbiter from the ET. There was no indication that a side load occurred on either the STS-34 or STS-32 flights. The most probable cause of the STS-34 anomaly was determined to be improper sequencing of the ET/Orbiter mating procedure resulting in a yaw moment that could bend the bolt. Sequencing employs Ground Support Equipment (GSE) (H72-0590) that could produce the required loads. Improper sequencing would not lead to early, uncontrolled separation of the ET and Orbiter. However, a bent bolt extended into the airstream could result in excessive localized heating during reentry. An additional cause could be the use of the Orbiter transporter, which moves the Orbiter from the Orbiter Processing Facility (OPF) to the Vehicle Assembly Building (VAB). Since the transporter was first used on STS-32, it is possible that the bolt was deformed at the Orbiter-to-transporter mate.

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)		There were no anomalies recorded during STS-32 ET/Orbiter mating. A bent stop bolt is a criticality 3 failure. Analysis indicated that a moment of 430-2100 in-lb could locally deform the bolt end. This moment could be generated by either side-to-side movement during normal handling or by the small, pyro-initiated rocking motion at separation. The rocking motion was first seen during review of pyro qualification test film. The bolts used in the qualification tests also exhibited similar local flat spots. (Note that the rocking motion was not sufficient to cause the bolt bending experienced on STS-34.) New mating procedures were developed to alleviate this problem. These procedures were formalized and will also be used for Orbiter-to-transporter mate. The STS-32/OV-102 ET attach/separation assembly was removed at Dryden and sent to RI/Downey for failure analysis. ***This anomaly was resolved for STS-36.***
15	Pilot seat down drive motor did not operate. IFA No. STS-32-27 HR No. ORBI-256C ORBI-340 ***There were no seat brake/motor anomalies on STS-36.***	During preparation for descent, Pilot Wetherbee attempted to make seat adjustments. The seat would drive up, but not in the down direction. Repeated attempts to lower the seat failed. The forward and back drive was not used or tested. The most probable cause of this failure was a defective down limit switch. Ground test showed that the seat was operating nominally. Analysis of the removed limit switch will be performed. The flight effect was crew inconvenience if seat height cannot be adjusted. ***This anomaly was not a safety concern for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
16	Backup Flight Computer (BFC) General Purpose Computer (GPC) errors - Input/Output (I/O) Terminal B. IFA No. STS-32-22 HR No. ORBI-066 ORBI-334 ***There were no GPC anomalies on STS-36.***	GPC #5, where the Backup Flight System (BFS) software was resident, registered numerous GPC error code 41s (illegal engage/I/O term B). Real-time data analysis indicated that the GPC #5 "Term-B" discrete was toggling. If the GPC #5 discrete is toggling or fails hard "0", the backup BFS/GPC cannot gain control of the 8 flight buses. The error was the result of the BFS detecting no I/O terminate B discrete when the engage discretes are not present. The error was logged approximately 43 times before the GPC was halted. As a result, the BFS was moved from GPC #5 to GPC #2 and reinitialized. This left STS-32 with 3 primary and 1 backup GPC for entry and reduced fault tolerance to a single failure. The BFS software is normally loaded into GPC #4 in the event that GPC #5 is determined bad. Because of the preflight concern with Kemet capacitors in the GPC #4 Input-Output Processor (IOP), it was decided before launch to use GPC #2 as the alternate BFC. KSC was able to recreate the problem; however, when the Breakout Boxes (BOBs) were installed the problem did not recur. The BFC connector J4 and IOP connector J5 were inspected. Power was cycled to the GPC, the GPC output switch was moved, wires were wiggled, and the problem would not repeat. Additional troubleshooting proved to be nonproductive. BFC #2 and the IOP #5 were removed and replaced. A confidence test was completed on IOP #5 input receiver (DI 13). If this anomaly occurred prior to launch, the launch would have been held until the cause of the problem was determined, or the launch would have been scrubbed if it was determined that the BFC/GPC was bad.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
16 (Continued)		To date, there have been several IOP circuitry failures; however, there were no failures involving erroneous I/O terminate discretes. None of the IOP circuitry failures were determined to be generic in nature. There were 2 BFC transmit circuitry failures identified involving I/O terminate discretes. Both failures occurred during acceptance testing, and both are still under investigation. Rationale for STS-36 flight was: • Launch would occur only if there was a good 5-GPC set. • There is built-in failure tolerance and redundancy (2-fault tolerant). • Proven crew workaround procedures were in place. • GPC switch or cabling would fail open ("0"); therefore, they were not suspect prior to launch since the problem identified failed high ("1"). ***This risk was acceptable for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1	Main Combustion Chamber (MCC) aft end debond found on engine #2022. IFA No. STS-32-ME-01 HR No. ME-B5 (All Phases) ***There were no MCC debonds on STS-36 SSMEs.***	On engine #2022, a 5/64" diameter MCC debond was detected post STS-32 flight by ultrasonic inspection. It was located 1/2" from the edge and in line with nozzle tube #664. An aft region internal fuel leak is a criticality 1 failure and is assumed to be rapid and extensive, resulting in High-Pressure Fuel Turbopump (HPFTP) cavitation and Liquid Oxygen (LOX) rich operation. The test history of this engine included 16 starts and 4650 sec. The debond was limited to an area between adjacent feed slots in line with nozzle tube #664 (2 affected channels). Postflight leak checks verified no leak at the bond line. There was no fabrication or assembly history found which was indicative of a problem. The failure was consistent with previous bond line failure assessment. The debond initiated at the aft end of the feed slots, resulting most likely from an undetectable flaw or marginal bond in this region. The defect could then propagate as a result of start/shutdown transients (highest strain to bond line). A proof test screens gross bond deficiencies. Also post-proof ultrasonic inspection detects debonds. Current data on this type of condition indicates that the propagation rate is slow and stable; there is a low probability of a massive bond line failure. The MCC was returned to Canoga for rebuild/repair prior to reuse. Engine #2022 was replaced by engine #2012. (The same type failure was experienced on engine #2031, post STS-29 flight. See STS-30 Postflight MSE, Section 5 - STS-29 Inflight Anomalies, SSME 1 for additional related information.)

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u> 1 (Continued)		Rationale for STS-36 flight was: • OV-104 units passed leak test. • OV-104 units had no fabrication discrepancies or proof test debond indicative of a marginal bond. • Ultrasonic inspection of OV-104 MCCs showed no disbond. • Any unacceptable leakage would have been detected by the HPOTP turbine discharge temperature redline, and provides a safe engine shutdown. • There is a low probability of massive bond line rupture. – 7600-pounds per square inch gage (psig) proof test detects gross bond deficiency. – There has been no premature cutoff with disbond (3 units/4 leaks). – Prior leaks were undetectable in engine performance. • Disbond propagation rate is slow based on data from engine #2108/MCC #2007. There was no detectable change after 500-sec test and no performance effect after 3 tests/903 sec. *This anomaly was resolved for STS-36.*

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
2	Gouges found in the MCC throat area of engines #2024 and #2028. IFA No. STS-32-ME-02 STS-32-ME-03 HR No. ME-B5 (All Phases) ***There were no gouges reported from postflight inspection of STS-36 SSMEs.***	Gouges were found in the throat area of the MCC on engines #2024 and #2028. It was thought that these gouges were introduced when the leak check throat plug was installed prior to flight. These gouges should have been found prior to flight and polished-out. Imperfections of this type in the MCC throat area could cause localized hot spots leading to burnthrough. The gouge found on engine #2028 measured approximately 2" long by 0.080" wide by 0.009", deep with some raised metal. The gouge was caused by the engine horizontal installer during removal/installation after STS-28. The engine #2028 MCC liner was repaired using a cell plating process to deposit copper in the gouge area. A NASA/contractor team was formed to revise procedures or modify equipment to eliminate or minimize engine handling damage. In addition, the launch and landing site personnel were counseled on the importance of hardware inspections. The MCC gouge on engine #2024 was noted 6" out from the throat area at the 6-o'clock position. The gouge measured approximately 0.25" long by 0.24" wide by 0.10" deep. The gouge was caused by a "B" nut which is tethered to the upper throat plug. The "B" nut is used to cap the upper throat plug bleed valve. Investigation using an MCC proof-load test article and an upper throat plug found that the "B" nut swinging on its tether could inflict gouges of the dimensions noted. The engine #2024 MCC liner was repaired by reducing the stress concentration in the area of the gouge. ***This anomaly was resolved for STS-36.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Upper strut Ethylene Propylene Diene Monomer (EPDM) cover partially missing. IFA No. STS-32-B-01 HR No. B-30-06 Rev. C C-60-03 Rev. B ***There was no report of missing EPDM on STS-36 Solid Rocket Boosters (SRBs).***	During postflight inspection at Kennedy Space Center (KSC), both left and right SRBs were missing some EPDM and Room-Temperature Vulcanizate (RTV) Q3-6077 materials from the upper strut location. The upper strut EPDM was partially missing, and the unprotected areas showed heat effects. A 5" section of EPDM cover was missing on the RH side, and a 4" section of EPDM cover was missing on the Left-Hand (LH) aft sides of the upper struts. The Q3-6077 high-temperature silicone that covers and protects the PR-855 foam from heat damage was missing below the lost EPDM rubber on both the RH and LH struts. The PR-855 foam showed heat effects on both the LH and RH struts. Specific heat effects on the RH SRB included: 2 cables (A-bus power and upper strut firing line) were found with heat discoloration on the outer YR-364 tape; 5 sealant caps and PR 1422 were eroded. Thermal analysis indicated that the YR-364 tape should protect the cables. It is possible that damage was caused by aeroheating during descent. The previous worst-case damage was a 3" tear in the EPDM cover on STS-27 and a small piece (1/2" x 1/2") missing on STS-28. No previous heat effects were found on the cables. The edges of the EPDM covers are typically charred and frayed. The areas under investigation included: evaluation of the EPDM bond line, evaluation of the Q3-6077 failure mode, analysis of the heat damage to the PR-855, evaluation of the extent of the heating effects on the cables, and design evaluation of the upper strut closeout.

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1 (Continued)		Rationale for STS-36 flight was: • Thermal protection of cables through ascent is assured by blast tape; the Thermal Protection System (TPS) provides additional protection. • Cables are routed in different bundles in the same area. ***This anomaly was not a safety concern for STS-36.***
2	External Tank Attachment (ETA) ring aft Instrument and Electronics Assembly (IEA) cover sooted. IFA No. STS-32-B-02 HR No. B-60-24 Rev. C ***There were no similar signs of sooting on STS-36 SRBs.***	The ETA ring aft IEA middle cover was sooted on the aft inside surface. The function of the IEA was not affected. The minor sooting was confined to a small area on the cover. The problem was attributed to the installation of "larger" Hi-Lok fasteners, preventing proper fit of the cover. As a result of an inspection of this ETA ring following STS-29, several oversized holes were drilled to accommodate 5/16" Hi-Lok fasteners in place of the normally-used 1/4" size. Since the larger fasteners were longer in length, the fasteners protruded and held the cover up approximately 0.1", thus allowing a hot-gas path into this area. Build paper on all ETA rings was checked to verify proper installation. This condition was peculiar to the STS-32 ETA ring. The ring was returned to its proper configuration prior to reuse. ***This anomaly was not a safety concern for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
3	Broken fastener found on STS-32 LH SRB upper strut fairing. IFA No. STS-32-M-03 HR No. INTG-081A INTG-134A ***There were no missing or broken upper strut fairing fasteners found on STS-36 SRBs.***	During postflight disassembly of the STS-32 SRBs, a broken fastener was found in the LH upper strut fairing or "milk-can". Proper fastener material properties and heat treatment were confirmed by analysis of the failed fastener. Material analysis also concluded that the failure was due to torsional overload. This conclusion led to the determination that the overtorquing occurred prior to launch, based on the fact that there are no torsional forces exerted on this fastener during flight. Water impact loading would have resulted in shear failure. There were no problems reported during the installation of STS-36 upper struts. ***This risk factor was resolved for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Right SRM Safe and Arm (S&A) gasket depression on secondary seal. IFA No. STS-32-M-01 HR No. BC-03 Rev. B BI-02 Rev.B ***There were no depressions found in any gasket removed from STS-36 SRMs.***	During postflight inspection of the right SRM S&A gasket, a small depression was found in the crown of the secondary seal aft face. The crown of the right SRM S&A gasket secondary seal was depressed inward at the 0° location. The depression measured approximately 0.050" circumferentially by 0.026" radially by 0.0025" deep. This gasket was previously flown on STS-26R (RSRM-1); however, no anomaly was detected during the STS-26 postflight inspection since the gasket was not inspected within 1/2 hr after removal from the joint. The gasket was later inspected in accordance with the old gasket inspection requirements for reuse on the STS-32 mission. An additional inspection of this seal was performed when an igniter seal void was discovered for the STS-28 mission. This supplemental gasket inspection requires a 3-hr compression test in a plexiglas fixture, with a post-compression touch inspection within 1/2 hr after removal from the fixture. No defects were found. This procedure was documented in the latest release of STW7-2790 (Rev D) for gaskets to be reused, which meets the definition of a "used" gasket. This gasket did not meet the criteria of a used gasket since it was not touch inspected for approximately 20 hr after STS-26 disassembly. Review of the igniter Gask-O-Seals installed on STS-36 SRMs found the following: • The LH inner gasket S/N 2204R1 was last used on STS-26. It was inspected after disassembly within the required 30 min. Prior to installation on the STS-36 SRM, it successfully passed the post-compression touch test after 3 hr of compression under plexiglas. • The RH inner gasket S/N 090 was new. It successfully passed the post-compression touch test after a 3-day compression under plexiglas.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)		• Both the LH and RH outer gasket S/Ns 089 and 094 were new. Both successfully passed the post-compression touch test after a 3-day compression under plexiglas. • The LH S&A gasket S/N 6801 was previously used. It was not inspected after disassembly. Additionally, it was not compressed under plexiglas per the new procedures. • The RH S&A gasket S/N 8202 was previously used. It was not inspected after the last disassembly. Additionally, it was not compressed under plexiglas per the new procedures. A decision was made to replace both S&A gaskets with Gask-O-Seals that were properly compressed and touch tested. Properly tested replacements were available at Thiokol, and the changeout was accomplished at the Pad on February 4, 1990. All gaskets passed post-installation leak checks. The decision relative to the inner and outer igniter gaskets was not to require changeout because the gaskets successfully passed the post-compression touch test. The following corrective actions were implemented: • Reviewed pedigrees of all gaskets installed on flight and test motors. • Replaced gaskets having no touch inspection performed within 1/2 hr after undergoing 3-day compression with gaskets that were properly touch-inspected.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)		• Created new S&A gasket dash numbers to preclude use of gaskets already accepted per the old requirements. • Changed the reuse inspection specification to require every gasket to have a documented 3-day compression with a touch test within 1/2 hr prior to each reuse. ***This risk factor was resolved for STS-36.***
2	Raised areas found on the igniter inner Gask-O-Seal. IFA No. STS-32-M-02 HR No. BC-03 Rev. B ***There were no similar Gask-O-Seal anomalies found on STS-36 SRMs. There was some localized damage to the Gask-O-Seal retainer as a result of a putty blowhole. A recent decision was made to disallow reuse of igniter Gask-O-Seals.***	In addition to the voids found in the S&A Gask-O-Seal on the STS-32 RH SRM, postflight inspection of the RH igniter inner Gask-O-Seal S/N 060 found small raised areas or bulges on the cushion and in the valleys of the seal. These raised areas were located intermittently around the full circumference of the seal. No raised areas were found on the seal crown and, therefore, did not affect the sealing function. These bulge types were not previously seen. S/N 060 was used twice previously, on an igniter qualification test and on the igniter used on TEM-3 (Test Evaluation Motor). It was put through reuse inspection prior to installation on the STS-32 RH SRM. Reuse inspection criteria do not allow for raised areas, such as those seen, but do allow delaminations up to 0.05" in length. According to records, S/N 060 passed all reuse inspection and test criteria. A team, represented by Marshall Space Flight Center (MSFC), Thiokol, and Parker Seal (vendor), evaluated S/N 060. Dissection of the seal indicated that the raised areas were present for some time, most probably since manufacture.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2 (Continued)		Rationale for the use of Gask-O-Seals on STS-36 included: • Raised areas in the locations found would not affect the sealing function; raised areas on the crown of the seal would. • Gask-O-Seals installed on STS-36 SRMs were visually inspected for sealabilty under compression, touch inspected, and passed all leak checks. ***This risk factor was resolved for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ET</u>		
1	Review of the ET separation photos from STS-32 showed 4 Spray-On Foam Insulation (SOFI) divots in the bipod area. IFA No. STS-32-T-01 HR No. INTG-008 INTG-037B INTG-081A ***Separation photographs were not taken because ET separation occurred in darkness.***	Postflight review of STS-32 ET separation photos found 4 SOFI divots just forward and underneath the bipod area. These divots were similar to those seen in STS-28 ET separation photos. On STS-28, divots were seen above the RH bipod spindle. Divots seen on STS-32/ET-32 were into the Isochem layer. Additionally, these divots had the same appearance as divots noted on flights before the implementation of intertank TPS vent hole modification. No Orbiter damage was attributed to the existence of these divots. Vent holes are drilled through the TPS in the intertank. It was thought that, due to tolerance stack up of the TPS (i.e., thicker areas of TPS), vent holes were not drilled to the proper depth. Additionally, a review found that the vent holes may not have been drilled at the proper angle. Rationale for flight of STS-36/ET-38 was that the vent holes on the intertank were verified for proper depth. ***This risk factor was resolved for STS-36.***

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STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
KSC		
1	RH aft Integrated Electronics Assembly (IEA) bent pins. IFA No. STS-32-K-03 HR No. B-00-17 Rev. B ***There were no similar connector pin anomalies discovered during STS-36 postflight inspection.***	During postflight assessment, a connector in the RH aft IEA was found to have 2 bent pins. Pin #64, which is wired as a spare, was bent 90° flat and was nearly touching pin #58. Pin #66, which was a wired shield, was bent 180° into a hook shape. Since these pin are wired spares, they are not checked out during final functional testing after final mating of the cables to the IEA. The pins were bent during mating of the cable to the IEA. Adjacent pins in the connector control vital Thrust Vector Control (TVC) functions. Rationale for STS-36 flight was: • The File II and File V system check verified every functional path within the IEA after final connector mating. • By design, redundant functions were not routed through the same connector. • Adjacent pins, if shorted together, would not cause loss of a critical function. • Operations and Maintenance Requirements Specification (OMRS) system check verified every functional path within all circuits. ***This anomaly was resolved for STS-36.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>MCC</u>		
1	State vector uplink incident. IFA No. STS-32-MOD-01 HR No. ORBI-066 ***There were no state vector uplink problems during STS-36.***	An off-nominal maneuver was observed in all 3 axes during the crew sleep period on FD 9. The vehicle began to roll nose-to-tail following a state vector update that was later determined to be erroneous. The vehicle roll rates were arrested by manual intervention by the crew. A new state vector was uplinked later to allow the crew to reselect the auto-track mode. An investigation team was established to determine the cause of this incident and to make recommendations for corrective action to prevent future occurrences. The findings indicated that the state vector uplink incident was caused by an operator error. The erroneous operator procedural response was clearly outside the "trained to" and commonly expected procedures for this scenario. The basic system design and procedures associated with every aspect of this incident are mature and sound. The ground system and onboard system hardware and software worked as designed. Based on the investigation team findings, the following was implemented for STS-36 and subsequent missions in order to preclude recurrence: • The Integrated Communications Officer (INCO) console handbook was updated to add cautionary notes prohibiting the use of manual execute (override) in the presence of an indicated data reject message. • Handshakes between uplink owners and the INCO console were required when passing direction cosine matrix numbers. • Handshakes between the INCO console and the INCO backroom console were required for all critical commands and command troubleshooting.

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
MCC		
1 (Continued)		• The command load owner and flight director were notified of all manual loads or line-by-line corrections. Command load owner and flight director concurrence was required. Additional long-term corrective actions are under consideration. ***This anomaly was resolved for STS-36.***

SECTION 6

STS-34 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-34 mission. Each anomaly is briefly described, and risk acceptance information and rationale are provided.

SECTION 6 INDEX

STS-34 INFLIGHT ANOMALIES

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Engine Interface Unit (EIU) #3 momentary 60-kilobit (Kbit) data stream loss. IFA No. STS-34-02 HR No. INTG-021A INTG-065 INTG-072 INTG-165 ***There were no EIU failures on STS-36.***	EIU #3 Built-In Test Equipment (BITE) bit #13 set and the 60-Kbit data stream were lost, both momentarily. Recurrence would have resulted in loss of 60-Kbit data. The problem did not recur during this mission. Redundancy exists in the EIU for critical command and data paths; however, the BITE is not flight critical. The worst-case effect is loss of Space Shuttle Main Engine (SSME) performance data to the Operational Instrumentation (OI) recorder; the data is not mission essential. The 60-Kbit data is used to confirm critical Launch Commit Criteria (LCC) for Liquid Oxygen (LO_2) dome temperature and ice detection. The most probable cause was BITE circuitry failure. All 3 EIUs were replaced with modified EIUs during the STS-36 flow. ***Not a safety concern for STS-36.***
2	Auxiliary Power Unit (APU) #1 fault to high speed. IFA No. STS-34-04 HR No. ORBI-031 ***There were no APU controller failures on STS-36. There was, however, an APU #1 flex hose leak (IFA No. STS-36-08) and a hydraulic system depressurization anomaly (IFA No. STS-36-17) on STS-36. There is no indication that these anomalies are related.***	APU #1 experienced an uncommanded speed shift to the high-speed band at L+2.5 minutes (min) during ascent. This speed shift was intermittent over a 4- to 5-second (sec) period and was permanent thereafter. The crew commanded the APU to high speed 15 sec after the uncommanded shift to avoid alarms. The APU operated satisfactorily at high speed for the remainder of ascent. Troubleshooting found that the speed shift was due to the pulse control valve failing open. This valve varies the fuel flow to the APU to regulate speed. After failure of the pulse control valve, the shutoff valve was used to control APU speed. APU #1 was not turned on until Mach 10 and was turned off at postlanding wheel stop. APU #1 Gas Generator Valve Module (GGVM) was sniffed checked, and insulation connectors were visually inspected; there was no indication of propellant leakage and no obvious external GGVM deformity.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)		The APU controller was sent to Sundstrand for troubleshooting. During testing in an ambient temperature environment of 110°F, Sundstrand was able to duplicate this anomaly twice. The controller was left at ambient room temperature overnight and then returned to a temperature of 110°F. At the higher temperature, Sundstrand test conductors repeated the anomalous condition 5 additional times. The pulse code card in this controller was found to be susceptible to high thermal environments. The problem was isolated by Raytheon to an intermittent open 2N2222 transistor, Lot Date Code (LDC) 8131, in the pulse control circuit within APU controller Serial Number (S/N) 311. This was the first Orbiter use of this controller. The Line Replaceable Unit (LRU) criticality is 1R2. Analysis of the failed transistor indicated that circuit failure to open was caused by depletion (thinning) of the gold wire at the transistor base junction due to "purple plague". Cross-sectional analysis showed extensive voiding between the bond wire and the pad. Five transistors on the same circuit board, with the same part number and LDC as the failed part, passed electrical tests and were subsequently destructively analyzed. All passed nondestructive bond pull tests; 2 parts were destructively pull tested with good results. Two transistors were cross-sectioned and showed excellent bond-to-pad junctions. Analysis of 4 additional transistors from the Solid Rocket Booster (SRB) program, with the same part number and LDC as the failed part, resulted in all findings being normal. The rationale for flight was based on the fact that an affected APU can operate in high speed while the other APUs operate normally. The increased risk due to APU overspeed was acceptable. There was no evidence to indicate that this was a generic failure problem. *Not a safety concern for STS-36.*

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3	Multiplexer-Demultiplexer (MDM) Flight-Critical Aft (FA) #1 Input/Output (I/O) errors. IFA No. STS-34-05 HR No. ORBI-038 ***There were no MDM failures on STS-36.***	At L+35 min, MDM FA #1 failure was detected by both the Primary Avionics Software System (PASS) and Backup Flight System (BFS) just prior to Orbital Maneuvering System (OMS) #2 burn. There was no response from the MDM primary port for 2 consecutive return word commands. I/O reset alone did not restore communications with MDM FA #1. Power cycling and I/O reset temporarily restored communications. The problem recurred and was restored by cycling power a few times. The crew was able to recover FA #1 operation for the remainder of the mission by moding to the backup port. Loss of the MDM FA primary port was acceptable during flight due to redundancy. Postflight troubleshooting at Dryden prior to MDM powerdown confirmed that port #1 was not communicating. The MDM was removed and replaced at Kennedy Space Center (KSC). While there were 33 instances of loss of 1 MDM port, MDMs have 2 ports and re-porting can be accomplished. Both ports must fail in order to lose an MDM. Loss of 1 port prior to launch is acceptable. Certain critical MDMs are themselves redundant; therefore, loss of an MDM FA results in a minimum duration flight. Loss of a flight-critical MDM is an exception to the single-fault tolerant rule because loss of a single MDM reduces redundancy in multiple systems. This increases the risk that a single lowest-replaceable-unit failure in any 1 of several systems could put the Orbiter at a zero-fault tolerance level. Loss of 2 FA MDMs results in a next Primary Landing Site (PLS) mission termination. Rationale for flight of STS-36 was based on the acceptability of a loss of any single MDM FA port (internal redundancy) and the loss of 1 FA MDM (external redundancy and minimum duration flight). ***This anomaly was resolved for STS-36.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	APU #2 Gas Generator (GG)/Fuel Pump (FP) heater "A" inoperative. IFA No. STS-34-06 HR No. ORBI-250 ***There were no APU heater failures on STS-36.***	APU #2 FP/GGVM system "A" heaters did not respond when selected. System "B" heaters were selected and operated acceptably; the "B" heater was cycling high. (See Orbiter 8 below.). Postflight testing at Dryden indicated that the "A" heaters were operating properly. Thermostat S27A was removed and replaced; retest was successful. The thermostat worked properly during vacuum testing at Johnson Space Center (JSC). It was sent to Sundstrand, and testing/troubleshooting could not duplicate the problem. APU #2 heater system was thoroughly examined. All single-point failure components/wiring that could have caused the anomaly were replaced. Retest was performed per Operations and Maintenance Instruction (OMI) V1019. The same problem occurred on STS-27, STS-30, and STS-34 missions. Had the problem recurred on STS-36, the "B" heater system would have been used. ***Not a safety concern for STS-36.***
5	Flash Evaporator System (FES) Hi-load inboard duct temperature low. IFA No. STS-34-07 HR No. ORBI-276B ***There were no FES temperature anomalies on STS-36.***	During ascent, post-Main Engine Cutoff (MECO), the Hi-load inboard duct temperature was observed to be lower than expected. The flash evaporator is the primary heat sink during ascent, initialized at approximately 140,000 feet (ft) by avionics software. Both heaters were enabled on the Hi-load duct. Approximately 3 min later, the crew shut down FES primary "A" and switched to secondary; the temperature continued to decrease. The system stabilized under radiator flow. Heaters were left on for bakeout. On Flight Day (FD) 2, the topping FES functioned properly on primary "A" and primary "B" controllers.

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)		Data analysis indicated that the lower than expected duct temperature was created by high heat load and transients induced by the Radioisotope Thermoelectric Generator (RTG) cooling loop. RTGs will not fly again until STS-41. ***Not a safety concern for STS-36.***
6	APU #3 seal leak into drain bottle. IFA No. STS-34-08 HR No. ORBI-100 ***This anomaly did not repeat on STS-36.***	APU #3 cavity seal drain line pressure increased, and fuel pump inlet pressure decreased. The possible cause of this anomaly was a leak in the static seal. The drain bottle was drained and checked at KSC to determine if the seal leak had degraded. The seal had not degraded; catch bottle quantity was within acceptable limits. The same anomaly occurred on STS-30. The STS-30 drain bottle contained 30 cubic centimeter (cc) of propellant. This anomaly is unique for OV-104, APU #3. No leaks have been experienced an OV-103 or OV-102. ***Not a safety concern for STS-36.***
7	Right OMS engine cover heater system "B" failed off. IFA No. STS-34-09 HR No. ORBI-120 ***There were no OMS heater anomalies on STS-36.***	During heater configuration to "B" heaters, the Aft Propulsion System (APS) right pod (RP03) "B" heaters failed to activate. The pod was removed. Investigation found a recessed pin in the heater connector. The connector was replaced. ***Not a safety concern for STS-36.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8	APU #2 fuel pump heater "B" cycling high. IFA No. STS-34-10 HR No. ORBI-250 ***There were no similar anomalies on STS-36.***	APU fuel pump heater "B" cycled erratically toward higher temperatures. This anomaly was possibly related to the failure of APU #2 fuel pump heater "A". (See Orbiter 4 above.) Thermostat S27B was removed and replaced; retest was successful. The thermostat was returned to Sundstrand for failure analysis. There was no indication of improper APU heater operation on OV-103 or OV-102. ***Not a safety concern for STS-36.***
9	Cryogenic Oxygen (O_2) manifold #2 isolation valve did not close. IFA No. STS-34-12 HR No. ORBI-303A ***There were no similar anomalies on STS-36.***	The crew attempted to close the cryogenic O_2 manifold tank #2 valve on panel R-1 per the sleep configuration. The crew reported that they held the switch for 5 sec. There was no talkback. No switch discrete was received. The valve closed properly on the first troubleshooting step while on-orbit. Postflight, the valve opened properly. There is redundancy in the valves. Review of the valve design found that it will lose the closed indication when in the relief mode. Concern would be raised if the closed indications have not been received during Main Propulsion System (MPS) dump or during reentry. ***Not a safety concern for STS-36.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10	Right vent door #3 motor #1 operating on 2 phases. IFA No. STS-34-19 HR No. ORBI-178A ***There were no further problems reported on STS-36.***	During prelaunch and landing configuration of vent doors, the right vent door #3 motor #1 operated on 2 phases. This occurred 3 times in flight. Phase B was lost when the door opened, and phase C was lost when the door closed. This anomaly also occurred with the same door motor during the STS-30 turnaround flow. During that flow, the problem occurred twice in 50 cycles. The motor control assembly was removed and replaced. Troubleshooting at Rockwell International (RI)/Downey could not duplicate any suspect relay failure during 900 relay cycles. Postflight troubleshooting at KSC repeated the anomaly; 1 phase of the Power Drive Unit (PDU) was found open. The vent door PDU was replaced and retested satisfactorily. Vent door motors are redundant and will operate on 2 phases, as demonstrated during STS-34. ***Not a safety concern for STS-36.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11	External Tank (ET)/Orbiter Liquid Oxygen (LOX) aft separation hole plugger failed. IFA No. STS-34-20 HR No. ORBI-302A ***There were no hole plugger failures on STS-36.***	The ET/Orbiter LOX aft separation hole plugger failed to extend fully by approximately 2" at ET separation. Postflight inspection found jamming by a detonator booster and detonator. A crushed backshell from the right aft connector was found on the runway after the ET umbilical door was opened. There is concern that loose debris could block the ET umbilical door from closing, resulting in the possible loss of the vehicle during reentry. Rationale for flight of subsequent missions is based on the probability being remote of escaping fragments preventing ET umbilical door closure. The vehicle performs a maneuver at separation away from the ET and moves away from possible escaping debris prior to ET umbilical door closure. ***This risk factor was acceptable for STS-36.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	Main injector heat shield retainer ring segment failure on engine #2029. IFA No. STS-34-E-01 ***There were no similar anomalies on STS-36 SSMEs.***	Three segments of the retainer ring were found broken on STS-34 Main Engine (ME) #3 (engine #2029). One piece was still missing (as of the STS-36 Orbiter Rollout Review on January 10, 1989). Engine #2029 was taken out of flight service until the piece was found. A similar problem occurred on the next flight (STS-33) on ME #3, engine #2107. However, all pieces were found, and the engine was cleared for flight. This was not a constraint to Orbiter Processing Facility (OPF) rollout of STS-36. The MEs on STS-36/OV-104 included Engineering Change Proposal (ECP) 620 that changed the configuration of the main injector heat shield retainer ring to prevent the condition which caused the Inflight Anomalies (IFAs) on STS-34 and STS-33. ***This risk factor was acceptable for STS-36.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Right SRB Holddown Post (HDP) #2 broached and shoe lifted from Mobile Launch Platform (MLP) during liftoff. IFA No. STS-34-B-01 HR No. INTG-164 B-00-15 B-00-17 *There were no HDP stud hangups on STS-36. Nearly all of the debris was contained.*	The holddown stud at HDP #2 (Right-Hand (RH) tension post) hung up at liftoff. This resulted in broaching of the right SRB aft skirt and thread impressions at that HDP bore. Review of the liftoff photographs found that the shoe on the MLP at HDP #2 lifted 2-1/4" at the same time. In addition, thread imprints were noted on 7 of the 8 SRB HDP feet during postflight evaluation. Analysis by United Space Boosters, Inc. (USBI) indicated that vehicle launch performance would not be affected if all 8 studs hung up, provided that the frangible nuts were released. Stud hangups were recorded on 5 previous flights (STS-2, STS-4, STS-51I, STS-51J, and STS-61A). Major broaching of aft skirt HDPs was experienced on 4 prior flights. Minor broaching and thread impressions were recorded on 46 HDPs on 10 previous flights. Lifting of MLP holddown shoes was seen on STS-2 and STS-29. Marshall Space Flight Center (MSFC) organized a tiger team to investigate stud hangups and the influences of recent design modifications on the holddown Debris Containment Systems (DCSs). Their review of liftoff films found similarities between the STS-34 occurrence and previous launches with stud hangups. Review of build papers relating to HDP installations revealed no anomalies. Frangible attach stud modifications in the debris containment device implemented prior to STS-28 should provide sufficient time for stud ejection. The modification reduced the holddown stud ejection velocity from 222 inches per second (in/sec) to 184 in/sec. This increased the time for stud ejection from 52 millisecond (msec) to 63 msec. With a 250-msec SRB liftoff time, 63 msec should be sufficient time for stud ejection.

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1 (Continued)		The tiger team also analyzed the effect of biasing of the MLP spherical bearings radially inward on the stud. Biasing of the spherical bearings was performed to increase the aft skirt Factor of Safety (FOS) for STS-34. The total HDP shoe/MLP spherical bearing mismatch for STS-34 was determined to be less than the mismatch on STS-27, STS-28, and STS-30. Compressive load on HDP #2 at frangible nut detonation was sufficient to prevent aft skirt shoe motion. The tiger team investigation determined that there was no evidence that biasing of the spherical bearings contributed to the stud hangup. MSFC analysis indicated that vehicle liftoff would be unaffected even if hangups occurred at all 8 HDPs, provided that all frangible nuts separated properly. An RI analysis conducted in conjunction with MSFC concluded that 1 or 2 stud hangups will not adversely affect vehicle liftoff dynamics or clearances between the vehicle and facility. The RI evaluation also concluded that the spherical bearing/shoe assembly will not break free and become a debris source. No broaching was experienced on STS-32. However, the DCS only contained an average of 57% of the total debris produced. ***This anomaly was resolved for STS-36.***

SECTION 7

STS-36 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the OV-104/STS-36 mission.

SECTION 7 INDEX

STS-36 INFLIGHT ANOMALIES

SECTION 7 INDEX - CONTINUED

STS-36 INFLIGHT ANOMALIES

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Engine #2027, nozzle #2027 bluing. IFA No. STS-36-I-01 HR No. ME-B7 (All Phases)	During STS-36 postflight inspection, approximately 3" of bluing was noted on nozzle #2027 aft manifold adjacent to the High-Pressure Oxidizer Turbopump (HPOTP) primary drain exit line. This bluing was similar to that seen on STS-33 (IFA No. STS-33-I-01) and was believed to be due to a new nozzle phenomenon. On STS-33, the nozzles on both engines #2031 and #2107 were new; however, only the nozzle on engine #2107 showed bluing. Rocketdyne analysis found, through Rockwell hardness tests, that there was no annealing in the area of the discoloration. They approved the nozzle for further flight use. Causes resulting from contamination, ascent heating, improper material properties, and flight profile were ruled out. The most probable cause of the bluing was descent heating during a steep reentry profile. Both STS-33 and STS-36 were Department of Defense (DoD) missions with high inclinations (actual inclinations are classified).

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Fuel Cell (FC) #2 Alternating Current (AC) phase "A" inverter failure. IFA No. STS-36-01 HR No. ORBI-127A	During the first launch attempt, FC #2 AC phase "A" inverter had numerous voltage and current fluctuations in a 2-minute (min) period. Fluctuations were from 112 volts AC (VAC) to 122.8 VAC; Operational Maintenance Requirements and Specifications Document (OMRSD) limit is 110-120 VAC. Inverter #4, Serial Number (S/N) 51, was removed from avionics bay #2 and replaced with S/N 42. FC #2 was retested satisfactorily prior to the next launch attempt. S/N 51 was returned to the vendor for failure analysis. The vendor was able to repeat the failure mode. The problem was isolated to loose connections within the inverter. Four screws were found improperly torqued and loose. Inspection and retorque of 5 other suspect inverters were completed.
2	Liquid Hydrogen (LH_2) 17" disconnect "B" open indication intermittent. IFA No. STS-36-02 HR No. ORBI-306	The LH_2 17" disconnect open indication dropped out for approximately 12 seconds (sec) during fast fill. The ground launch sequencer software issued an LH_2 stop fill command. No explanation was found for the dropout. Because the Launch Commit Criteria (LCC) requires only 1 of 2 indications and the "A" indication was good, the LH_2 fast fill was resumed. The indication was normal for the remainder of the launch preparations.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Reaction Control System (RCS) Thruster R3D failed off during External Tank (ET) separation. IFA No. STS-36-04 HR No. INTG-172	Chamber pressure in thruster R3D S/N 228 did not reach the required pressure within the specified time period; therefore, the Redundancy Management (RM) system deselected the thruster. This failure occurred during ET separation and is a Crit 1R/3 failure (there are 3 redundant, down-firing thrusters on each Orbital Maneuvering System (OMS) pod). Previous experience indicated that this failure was due to the oxidizer valve poppet not opening. Contamination, in the form of a "varnish" type deposit of nitrates, was found on oxidizer valve poppets of other failed thrusters. This contamination is created when the oxidizer (nitrogen tetroxide (N_2O_4)) comes in contact with moisture. It was previously noted that the rain cover was found missing from R3D prior to launch. A puddle of water was found in the R3D thruster throat and was successfully educed. The presence of water could be a contributor to R3D failure. The thruster was removed and sent to the vendor for failure evaluation. Visual inspection at Dryden found no contamination in the thruster throat. Failure analysis at Marquardt determined that nitrates were formed on the oxidizer valve poppet, preventing it from opening in the allotted time.
4	Right RCS manifold #1 oxidizer isolation valve position indication intermittent. IFA No. STS-36-06A HR No. ORBI-244	At L–4 sec, the right RCS aft oxidizer manifold #1 isolation valve open indication changed to not open. This mission indication caused RCS RM software to annunciate an "RM DLMA MANF" message and to override the right RCS manifold #1 closed indication at Solid Rocket Booster (SRB) separation. The right RCS manifold #1 was overridden to open via the override display after ET separation. The open indication returned when the crew moved the right RCS manifold #1 switch from "GPC" to "OPEN" after the OMS-2 burn. Troubleshooting at Kennedy Space Center (KSC) found no wiring anomalies. The isolation valve actuator was changed out.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	Left RCS 3/4/5 "B" oxidizer tank isolation valve open position indication intermittent. IFA No. STS-36-6B HR No. ORBI-244	At L−7 sec, the left RCS 3/4/5 "B" oxidizer tank isolation valve open indication changed to not open; the indication returned to open during a 2-sec period. The valve position indication worked well for the remainder of the mission. This failure was not representative of previous contaminated switch problems that occurred on-orbit and did not clear. This could be due to an erroneous data problem or loss of telemetry.
6	Left RCS 1/2 oxidizer crossfeed valve closed position indication intermittent. IFA No. STS-36-6C HR No. ORBI-244	At L−45 sec, the left RCS 1/2 oxidizer crossfeed valve closed indication changed to not closed, identifying the loss of closed position. This loss-of-position indication occurred during a high-vibration period. The closed indication returned when the crew performed the post-OMS-2 burn reconfiguration, moving the left RCS 1/2 crossfeed switch from "GPS" to "CLOSED". This failure was not representative of previous contaminated switch problems that only occurred on-orbit and did not clear. A potential failure mode was that the Launch Process Sequencer (LPS) command could equal the maximum valve travel time (1.3 sec); therefore, the valve was potentially not driven into the stops/detente. Corrective action would be to increase the LPS command to 2.0 sec.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	Water Spray Boiler (WSB) #2 vent system "A" heater failed. IFA No. STS-36-07 HR No. ORBI-244	WSB #2 vent system heater "A" began to degrade about 75 min after initial activation. Heater "B" was selected and operated nominally. Heater "A" was reselected for reentry, was slow to come up, and operated erratically during reentry. This was a repeat of IFA No. STS-34-18, which was closed as unexplained because it could not be repeated. Troubleshooting for the STS-34 anomaly included operating the "A" and "B" heaters for a number of cycles and shaking the wiring and connectors. Nozzle heater troubleshooting showed nominal operation. The WSB controller was removed and sent to the vendor for failure analysis. Testing at the vendor failed to duplicate the anomaly. Alternate heater activation sequences are being evaluated to obtain additional flight data.
8	STS-36 Auxiliary Power Unit (APU) #1 hydraulic flex hose anomaly. IFA No. STS-36-08 HR No. ORBI-036 ORBI-047A ORBI-184 ORBI-188	Hydraulic system #1 exhibited anomalous indications during STS-36 ascent. The reservoir pressure dropped as expected after SRB ignition, but unexpectedly continued to drop. The reservoir temperature increased as expected, but the 6% volume increase due to temperature rise did not materialize. Early during reentry, flight controllers concluded that hydraulic system #1 was leaking at a rate that could deplete the fluid and cause system #1 shutdown prior to landing. Controllers directed the crew to temporarily select "low pressure" on system #1 in order to reduce the leakage rate in an attempt to assure system #1 availability during the approach and landing at Dryden Flight Research Center (DFRC). The action was successful, but the reservoir volume had decreased to 27% by the early postlanding APU shutdown. Postflight inspection of the aft compartment found hydraulic fluid sprayed over most of the aft compartment components. Hydraulic leakage was tracked to a high-pressure flex hose S/N 153 that was removed and sent to Rockwell International

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		(RI) for failure analysis. The flex hose, Part Number (P/N) ME271-0079-1129, is certified to MIL-H-38360A. It has a 0.045"-thick teflon liner surrounded by 2 braids of stainless steel and covered by a 1/8" rubber chaffing strip. It is 42" long with an inner diameter of 1". Installation is supported by a saddle with a slight bend. This type of flex hose is proof-pressure tested at 6000 pounds per square inch (psi). A 1/2" split was initially found in the stainless steel braid at the leak site. This cut indicated that the leak was not caused by external forces (i.e., twisting). Failure analysis at RI found a pin hole in the teflon liner, 19" from the swage fitting. Additionally, 2 kinks found on the teflon liner about 1 1/2" from the pin hole were considered contributors to the leak. X-ray inspection found no cracks in the metal end fitting. RI performed a pneumatic test of the flex hose, submerged in water, to verify the leak. Subsequent to leak verification, an extensive teardown was performed. Titeflex, the vendor, assisted RI with the failure analysis. Similar flex hoses are used in NASA, DoD, and commercial aircraft applications. A review of flex hose applications found no other similar failures of high-pressure flex hoses in any application. All leaks were found to have originated at the hose fitting end(s). Three flex hose leaks occurred in Orbiter Program applications. All leaks were in fittings, not in the liner. The first was in August 1975, where a flex hose failed proof test due to a thin fitting wall. The second was experienced in November 1979; a fitting leak indication was found, but further examination and subsequent use found no repeated problem. The third was found during postflight inspection of STS-1 hydraulic systems. A crack was found in a fitting that previously passed proof tests. Subsequent failure analysis found that this surface crack did not leak or degrade the performance of the fitting.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		The failed hose was initially installed on OV-104 during original production at Palmdale. High-pressure flex hoses on OV-104 have experienced the least operating time of all Orbiter vehicles, approximately 9 hour (hr). OV-103 flex hoses have the most operating time, approximately 14 hr. The actual leak originated at the center of the flex hose, approximately 16" from the crimp sleeve on the elbow end. External leakage was through a longitudinal split in the chafe guard, occurring approximately 12" from the leak in the teflon liner. The area around the split appeared as a blister in the extrusion skin. An area of permanent deformation, or kink, was also observed approximately 1" from the leak site. Microscopic examination of the leak site confirmed that the leak was at a longitudinal surface crack, originating in the inner diameter of the liner. A semi-elliptical shaped flaw, 0.180" x 0.036", had grown and broken through the teflon liner to a leak site of 0.020". X-ray examination found no break or disturbance in the braid layers. Laboratory tests using fractography were performed on three 1/2" sections of the failed teflon liner. The sections were used as test coupons and subjected separately to monotonic loads with and without a 0.180" x 0.036" notch and to fatigue cycling. Results of fractographic analyses of these coupons, and comparison to the failed area of the flex hose, determined that the initial surface crack or flaw formation mechanism was unknown; it could not be conclusively tied to overload, fatigue, or sustained stress. In addition, the internal surface crack was extended by Low-Cycle Fatigue (LCF) from the pressure rippling effects associated with pump operational characteristics; this led to final flaw breakthrough and a stable leak.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8 (Continued)		In another test, a remnant portion of the failed hose was bent to a radius of approximately 5" to evaluate kinking and buckling characteristics. This radius produced 2 transverse buckles on the compression side and 1 longitudinal buckle on the tension side. Compression-side buckles were found to have a similar appearance as buckles found approximately 1" from the STS-36 flex hose leak site. Because longitudinal craze damage, or "whitening", was observed near all buckles of the failed flex hose liner, effects of flex hose pulsing on the craze area were also examined. This was initiated because of the theory that mishandling or unacceptable bending of the flex hoses had occurred and, with the pulsing effects induced by normal operation, led to the failure on STS-36. Fatigue tests were performed. After 1.3 million cycles, no cracks were found in the craze area. This represented only half the number of cycles in a single flight for one of these hoses; however, these tests were performed using a much higher load profile. Similar tests were performed with a teflon hose liner notched to simulate the surface flaw on the failed hose. Through the cycling of the hose, growth of this flaw or crack was achieved. ***The results of these tests led to the conclusion that a crack cannot occur as a result of bending or mishandling alone; a crack initiated by improper sintering or other flaws can grow as a result of mishandling and operational cycling.*** Chemical properties of the failed teflon liner were analyzed and compared to reference properties for unsintered teflon, minimally sintered teflon, and teflon after maximum sintering. These inputs were provided by duPont, the teflon manufacturer. Analyses of the failed teflon through infrared spectrophotometry and differential scanning calorimetry techniques – comparison to reference properties of the melting point, specific gravity, heat of transition, and tensile strength – determined that the failed teflon liner was minimally sintered.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		"Bump extrudate" occurs during manufacture after the liner is extruded but not cured. The liner may bump against the side of the extrusion machine. This results in a bump that, when cured, will not achieve proper temperature. The bump extrudate phenomenon causes minimum sintering, believed to be the originating factor leading to the STS-36 flex hose failure. This type of flaw should be caught during quality inspection testing after manufacture through a crush and roll test. The crush and roll test flattens the teflon liner under glass for observation and detection of similar flaws. The lot from which the failed STS-36 flex hose was made did not have a quality stamp, or buy-off, next to the crush and roll test on the buy-off sheet. All other required tests had the appropriate quality stamp of approval next to the test step. It is believed, therefore, that either the crush and roll test was not performed on this lot of teflon liner or an anomaly was discovered which the quality inspector could not accept. The results of the failure analysis and investigation were: • The leak resulted from a single crack in the teflon liner, which grew by fatigue from a surface flaw on the inner diameter. • The surface flaw formation and growth was facilitated by the combination of: – Local incomplete sintering in the flaw area, probably caused by a phenomenon called "bump extrudate". – Minimally complete sintering of the entire liner.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE

ORBITER

8 (Continued)

- Buckling of the hose, possibly caused by mishandling.
- Stressing (LCF) induced by the operational environment.

- It is possible that a test was omitted which would have detected the lack of sintering.
- There is a low probability that the combination of the factors which led to the STS-36 flex hose failure will be repeated.
- There was no indication that this is a generic teflon liner problem.

The Johnson Space Center (JSC) Safety Division researched the concern for fire resulting from a hydraulic fluid leak into the aft compartment. Two locations on each APU exceeded the autoignition point of MIL-H-83282A hydraulic fluid used in the Orbiter. These were the injector well (1200°F) and the interface area of exhaust duct to APU housing (1100°F). These surfaces are covered with insulation and stainless steel foil, except the injector well which has Kao-wool insulation that is also a liquid barrier. Except for "smart leaks", it is not credible for the hydraulic fluid to come in contact with the high-temperature areas.

The number and location of hydraulic fluid autoignition temperature exceedences on the main engines have not been completely catalogued to date. There are several which may approach or exceed the autoignition temperature and are not insulated or isolated.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		Tests performed at White Sands showed that spraying a hydraulic mist at 3000 psi at an oxyacetylene torch, or at an arcing energy source at 150 amps, could cause the hydraulic fluid to ignite. Varying the distance to the ignition source varied the size of the flame cone, but the flame did not propagate upstream. No spray test onto a hot surface was performed, so the potential for lowering the autoignition temperature due to a fine spray is unknown. Therefore, it was necessary to consider other mitigations to the potential for igniting a hydraulic fluid leak in the aft compartment. APU/hydraulics operation is limited to ascent and reentry phases. On a straight-line basis, assuming no purge effects, hydraulic fluid combustion is oxygen dependent and could be sustained in a practical sense only to an altitude of 80,000 ft. Because the autoignition temperature increases with decreasing pressure, autoignition temperatures rise such that the threat would no longer exist. The threat, both on ascent and descent, is mitigated by purge effects. The Gaseous Nitrogen (GN_2) prelaunch purge dilutes possible air intrusion and remains positive during ascent due to decreasing pressure as the Space Shuttle climbs. During reentry, a Main Propulsion System (MPS) helium purge is initiated at approximately 80,000 ft and continues until wheel stop plus 100 sec. While there is a finite probability of hydraulic fluid ignition , the mitigation measures make the probability low and a secondary risk to the potential of losing the use of a hydraulic system due to hydraulic fluid depletion.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	Cathode Ray Tube (CRT) #4 screen went blank. IFA No. STS-36-09	CRT #4 screen went blank during on-orbit operations. Power cycling provided only temporary recovery of the CRT. The unit was inoperative for the remainder of the flight. CRT #4 was removed at KSC and sent to the vendor for failure analysis. Preliminary assessment did not reveal a generic problem. There are 4 CRTs available; 2 or more CRTs must fail before there is a mission impact. Failure analysis at the vendor isolated the problem to the horizontal deflection amplifier. Further analysis found a cracked solder joint on a capacitor in the amplifier circuit.
10	Oxygen (O_2) bleed orifice leak. IFA No. STS-36-10 HR No. ORBI-299	During cabin operations at 10.2 psi, the crew manually controls the O_2 content of the atmosphere. Presleep O_2 partial pressure was 2.85 psi. Per procedures, the crew connected the bleed orifice, resulting in a rise of O_2 partial pressure to 2.9 psi during the sleep period. The crew tightened the elbow B-nut in an attempt to slow any possible leak, and the bleed orifice functioned nominally. It was thought that the rise in O_2 partial pressure could have been due to a manual control lag effect. The O_2 bleed orifice was checked at JSC, with no discrepancies found.
11	Free water found near humidity separator "A". IFA No. STS-36-11 HR No. ORBI-051 ORBI-254 ORBI-321A	The crew found 1 to 2 cups of water outside humidity separator "A" during required inspection procedures. The Waste Water Management System (WWMS) wand was used to recover the free water. The crew reconfigured to humidity separator "B" for the remainder of the mission. Because of recent free water problems experienced on STS-32/OV-102, a decision was made to remove the humidity separator package and send it to the vendor for failure analysis. Vendor inspection at Dryden found contamination in an area of the separator where it had not previously been seen or experienced on other problem humidity separators.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)		While the free water failure mode has not been defined, crew procedures are in place which require periodic inspection of the humidity separators and surrounding areas for free water. Contingency deorbit can be implemented should both humidity separators and workaround procedures fail.
12	Thruster R4R failed off during pre-entry hot fire test. IFA No. STS-36-12 HR No. ORBI-119	During thruster R4R S/N 235 firing, the chamber pressure did not reach the required pressure within the specified time, and the thruster was deselected by the RCS RM system. This was a failure mode similar to thruster R3D failure. Contamination of the oxidizer valve poppet was suspected. Prior to launch, the rain cover of R4R was found soaked with water. Thruster R4R was removed and sent to the vendor for analysis. Visual inspection at Dryden found no contamination in the thruster throat. Failure analysis at Marquardt determined that nitrates were formed in the oxidizer valve poppet, preventing it from opening in the allotted time.
13	Flash Evaporator System (FES) controller "A" shutdown. IFA STS-36-14 HR No. ORBI-276B	FES controller "A" shut down during on-orbit operations. The shutdown occurred when the water dump mode was initiated. The crew selected the high radiator set point in an attempt to correct the problem. A transducer sensed inadequate FES cooling which resulted in FES controller "A" shutdown. The crew then cycled the radiator switch twice; the FES came online and performed nominally. Troubleshooting at KSC found no problems with the controller.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14	STS-36 hydraulic system depressurization anomaly. IFA No. STS-36-17 HR No. ORBI-036 ORBI-047A ORBI-184 ORBI-188	During reentry, hydraulic system #1 pump pressure did not respond properly when commanded to the low-pressure mode to reduce the rate of a suspected hydraulic leak. Initially, pressure dipped to 2100 psi and then leveled off at 2600 psi for 5 min before dropping to 650 psi; nominal low pressure is 800 psi (500- to 1000-psi range). The low-pressure mode is normally required only for APU start. This was the first time to command a hydraulic system to "low" during reentry; however, periodic load tests verified this capability. Pressure was returned to normal shortly before landing and shutdown after wheel stop. Pump teardown revealed severe scoring in the piston cap through the entire bore, 360° in circumference. Additionally, 6 score marks were found through the housing bore hardcoat anodize. Burnishing was also indicated on the piston. This pump was previously flown on OV-099 and was removed after 5 flights to investigate an anomalous APU vibration. APU pump testing at JSC determined that the vibration was caused by the APU. After being subjected to this anomalous vibration environment on OV-099 and testing at JSC, the pump went through acceptance testing prior to installation on OV-104. STS-36 was the fourth flight for this pump on OV-104. In addition to operation on OV-099 and OV-104 and testing at JSC, the pump was used with 3 other hydraulic systems at Abex (the vendor), the KSC Orbiter Processing Facility (OPF), and at Sundstrand. This was significant because analysis of cap and piston fluid samples found 73 particles larger than 100 microns; the largest were found to be 300 series Corrosion Resistant Steel (CRES), MP35N, and iron oxide, ranging from 125 to 360 microns. None of these materials are used in the cap or piston. The source of this contamination was unknown; however,

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)		because the pump operated with several different hydraulic systems, contamination could have been introduced to the cap or housing bore anywhere along the way. Because contamination was found, a possible scenario for the STS-36 anomaly was that a particle lodged between the large end of the depress piston and the housing bore. To do this, the particle would have to penetrate the hardcoat temporarily and prevent the piston from completing the stroke. It is believed that the piston finally overcame the resistance of the contamination and returned to nominal depressurization mode operation at 600 psi. In addition to use on the Orbiter, this type of hydraulic pump is also used with the SRB Hydraulic Power Units (HPUs) and in many commercial applications. All available pumps were examined for similar scoring. Examination of SRB HPU pumps found light localized scoring at the inner edge of the piston cap and light indications of wear further inside. This scoring was not nearly as bad as that seen on the anomalous STS-36 pump. Note that the SRB HPU pumps are operated for a relatively short time period, but operate in a higher vibration environment than the Orbiter pumps. Examination of 2 spare Orbiter pumps found only light localized scoring at the piston cap inner edge. This scoring did not compare with that seen on the anomalous STS-36 pump and was slightly less than the scoring found on the SRB pumps. These 2 pumps were located at Abex and had only operated during acceptance testing. No indication of wear was found in 2 test stand pumps located at JSC. These 2 pumps were considered in good shape; however, there was no record of operating time. A key factor in determining whether the scoring and wear were operating-time related was the examination of the 3 pumps operating in the Flight Control Hydraulics Laboratory (FCHL). The FCHL pumps had total operating times ranging from 682 to 916 hr. These pumps had experienced thousands of

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)		depressurization/pressurization cycles and hundreds of operating hours in the depressurization mode. Examination of these pumps found only indications of incipient wear and no scoring. The pumps were considered to be in good condition. The vibration environment experienced by the FCHL pumps is extremely low compared to flight environments on the Orbiter and SRB. Examination of other pumps at Sundstrand and in commercial industry was underway; however, there was no indication that major problems would be found. The results of the STS-36 pump anomaly investigation and examination of available pumps were: • The STS-36 depress anomaly was not a hard failure. – The anomaly cleared itself. – The anomaly occurred during off-nominal operations. – There was no prior history of depressurization problems with this or other pumps. • Examination of other pumps determined that the piston cap and housing bore scoring found in the anomalous pump was by far the worst case.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)		• The anomalous pump was subjected to an unique set of operating environments: – Exposure to excessive vibration on OV-099. – Exposure to 6 different hydraulic systems increased the opportunity for introduction of contamination. • Piston cap scoring was apparently not operating time related. – FCHL pumps indicated slight scoring of the piston caps. – JSC test stand pumps indicated very little evidence of scoring. – SRB pumps indicated the beginnings of localized wear after a relatively short operating time. • Anomaly characteristics and failure analysis evidence were consistent with transient contamination.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Left SRB ordnance ring pin embedded in External Tank Attach (ETA) ring foam. IFA No. STS-36-B-01 HR No. INTG-081A	Postflight disassembly of STS-36 SRBs found 3 pins missing from the forward skirt frustum attach ordnance ring area. One pin was found embedded in the ETA ring foam. Loss of these pins had been seen previously and was attributed to water impact; however, this was the first time that a pin was found. Inspection of the pin retainers P/N 10172-0010-001 found that the ends of some of the clips were bent or spread in a way that compromised pin retention. Inspection of 180 retainers revealed that 4 were spread to the point of losing all pin retention capabilities. Three were found bent almost to this point, and 30 additional pin retainers were deformed, but not to the same extent. Retainers are reused after inspection in accordance with United Space Boosters, Inc., (USBI) refurbishment specification 10SPC-0131C. The refurbishment specification requires inspection with no reuse if the pin retainer is "bent out of print". A determination was not made whether the dimension in question is measured prior to reuse. For STS-31 and STS-35, ordnance pins were positively locked in place using a fastener/daisy chain lockwire configuration per Engineering Change Proposal (ECP) 2779. A maximum of 6 pins in series were lockwired together. Thermal qualities of the Inconel lockwire exceeded the maximum heating conditions experienced during flight. For STS-38 and subsequent flights, a new design retainer clip will be used that maintains the pin in place throughout the flight profile.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
2	Left SRB frustum Main Parachute Support System (MPSS) restraint strap bolt missing a nut. IFA No. STS-36-B-02	Postflight inspection of the left STS-36 SRB found that a nut was missing from a MPSS restraint bolt. The main parachute associated with this nut performed nominally. There are 28 additional restraint links on top of the deployment bag and 16 supports on the bottom. Photographic evidence confirmed that the nut was properly installed at assembly. Analysis indicated that the nut was on the bolt at one time, but was not properly torqued. There was no indication of when the nut came off and no indication of distress on the bolt threads. The SRB decelerator subsystem is a Crit 3 function. Because failure would only impact reuse, this is not a safety-of-flight issue.
3	Safety wire missing from Gaseous Nitrogen (GN_2) purge line in RH aft skirt. IFA No. STS-36-B-03 HR No. B-00-15	During postflight inspection of the RH SRB aft skirt, safety wire was found missing from a B-nut in the GN_2 purge line assembly. It is believed that the safety wire was never installed. USBI analysis determined that the installation torque of 480 in-lb results in a safety factor of 11.55 against the B-nut backing off without the safety wire installed. Build paper indicated that the B-nut was installed and properly torqued. Preload is 5.8 times the flight load. The B-nut is qualified without safety wiring. There was no record of a B-nut backing off on prior flights. Inspection of STS-31 B-nuts found that the safety wire was correctly installed. A specific flagnote was added to the drawing to clearly define lockwire requirements. A design change will require drilling the B-nut for lockwiring; this will be in effect for STS-41.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
4	Left SRB drogue parachute first stage reefing line cutter failure. IFA No. STS-36-B-04	During decent of the left SRB after separation, the drogue parachute first stage reefing line cutter failed to fire. This was the first occurrence of this failure since STS-26 and was identical to failures experienced on STS-1 and STS-8. Postflight inspection found the cutter actuation lanyard failed to pull the firing pin. The lanyard was not recovered. Closeout photographs showed that the lanyard was in place prior to flight. The redundant reefing line cutter functioned properly. Test data demonstrated that parachute deployment which causes the lanyard to pull the shear pin at an angle greater than 140° would result in the cutter failure. Reefing line cutters are Crit 3R2 hardware. Line cutter failure is a reuse issue and is not considered a safety-of-flight issue. Redundant cutters provide backup in case of a single cutter malfunction.
5	Right and left SRB ETA ring found with cable tie-wraps disengaged from electrical cable assemblies. IFA No. STS-36-05	During postflight inspection of both left and right SRB ETA rings, several cable tie-wraps were found disengaged from the electrical cable assemblies. Three cable tie-wraps were found disengaged on the right SRB, and 2 on the left. Disengagement was attributed to water intrusion during tow back that weakened the tie-wraps. Cable tie-wraps with metal locking mechanisms are under evaluation for future flights.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
6	Right SRB frustum Marshall Trowellable Ablator No. 2 (MTA-2) debonds. IFA No. STS-36-B-07 HR No. B-60-12 Rev. C DCN4 C-00-04 Rev. B DCN2	During postflight inspection of the right SRB frustum, 2 areas of MTA-2 debond were found. This was the first instance of MTA-2 debond since it was first used. Evidence indicated that the debonds occurred during or after frustum separation. Material analysis performed on the debonded MTA-2 sections concluded that the voids were the result of air bubbles introduced during material application. This conclusion was based on the size, location, and configuration of the voids. Analysis demonstrated adequate MTA-2 strength to preclude loss, even with voids similar to those found. MTA-2 application methods are being evaluated to minimize voids.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1	Right Solid Rocket Motor (SRM) igniter/forward dome boss interface surface metal pitting and Gask-O-Seal damage. IFA No. STS-36-M-01 HR No. BC-02 Rev. B BC-03 Rev. B	During disassembly of the STS-36 booster assemblies, a blow hole was found at the 175° position in the igniter vacuum putty. This putty was laid-up between the igniter and the forward dome as a thermal barrier to stop hot gas excursion to the igniter-to-case sealing surfaces. The blow hole measured 0.3" circumferentially at the igniter adapter and widened to 2.5" circumferentially at a position 4" below the adapter. Blow holes through the putty have been experienced on approximately 65% of all flight and test SRMs/Redesigned Solid Rocket Motors (RSRMs); however, the results were not as severe as that witnessed on the STS-36 Left-Hand (LH) SRM. Significance of this occurrence was the discovery of a depression, or pitting, in both the inner diameter of the forward dome and the outer diameter of the igniter chamber body, as well as a missing portion of the cadmium plating on the inner igniter gasket seal. Sooting was also seen around the outside of the inner igniter gasket seal, extending approximately 100° in either direction from the blow hole. Blow holes observed on the STS-27 Right-Hand (RH) SRM and on Transient Pressure Test Article (TPTA) 1.2 also resulted in corroded metal surfaces and cadmium plate similar to the STS-36 LH SRM. In these cases, the minimum blow hole circumferential measurement was 0.16". This supported the belief that corrosion is not worse with smaller blow holes. The blow hole through the putty was large enough to allow sufficient hot gas to pass to clean the putty off the surfaces of the forward dome and igniter case. Pitting on both of these surfaces was believed by MSFC and Thiokol Corporation metallurgists to be due to "corrosion" as opposed to hot gas "erosion". The hot propellent gases contain a large amount of chlorine, hydrogen chloride, and other corrosive materials. The chlorides were believed to be the primary cause of the corrosion; however, the corrosion process was continued by sea water after splashdown. The pitting is refurbishable; igniter chamber pitting was measured at 1-2 mils in depth.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)		In addition to the pitting of the forward dome and igniter case, examination found missing cadmium plating from the Gask-O-Seal over an area of 1.5" circumferentially by 0.15" radially in the area of the blow hole. Metallurgical analysis found powdery cadmium chloride. The melting point of cadmium is 610°F, and it oxidizes when heated. From hardness testing, observations of metal corrosion, and removal of cadmium plating from the Gask-O-Seal, it was estimated that the igniter joint experienced a temperature in the range of 450°F to 550°F. This finding indicated that the cadmium was removed through corrosion as opposed to melting. Thiokol engineers stated that cadmium stripping is acceptable as long as there is no damage or degradation of the elastomer seal. In this case, no degradation of the elastomer was found. Analysis by Thiokol showed that seal performance is acceptable with exposure to temperatures up to 800°F. The volume on the seal side of the blow hole was very small (3.8 in^3 versus 15 in^3 for a field joint and nozzle-to-case joint). It had a 0.61 sec fill time, and there was no circulation producing additional flow in this area. Therefore, the temperature rise was limited to less than 800°F. Flow/thermal analysis of a worst-case blow hole, measuring 0.1" circumferentially, was performed. A blow hole of 0.1" is considered worst-case because no blow hole has been observed to be less than 0.16", and blow holes less than 0.1" would tend to self-plug. For this size blow hole, the void fill time was determined through this analysis to be 2.4 sec. No damage to the seal would result because the seal surface temperature would be below 450°F, well within the 800°F limit. The analysis showed, however, that the cadmium on the retainer would be exposed to temperatures greater than 610°F, the melting point for cadmium, for a period of

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)		1.2 sec until flow stagnation would occur. It was determined that even if the cadmium melts, no embrittlement or damage to the elastomer is expected. The fact that damage seen on STS-27 and TPTA, with a blow hole of 0.16", was similar to that seen on STS-36 is an indicator that the analysis results were conservative. Worst-case thermal analysis of the igniter chamber steel indicated that the surface temperature rose to 2750°F. This prediction was based on the pitting seen, less than 2 mils in depth. At 2750°F, analysis showed that there was no loss in structural margins of safety. Stresses in the heat-affected zone range from 40-to-140 ksi. The overall joint capability was not compromised by the localized heat-affected zone. The joint Factor of Safety (FOS) was demonstrated by burst tests to be greater than 1.8. Based on the localized heat-affected zone experienced on STS-36, the remaining margin of safety was greater than 0.3. The only resulting concern with a localized heat-affected zone is reuse because of the loss of corrosion protection on the metal surface. A thorough analysis of the likelihood of circulation flow within the igniter joint found no mechanism to generate circulation in the joint. The igniter joint is unlike the nozzle-to-case joint, where nozzle gimballing occurs, or the field joints. In both cases, the dynamic environment provides the potential for creation of a delta pressure in the joint, leading to circulation. In addition, of all SRM igniter joints experiencing putty blow holes, none has been seen with more than 1 blow hole.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
2	Secondary O-ring damage to igniter plug. IFA No. STS-36-M-02 HR No. BN-03 Rev. B	During disassembly of the LH SRM, a flaw was found in the secondary O-ring on the igniter port plug. This port plug secures the hole in the igniter adapter formerly used for the igniter pressure transducer. Primary and secondary seals are redundant. The flaw consisted of material separation, resembling a slit, near the mold line on the inner diameter of the O-ring. The slit measured 0.060" in depth and extended around approximately 50% of the O-ring circumference. A similar material separation was found on the STS-34 RH igniter O-ring. The damaged secondary O-ring on STS-36 passed leak tests at 2159 psia, the maximum environmental operating pressure of the igniter. The most probable cause of the slit was damage at assembly by the edge of the dovetail groove used to hold the O-ring, due to excessive grease in the groove. Excessive grease was found in the dovetail groove of the failed port plug. Excessive grease causes an overfill condition, trapping the O-ring between the edge of the dovetail and the igniter adapter. Circumferential separations of this type are on the inner diameter of the O-ring; therefore, the top and bottom sealing surfaces should not be compromised. Damage to O-ring surfaces was on the face seal, with no gap opening. A pressure test was performed on the damaged O-ring without the primary seal in place. The damaged seal passed the pressure test.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>KSC</u>		
1	Missing washer on Debris Containment System (DCS) Holddown Post (HDP) #6. IFA No. STS-36-K-01 HR No. B-60-12 Rev. C DCN4	Postlaunch inspection of HDP #6 found that the required washer was missing. It was later determined that 2 washers were found on another HDP; only 1 is required. It is believed that the technician inadvertently installed 2 washers on one of the HDPs, thinking that he had installed one on each, per drawing. It should be noted that both Lockheed and NASA quality control inspectors certified that the washer installation was correct. Analysis showed no performance degradation with either no washer or 2 washers installed on any bolt. Installation procedures were changed to state that "1 washer is required on 1 bolt".

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
MCC		
1	Space Shuttle Main Engine (SSME) post-powerdown hardware failure indicated. IFA No. STS-36-D-01	Two erroneous SSME hardware failure identifiers were annunciated approximately 9 min after SSME controller powerdown. The SSME controller cannot generate failure identifiers once it has been powered down. Suspect was a data recording anomaly in the Mission Operational Computer (MOC). Troubleshooting was performed to determine susceptibility to erroneous data. This is not a safety-of-flight issue.

SECTION 8

BACKGROUND INFORMATION

This section contains pertinent background information on the safety risk factors and anomalies addressed in Sections 3 through 7. It is intended as a supplement to provide more detailed data if required. This section is available upon request.

APPENDIX A

LIST OF ACRONYMS

AC	Alternating Current
AFB	Air Force Base
APS	Aft Propulsion System
APU	Auxiliary Power Unit
ATP	Acceptance Test Procedure
BFC	Backup Flight Computer
BFS	Backup Flight System
BITE	Built-In Test Equipment
BOB	Breakout Boxes
BSR	Bite Status Register
CA	California
cc	Cubic Centimeter
ccs	Cubic Centimeters Per Second
CCV	Chamber Coolant Valve
CPM	Cell Performance Monitor
CR	Change Request
CRES	Corrosion Resistant Steel
CRT	Cathode Ray Tube
DCS	Debris Containment System
DFRC	Dryden Flight Research Center
DoD	Department of Defense
DR	Discrepancy Report
ECP	Engineering Change Proposal
EIU	Engine Interface Unit
EPDM	Ethylene Propylene Diene Monomer
EPS	Electrical Power Systems
EST	Eastern Standard Time
ET	External Tank
ETA	External Tank Attachment
	External Tank Attach

APPENDIX A

LIST OF ACRONYMS (CONT.)

F	Fahrenheit
FA	Flight-Critical Aft
FASCO	Flight Acceleration Safety Cutoff System
FC	Fuel Cell
FCHL	Flight Control Hydraulics Laboratory
FCV	Flow Control Valve
FD	Flight Day
FEC	Field Engineering Change
FES	Flash Evaporator System
FMEA/CIL	Failure Modes and Effects Analysis/Critical Items List
FOS	Factor of Safety
FP	Fuel Pump
FPCA	Forward Power Control Assembly
FRI	Flow Restriction Inhibiter
FRR	Flight Readiness Review
FSM	Fuel Supply Module
ft	Feet
GFE	Government Furnished Equipment
GG	Gas Generator
GGVM	Gas Generator Valve Module
GH_2	Gaseous Hydrogen
GN_2	Gaseous Nitrogen
GO_2	Gaseous Oxygen
GOX	Gaseous Oxygen
GPC	General Purpose Computer
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
H_2	Hydrogen
HCF	High-Cycle Fatigue
HDP	Holddown Post
HPF	High-Pressure Fuel
HPFTP	High-Pressure Fuel Turbopump
HPOTP	High-Pressure Oxidizer Turbopump
HPU	Hydraulic Power Unit
HR	Hazard Report
hr	Hour

LIST OF ACRONYMS (CONT.)

I/O	Input/Output
ICHR	Integrated Cargo Hazard Report
IEA	Instrument and Electronics Assembly
	Integrated Electronic Assembly
IFA	Inflight Anomaly
IMU	Inertial Measurement Unit
in-lb	Inch-Pounds
in/sec	Inches Per Second
in^3	Cubic Inch
INCO	Integrated Communications Officer
INTG	Integration
IOP	Input/Output Processor
JSC	Johnson Space Center
Kbit	Kilobit
KSC	Kennedy Space Center
L-2	Launch Minus 2 Days (Review)
lb	Pound
lbf	Pounds Force
LCC	Launch Commit Criteria
LCF	Low-Cycle Fatigue
LDC	Lot Date Code
LH	Left-Hand
LH_2	Liquid Hydrogen
LiOH	Lithium Hydroxide
LO_2	Liquid Oxygen
LOX	Liquid Oxygen
LPFTP	Low-Pressure Fuel Turbopump
LPOTP	Low-Pressure Oxidizer Turbopump
LPS	Launch Process Sequencer
LRU	Line Replaceable Unit
	Line Replacement Unit
LSFR	Launch Site Flow Review
LSOC	Lockheed Space Operations Company

APPENDIX A

LIST OF ACRONYMS (CONT.)

MCC	Main Combustion Chamber
	Mission Control Center
MDM	Multiplexer-Demultiplexer
ME	Main Engine
MECO	Main Engine Cutoff
MEOP	Maximum Expected Operating Pressure
min	Minute
MLP	Mobile Launch Platform
MOC	Mission Operational Computer
MPS	Main Propulsion System
MPSS	Main Parachute Support System
MRB	Material Review Board
MSE	Mission Safety Evaluation
msec	Millisecond
MSFC	Marshall Space Flight Center
MTA-2	Marshall Trowellable Ablator No. 2
mV	Millivolt
N_2O_4	Nitrogen Tetroxide
NASA	National Aeronautics and Space Administration
NSI	NASA Standard Initiator
NSRS	NASA Safety Reporting System
O_2	Oxygen
OI	Operational Instrumentation
OMI	Operations and Maintenance Instruction
OMRS	Operations and Maintenance Requirements Specification
OMRSD	Operational Maintenance Requirements and Specifications Document
OMS	Orbital Maneuvering System
OPF	Orbiter Processing Facility
OPO	Orbiter Program Office
OPOV	Oxidizer Preburner Oxidizer Valve
ORBI	Orbiter
OSMQ	Office of Safety and Mission Quality
OV	Orbiter Vehicle
oz	Ounce
P/N	Part Number
PASS	Primary Avionics Software System
PDU	Power Drive Unit
PLS	Primary Landing Site
POR	Power-On Reset

APPENDIX A

LIST OF ACRONYMS (CONT.)

ppm	Parts Per Million
PRACA	Problem Reporting and Corrective Action
PRCB	Program Requirements Control Board
psi	Pounds Per Square Inch
psi/hr	Pounds Per Square Inch/Hour
psia	Pounds Per Square Inch Absolute
psig	Pounds Per Square Inch Gage
PST	Pacific Standard Time
QD	Quick Disconnect
RCN	Requirements Change Notice
RCS	Reaction Control System
RGA	Rate Gyro Assembly
RH	Right-Hand
RI	Rockwell International
RM	Redundancy Management
RMS	Remote Manipulator System
RPC	Remote Power Controller
RSD	Range Safety Distributor
RSRM	Redesigned Solid Rocket Motor
RSS	Range Safety System
RTG	Radioisotope Thermoelectric Generator
RTLS	Return to Launch Site
RTV	Room-Temperature Vulcanizate
S/N	Serial Number
S&A	Safe & Arm
scfm	Standard Cubic Feet Per Minute
scim	Standard Cubic Inches Per Minute
sec	Second
SII	SRM Ignition Initiator
SIP	Strain Isolator Pad
SOFI	Spray-On Foam Insulation
SR&QA	Safety, Reliability, and Quality Assurance
SRB	Solid Rocket Booster
SRBTS	Solid Rocket Beacon Tracking System
SRM	Solid Rocket Motor
SSC	Stennis Space Center
SSME	Space Shuttle Main Engine
SSRP	System Safety Review Panel
SSV	Space Shuttle Vehicle

APPENDIX A

LIST OF ACRONYMS (CONT.)

STS	Space Transportation System
SUBS	Subsequent
TAL	Transatlantic Abort Landing
TCTI	Time Compliance Technical Instruction
TEM	Test and Evaluation Motor
TPS	Thermal Protection System
TPTA	Transient Pressure Test Article
TVC	Thrust Vector Control
USBI	United Space Boosters, Inc.
VAB	Vehicle Assembly Building
VAC	Volts Alternating Current
VSWR	Voltage Standing Wave Ratio
WSB	Water Spray Boiler
WSMR	White Sands Missile Range
WSTF	White Sands Test Facility
WWMS	Waste Water Management System
°/sec	Degrees/Second