MISSION SAFETY EVALUATION REPORT FOR STS-39

Postflight Edition

Safety Division

Office of Safety and Mission Quality

National Aeronautics and Space Administration

Washington, DC 20546

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REPORT FOR STS-39

Postflight Edition: August 30, 1991

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EXECUTIVE SUMMARY

After a delay of approximately two months due to a rollback from the pad to replace the External Tank door lug housing, Space Shuttle *Discovery* was launched from Kennedy Space Center (KSC) at 7:33 a.m. Eastern Daylight Time (EDT) on April 28, 1991. STS-39 was the first unclassified Department of Defense (DoD) Shuttle mission. The purpose of this mission was to test sensors and tracking devices for the Strategic Defense Initiative (SDI). The amount and quality of Air Force/SDI data collected significantly exceeded expectations.

On April 28, countdown proceeded normally through the T–20 minute hold. No significant problems were encountered except for the Operations Sequence (OPS)-2 recorder starting unexpectedly; it was stopped by an uplink command. The T–9 minute hold was extended to evaluate the OPS-2 recorder anomaly. It was determined that there was no problem with the Orbiter's computers. The countdown was then resumed, and Space Shuttle *Discovery* was subsequently launched.

STS-39/OV-104 ascent performance was nominal. However, approximately 15 minutes after Main Engine Cutoff (MECO), Auxiliary Power Unit (APU) #2 Fuel Pump (FP)/Gas Generator Valve Module (GGVM) coolant system "A" Flow Control Valve (FCV) was not pulsing as commanded. Coolant system "B" was selected, and nominal cooling occurred. Coolant system "A" was not used during entry, and therefore did not impact entry operations.

Payload operations were very successful. However, the sun sensor used for alignment of the Shuttle Pallet Satellite (SPAS) deployed on Flight Day (FD) 3 repeatedly swung SPAS into the wrong attitude. The cause of the original sun sensor anomaly was not determined. The SPAS was redeployed on FD 4, with no problems experienced. *Discovery* performed 60 rocket firings at various distance from the SPAS over a 36-hour period.

Discovery landed on KSC runway 15 at 2:55 p.m. EDT on May 6, 1991. This was the second time in 6 months that the Space Shuttle was diverted to KSC for landing because of high winds at Edwards Air Force Base (EAFB), California. This was also the seventh of 40 Shuttle missions to land at KSC in the history of the Space Shuttle Program. The Main Landing Gear (MLG) outer right tire shredded 3 of the 16 cords due to either an uneven landing or a maximum-force breaking test during rollout. Contributing factors to the tire cord shredding were the development of last-minute crosswinds and reluctance of the ground controllers to distract the Shuttle pilots with warnings of the low flightpath. As a corrective action, communication procedures will be modified for future flights to allow ground controllers more latitude to talk to the crew.

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 Director 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 Hazard Reports (HRs). Unresolved safety risk factors impacting the STS-39 flight were also documented prior to the STS-39 Flight Readiness Review (FRR) (FRR Edition) and the STS-39 Launch Minus Two-Day (L-2) Review (L-2 Edition). This final Postflight Edition evaluates performance against safety risk factors identified in the 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 Director 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-39 safety risk factors that represent a change from previous flights, factors from previous flights that have an 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 is 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 description of the STS-39 mission, including launch data, crew size, mission duration, launch and landing sites, and other mission- and payload-related information.

Section 3 - Contains a list of safety risk factors/issues, considered resolved or not a safety concern prior to STS-39 launch, that were impacted or repeated by anomalies reported for the STS-39 flight.

Section 4 - Contains a list of safety risk factors that were considered resolved for STS-39.

Section 5 - Contains a list of Inflight Anomalies (IFAs) that developed during the STS-37 mission, the previous Space Shuttle flight.

Section 6 - Contains a list of IFAs that developed during the STS-41 mission, the previous flight of the Orbiter vehicle (OV-103).

Section 7 - Contains a list of IFAs that developed during the STS-39 mission. Those IFAs considered to represent a safety risk 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 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-39 MISSION SUMMARY

2.1 Summary Description of the STS-39 Mission

After a delay of approximately two months due to a rollback from the pad to replace the External Tank door lug housing, Space Shuttle *Discovery* was launched from Kennedy Space Center (KSC) at 7:33 a.m. Eastern Daylight Time (EDT) on April 28, 1991. STS-39 was the first unclassified Department of Defense (DoD) Shuttle mission. The purpose of this mission was to test sensors and tracking devices for the Strategic Defense Initiative (SDI). The amount and quality of Air Force/SDI data collected significantly exceeded expectations.

On April 28, countdown proceeded normally through the T–20 minute hold; no significant problems were encountered. When countdown was resumed at the end of the T–20 minute hold, the Operations Sequence (OPS)-2 recorder started unexpectedly; it was stopped by an uplink command. Countdown then proceeded to the T–9 minute hold, which was extended to evaluate the OPS-2 recorder anomaly. It was determined that there was no problem with the Orbiter's computers. The countdown was then resumed from the T–9 minute point, and Space Shuttle *Discovery* was subsequently launched.

STS-39/OV-104 ascent performance was nominal. However, approximately 15 minutes after Main Engine Cutoff (MECO), Auxiliary Power Unit (APU) #2 Fuel Pump (FP)/Gas Generator Valve Module (GGVM) coolant system "A" Flow Control Valve (FCV) was not pulsing as commanded. Coolant system "B" was selected, and nominal cooling occurred. Coolant system "A" was not used during entry and, therefore, did not impact entry operations.

With the exception of one malfunction, the Quadruple Ion-Neutral Mass Spectrometer (QINMS) cover did not move when first commanded, all payload operations were nominal on Flight Day (FD) 1. However, the QINMS cover was commanded to move a second time, and operation was normal. The OPS-2 recorder that had unexpectedly gone to "record" prior to launch performed nominally.

On FD 2, the Cryogenic Infrared Radiance Instrumentation for Shuttle (CIRRIS) data acquisition quantity was above expectation; 9 of the 17 CIRRIS experiments were completed. By the end of the mission, the CIRRIS infrared telescope completed 15 of the 17 CIRRIS data takes; overall CIRRIS data acquisition was 150% of the planned pre-mission data.

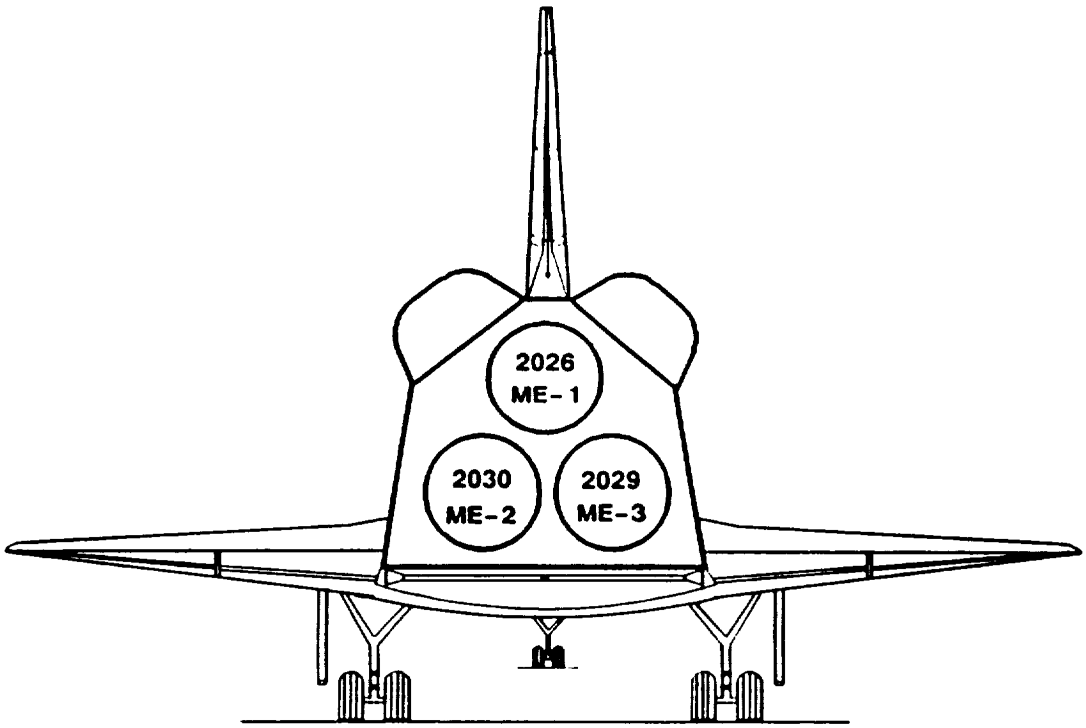
On FD 3, the Infrared Background Signature Survey (IBSS) was released using *Discovery's* Remote Manipulator System (RMS). The Shuttle Pallet Satellite (SPAS) was also deployed successfully. However, the sun sensor used for the SPAS alignment repeatedly swung it into the wrong attitude; this delayed the associated experiments for about 9 hours. The SPAS was redeployed on FD 4, with no problems experienced. *Discovery* performed 60 rocket firings at various distance from the SPAS over a 36-hour period. The cause of the original sun sensor anomaly was not determined.

On FD 5, the Chemical Release Observation (CRO) experiment "B" was deployed, and CRO activities were successfully completed on FD 6. The SPAS/IBSS payload was reberthed and placed on Orbiter power via the Remotely Operated Electrical Umbilical (ROEU).

Discovery landed on KSC runway 15 at 2:55 p.m. EDT on May 6, 1991. This was the second time in 6 months that the Space Shuttle was diverted to KSC for landing because of high winds at Edwards Air Force Base (EAFB), California. This was also the seventh of 40 Shuttle missions to land at KSC in the history of the Program. During landing, the Main Landing Gear (MLG) right-side tires contacted the runway before the MLG left-side tires; 168 feet from the threshold for the right tires and 383 feet for the left tires. The MLG outer right tire shredded 3 of its 16 cords due to either the uneven landing or a maximum-force breaking test during rollout. Rollout lasted for 56 seconds; braking pressure was applied for 36 seconds of this time. Contributing factors to the tire cord shredding were development of last-minute crosswinds and reluctance of the ground controllers to distract the Shuttle pilots with warnings of the low flightpath. As a corrective action, communication procedures will be modified for future flights to allow ground controllers more latitude to talk to the crew.

2.2 Flight/Vehicle Data

- Launch Date: April 28, 1991
- Launch Time: 7:33 a.m. EDT
- Launch Site: KSC Pad 39A
- RTLS: Kennedy Space Center, Shuttle Landing Facility
- TAL Site: Zaragosa, Spain
- Alternate TAL Site: Moron, Spain
- Landing Site: Kennedy Space Center, Runway 15
- Landing Date: May 6, 1991
- Landing Time: 2:55 p.m. EDT
- Mission Duration: 8 Days, 7 Hours, 22 Minutes
- Crew Size: 7
- Inclination: 57.0°, Direct Insertion
- Orbit: 140 x 140 Nautical Miles
- Orbiter: OV-103 (12) *Discovery*
- ET-46
- SRBs: BI-043
- RSRM Flight Set #15
- MLP #2



ENGINE	#2026	#2030	#2029
POWERHEAD	#2030	#2015	#2027
MCC*	#2104	#2026	#4012
NOZZLE	#4014	#4013	#4015
CONTROLLER	F19	F17	F22
FASCOS*	#11	#26	#28
HPFTP*	#4012R1	#4011R2	#2126
LPFTP*	#2025R2	#2027	#2029
HPOTP*	#2226R3	#4506R1	#4008R3
LPOTP*	#2224	#2020	#2121

* Acronyms can be found in Appendix A.

2.3 Miscellaneous Items of Interest for the STS-39 Mission.

- This was the second mission with the new AP-101S General Purpose Computer (GPC) in all 6 GPC positions (5 active, 1 spare).

- This was the second mission using the OI-08F software release for the Primary Avionics Software System (PASS) and the Backup Flight Software (BFS). This was also the second mission using the AR-02 software release in the Main Engine Controller (MEC).

- The Hydrogen Dispersal System was installed on Mobile Launch Platform (MLP) #2 prior to the first STS-39 pad evolution in February 1991.

- The 31 pounds per square inch gage (psig) Gaseous Oxygen (GO_2) Flow Control Valves (FCVs) were shimmed to the step #1 FCV configuration; 93% in high flow and 55% in low flow.

- During the first STS-39 pad evolution, an anomaly occurred with the Right-Hand (RH) Solid Rocket Booster (SRB) Range Safety System (RSS) Automatic Gain Control (AGC). The AGC signal level was reported to be 6 decibels (dB) below normal, but not out-of-specification. Troubleshooting could not be scheduled prior to rollback of STS-39 to the Vehicle Assembly Building (VAB). During the second pad test evolution, no problems were encountered with the RSS AGC. Troubleshooting performed to recreate the earlier anomaly was unsuccessful. A special open-loop test was also conducted with the RSS transmitter in the launch configuration (in the high-power, omni-directional mode). All RSS receivers responded properly during this test. This anomaly was dispositioned as unexplained; the Eastern Test Range Safety Organization concurred. No postflight anomalies were reported for this system.

2.4 Payload Data

The STS-39 mission was the first unclassified Department of Defense (DoD) dedicated Space Shuttle mission, highlighted by around-the-clock observations of the atmosphere, gas releases, Shuttle engine firings, subsatellite gas releases, and the Shuttle's orbital environment encompassing a wavelength range from infrared to the far ultraviolet.

The Remotely Operated Electrical Umbilical (ROEU) was flown for the first time on STS-39/OV-103. The ROEU provides a capability for remote separation (or connection) of electrical services between the Orbiter and the payload. System control (mate/demate process) is performed from the Orbiter aft flight deck. Active elements of the ROEU were incorporated in the Orbiter-mounted portion of the system located on the payload bay longeron.

Payload Bay:

- Air Force Program-675 (AFP-675) — collected infrared data to support the Strategic Defense Initiative (SDI) Program. AFP-675 is an experiment support system consisting of the Shuttle and 5 sensing instruments. The sensing instruments included:

    - Cryogenic Infrared Radiance Instrumentation for Shuttle (CIRRIS) — obtained simultaneous spectral and spatial measurements of airglow and auroral emissions.

    - Far Ultraviolet Camera (FAR UV) Experiment — used for imaging and photometry of natural and man-made far ultraviolet emission phenomena.

    - Uniformly Redundant Array (URA) — studied astrophysical sources of x-ray radiation.

    - Quadruple Ion-Neutral Mass Spectrometer (QINMS) — determined ion and neutral contamination species identification for CIRRIS.

    - Horizon Ultraviolet Program (HUP) — demonstrated the capability to measure spatial and spectral characteristics of the Earth's horizon as observed in the ultraviolet wavelength region and to analyze Shuttle contamination.

- Infrared Background Signature Survey (IBSS) — was composed of 3 separate elements: the Shuttle Pallet Satellite-II (SPAS-II), the Critical Ionization Velocity (CIV) package, and the Chemical Release Observation (CRO) experiment.

  - The SPAS-II element incorporated a liquid-helium-cooled infrared sensor, the Arizona Imager/Spectrograph (AIS) multispectral sensor, 2 low-light television cameras, and various support subsystems. SPAS-II was deployed from the Orbiter using the Remote Manipulator System (RMS), and gathered spectral and spatial data during several experiments.

  - The CRO experiment collected infrared, visible, and ultraviolet time-resolved radiometric data associated with the release of liquid rocket propellants in near Earth orbit. The CRO consists of 3 subsatellites and associated launcher systems. Each satellite was loaded with a different chemical: Monomethyl Hydrazine (MMH) — CRO A, Unsymmetric Dimethyl Hydrazine (UDMH) — CRO B, and Nitrogen Tetroxide (N_2O_4) — CRO C.

  - The CIV package investigated the interaction of neutral gases with the ambient weakly-magnetized plasma. The CIV element included 4 compressed gas canisters (xenon, neon, carbon dioxide, and nitrous oxide) that released plumes of gas out of the Orbiter bay upon crew command. The SPAS-II and the CIV monitor system observed the gas as it was released.

- Space Test Program-01 (STP-01) — consisted of 5 experiments, including instrument electronics and control systems mounted on a Hitchhiker-M cross-bay carrier.

  - Advanced Liquid Feed Experiment (ALFE) — studied zero-gravity performance of liquid collection systems. The ALFE provided the first spaceflight demonstration of an electronic pressure regulator and a series of ultrasonic propellant-level and flow-sensing systems that are key components of an advanced spacecraft propulsion system.

  - Ascent Particle Monitor (APM) — measured particle activity in the payload bay during the immediate prelaunch period and during ascent.

  - Ultraviolet Limb Imaging (UVLIM) Experiment — studied the composition of the atmosphere using measurements of the ultraviolet airglow.

- Data System Experiment (DSE) — evaluated the performance of the MIL VAX computer and Erasable Optical Disk (EOD) in a microgravity environment.

- Spacecraft Kinetic Infrared Test (SKIRT) — measured the atomic oxygen glow effect in visible, infrared, and ultraviolet spectra. The SKIRT consisted of 2 separate and independent components: the Gaseous Luminosity of Optical Surface (GLOS) and Circular Variable Filter (CVF).

• Multi-Purpose Experiment Canister (MPEC) — is a modified Get Away Special (GAS) canister containing an ejectable experiment.

• Ultraviolet Plume Instrument (UVPI) — obtained imagery and/or signature data of the Orbiter for space-based ultraviolet sensors on the UVPI satellite.

Middeck:

• Cloud Logic to Optimize Use of Defense Systems (CLOUDS)-1A-1 — is a hand-held 35 mm camera for use by the crew to take a series of high-resolution photographs of cloud formation, dissipation, and opaqueness.

• Radiation Monitoring Equipment (RME) III-03 — measured the rate and dosage of ionizing radiation to the crew at different locations throughout the Orbiter cabin. The hand-held instrument measures gamma ray, electron, neutron, and proton radiation, and calculates the amount of exposure.

SECTION 3

SAFETY RISK FACTORS/ISSUES IMPACTED BY STS-39 ANOMALIES

This section lists safety risk factors/issues, considered resolved (or not a safety concern) for STS-39 prior to launch (see Sections 4, 5, and 6), that were repeated or related to anomalies that occurred during the STS-39 flight (see Section 7). 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 STS-39 Safety Risk Factors

Orbiter 5	Potential for lube oil leak on STS-39/OV-103 Auxiliary Power Unit (APU) #2, Serial Number (S/N) 301, gearbox.	STS-39/OV-103 APU #2, S/N 301, was observed with signs of lube oil around the gearbox seam. This observation was made during lube oil drain operations. The exact source of the oil was unknown; oil was presumed to have leaked from the gearbox. Initial troubleshooting included wiping the gearbox seam clean and pressurizing the gearbox to 50 pounds per square inch absolute (psia), greater than 2 times the gearbox operating pressure. Post-pressurization inspection found slight traces of lube oil. The gearbox was serviced and again pressurized to 50 psia. Inspection found no new signs of lube oil; however, a minor pressure drop was noted during this period (less than 2 psia). This pressure drop was believed to have been due to pressure released through the seal cavity drain vent. The APU gearbox condition was reviewed by the Material Review Board and approved for flight "as is". During STS-39 entry, APU #2, S/N 301, lube oil outlet pressure was low, reading 25 psia; nominal outlet pressure is 40 to 50 psia (IFA No. STS-39-V-11). Postflight visual inspection of the aft compartment at KSC showed no indication of a leak. This was the first APU lube oil pressure-low anomaly in the Program. Previous lube oil anomalies were related to high lube oil outlet pressure. It was concluded that the low oil pressure was due to low oil quantity, which occurred prior to flight during the Orbiter Processing Facility (OPF) ullage check processing.

ITEM		COMMENT
Section 4: Resolved STS-39 Safety Risk Factors		
SRM 2	Inner Boot Ring (IBR) bond line separation on STS-38 Right-Hand (RH) Solid Rocket Motor (SRM).	No SRM IBR anomalies were reported on STS-39. Postflight inspection of STS-39 did, however, identify 11 circumferential areas of erosion/washing and wedgeouts in the nozzle cowl/Outer Boot Ring (OBR) (IFA No. STS-39-M-01). A trend assessment performed using Redesigned Solid Rocket Motor (RSRM) historical flight data indicated no general trend of degraded performance or increased wedgeouts or washouts. The trend assessment also determined that the STS-39 RH cowl wedgeouts represented the worst case to date for a flight motor. Relative to the OBR, the trend assessment illustrated a small decrease of the mean margin of safety; however, no violation of the safety factor requirements was indicated.

SECTION 4

RESOLVED STS-39 SAFETY RISK FACTORS

This section contains a summary of the safety risk factors that were considered resolved for STS-39. These items were 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 rationale for flight is based on findings resulting from System Safety Review Panel (SSRP), Prelaunch Assessment Review (PAR), and Program Requirements Control Board (PRCB) evaluations (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.

Hazard Report (HR) numbers associated with each risk factor in this section are listed beneath the risk factor title. Where there is no baselined HR associated with the risk factor, or if the associated HR has been eliminated, none is listed. Hazard closure classification, either Accepted Risk {AR} or Controlled {C}, is included for each HR listed.

The following risk factors in this section represent a low-to-moderate increase in risk above the Level I approved hazard risk baseline. The NASA Safety Community assessed the relative risk increase of each and determined that the associated increase was acceptable.

Integration 1	New Criticality 1 and 1R2 failure modes have been identified for the Rate Gyro Assemblies on the Orbiter and Solid Rocket Boosters.
Integration 2	Upgrade of Engine Interface Unit loss of output Critical Items List to Criticality 1/1.

SECTION 4 INDEX

RESOLVED STS-39 SAFETY RISK FACTORS

SECTION 4 INDEX - CONTINUED

RESOLVED STS-39 SAFETY RISK FACTORS

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	New Criticality (Crit) 1 and 1R2 failure modes have been identified for the Rate Gyro Assemblies (RGAs) on the Orbiter and Solid Rocket Boosters (SRBs). HR No. INTG-144C {C} INTG-165A {C} B-50-18 Rev. C-DCN2 {C} ***No Orbiter or SRB RGA anomalies were reported on STS-39.***	Review of recent RGA test data indicated a large output transient, up to 45% of full scale, that lasts approximately 10 seconds (sec) when power to an RGA is lost. It was previously believed that the RGA output would immediately drop to zero when power was removed. Redundancy Management (RM) software normally selects the second highest output from 1 of 4 SRB RGAs for further processing. Post-SRB separation during ascent, and during descent, RM selects the second highest output from 1 of 4 Orbiter RGAs. However, because it is now known that RGA output can stay high for as long as 10 sec after power is removed, the potential exists for RM to select, as the second highest value, erroneous output data from an RGA that loses power. Selection of this erroneous data could lead to loss of vehicle control and subsequent loss of the crew and vehicle. Reevaluation of the RGA Failure Modes and Effects Analysis (FMEA) for Orbiter and SRB RGA power circuits, and for the effects of simultaneous loss of power to 2 RGAs, identified Crit 1R2 failure modes for both the Orbiter and SRB RGAs. The first failure mode is a latent, redundant power feed circuit component (i.e., a remote power controller, a diode failing open, etc.). The second failure mode is loss of a second string redundant path and power feed to another RGA with a non-redundant power source. These 2 failures would result in simultaneous loss of power to 2 RGAs. A Crit 1/1 failure mode associated with the SRB RGAs was identified. No Crit 1/1 failure modes were identified for the Orbiter RGAs. In the case of the SRB RGAs, demate of a single connector (55W1P113/J3) on the Orbiter Master Event Controller (MEC) #2 or in the Orbiter Avionics Bay #5 feedthrough

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1 (Continued)	New Crit 1 and 1R2 failure modes have been identified for the RGAs on the Orbiter and SRBs.	(50W92P299/J99) would result in simultaneous loss of power to 2 SRB RGAs. Additionally, opening of all 3 poles of the 3-pole MEC #2 power toggle switch would also cause simultaneous loss of power to 2 SRB RGAs. Power distribution to the RGAs within the SRBs is isolated and redundant. However, analysis determined that a power bus transient greater then 3 milliseconds (ms) can cause the SRB Multiplexer-Demultiplexer (MDM) to experience a Power-On Reset (POR) and clear all previous commands. This will result in power-down of 2 of the 4 SRB RGAs commanded by the SRB MDM that experienced the POR. Coupled with the RGA power-down output transient, this could provide sufficient data for RM selection and result in a Crit 1/1 condition. No power transients have been experienced to date. However, the decision was made to cross-strap power to the forward SRB MDMs to alleviate this problem. This fix has some residual risk. If there are failures in 2 MDM current limiters of the same hybrid design, followed by a short circuit at the same regulator output, excessive current would be drawn from both MDM power supplies. In turn, this could cause a momentary power drop on both SRB power busses A and B. This would cause a POR of the aft MDM, resulting in catastrophic shutdown of both SRB Hydraulic Power Units (HPUs) and loss of Thrust Vector Control (TVC). The cross-strapped SRB power configuration was flown on STS-28. It is considered a lower risk than not cross-strapping the SRB MDM power and flying with several potential single-point failures in the Orbiter-side bus B circuitry that can lead to MDM POR and SRB RGA shutdown. There have been no SRB current limiter failures to date. A waiver to fly with the potential current limiter failure mode was approved for STS-39.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1 (Continued)	New Crit 1 and 1R2 failure modes have been identified for the RGAs on the Orbiter and SRBs.	Two Critical Item List (CIL) waivers, CR S50260D and CR S50260S, were submitted to address the newly identified Crit 1/1 and 1R2 conditions. These waivers were approved for STS-37/OV-104 and STS-39/OV-103. CR S50260D addressed component and power bus failures in the Orbiter that create the Crit 1R2 condition for Orbiter and SRB RGAs. CR S50260S addressed the 2 Orbiter connector demates/failures that create the Crit 1/1 condition for SRB RGAs. The existing Crit 1/1 CIL for the 3-pole MEC #2 power toggle switch was unchanged by these findings. Flight Rule (FR) 8-47 was prepared to reduce the risk of simultaneous RGA power loss and output of erroneous data. This FR directs the crew to deselect 1 Orbiter RGA when the first failure is detected. Ground test procedures were incorporated to verify the integrity of the SRB RGA backup power logic source during essential power bus tests at T−1 hour. Photographic documentation of the Crit 1/1 connectors was also mandated to ensure proper connector seating. Review of the applicable photographs and video tape determined that these connectors were properly installed on STS-39. Review of recent processing records determined that one of the critical connectors in question (55W1P113/J3) had not been demated since STS-26. Although the other connector (50W92PJ99) was demated during the STS-41 flow, no connector-related anomalies were reported in association with the STS-41 flight. The integrity of the STS-39 connector circuits was verified during the STS-39 flow. Additionally, a switch guard was installed over the MEC #2 power toggle switch to preclude inadvertent action by the crew. An effort is underway to provide a design solution to eliminate these failure modes from the system.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)	New Crit 1 and 1R2 failure modes have been identified for the RGAs on the Orbiter and SRBs.	To date, there have been no failures in the Orbiter and SRB RGA power circuits. There is also a low failure rate for critical power circuit components (i.e., remote power controllers and diodes). Rationale for STS-39 flight was: • Redundant power circuits were tested during normal flow processing. Additional prelaunch tests were identified to verify the SRB RGA backup power logic source. Power to the forward SRB MDMs was cross-strapped to preclude POR conditions in the event of Orbiter power transients. A waiver was approved to fly in this configuration. • RGA power circuit components and connectors have a high reliability. • Photographic documentation of critical connectors demonstrated proper connector installation. • Failure of either connector at the pad is instantly detectable (loss of SRB bus B). • Integrity of the 2 connectors in question is verified on each flow. • A switch guard was installed over the MEC #2 power toggle switch to preclude inadvertent actuation. *This risk factor was acceptable for STS-39.*

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
2	Upgrade of Engine Interface Unit (EIU) loss of output CIL to Crit 1/1. HR No. INTG-019 {AR} INTG-165A {C} ***No EIU anomalies were reported on STS-39.***	During the process of upgrading an EIU loss of output CIL for the POR condition, failure analysis revealed that an EIU power supply anomaly could delay Space Shuttle Main Engine (SSME) shutdown for a period up to 80 ms. Occurrence of this failure, coupled with a low-level sensor initiated Main Engine Cutoff (MECO), could cause Crit 1/1 effects. This power supply anomaly would cause the loss of 2 of 3 Main Engine (ME) command channels for 1 command cycle, consequently delaying engine shutdown for up to 80 ms. An 80-ms delay could result in violation of the Interface Control Document (ICD) requirement for 80-pounds (lb) minimum of Liquid Oxygen (LO_2) during SSME shutdown, thereby possibly causing High-Pressure Oxidizer Turbopump (HPOTP) cavitation. Ascent performance data verified that a Low-Level Cutoff (LLCO) of the engines can take place with no failure occurrences and all ascent systems operating within accepted performance dispersions. Since no failures are required to get to the LLCO situation, a single EIU power supply failure, coupled with a LLCO, could result in catastrophic effects (Crit 1/1). EIU failure data indicate that EIUs have experienced 20 power supply POR anomalies; however, no PORs have occurred during flight. An EIU design change has been identified to minimize POR occurrences. A 2-flight waiver, for STS-37 and STS-39, was approved.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)	Upgrade of EIU loss of output CIL to Crit 1/1.	Rationale for STS-39 flight was: • There have been no LLCOs experienced in the Shuttle program. • No EIU power supply anomalies have been experienced during flight. • Acceptable ascent performance dispersions are identified prior to each mission. • The exposure time frame for the EIU failure is small (80 ms). • The worst-case scenario of Crit 1 effects requires a slow shutdown sequence during LLCO together with the 80-ms delay caused by the EIU failure. • The probability of a catastrophic situation is extremely low ($2x10^{-8}$). ***This risk factor was acceptable for STS-39.***
3	Backup Flight Software (BFS) can hang in a "wait" state failure mode. HR No. ORBI-066 {AR} ***No BFS anomalies were reported on STS-39.***	During investigation of the STS-37 BFS anomaly (see Section 5, Integration 1) in the Shuttle Avionics Integration Laboratory (SAIL), a new BFS failure mode was discovered. OPS-transition testing was being performed when the BFS hung in the "wait" state during a transition from OPS-0 to OPS-3. This testing was considered abnormal, because the operator rapidly toggled in and out of OPS-3 for 30 transitions; normally, only single OPS transitions are performed. Transitions are not performed after Primary Flight System (PFS)/BFS synchronization.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
3 (Continued)	BFS can hang in a "wait" state failure mode.	This failure occurred only once. The investigation also included a code audit of the BFS and the Primary Avionics Software System (PASS). The code audit determined that the BFS only reads the counter once when processing the following critical functions: • OPS transitions • Input/Output (I/O) resets • Transitions from "halt" to "standby" mode. The increased processing time of the new General Purpose Computers (GPCs), together with the BFS reading the counter only once for critical functions, provides a 3-ms window for the BFS to be hung in the "wait" state when rapid, repeated critical functions of these types are performed. The PASS was determined through the code audit to read the counter at least twice for all critical functions. A Discrepancy Report (DR) was written to identify and resolve the problems associated with this discovery. However, problem resolution was not implemented prior to STS-39. The risk of flying with this BFS problem was considered very low because off-nominal transitional operations must be performed during random 3-ms windows of opportunity for the "wait" state failure to occur. If this failure does occur, the crew would be required to perform the BFS Initial Program Load (IPL). Complete failure of all 4 primary GPCs would be required before a "wait" state failure would become a Crit 1/1 condition.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
3 (Continued)	BFS can hang in a "wait" state failure mode.	Rationale for STS-39 flight was: • Off-nominal transitional operations must be performed; transitions are not performed after PFS/BFS synchronization (unlike the SAIL failure case). • There is a 3-ms window in which off-nominal operations must be performed to achieve the "wait" state failure (<5 out of 65,536 ms). • Failure of 4 primary GPCs would be required for a "wait" state failure to become a Crit 1/1 condition. ***This risk factor was acceptable for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	OV-102 20-psi helium regulator leak. HR No. ORBI-306 {AR} ***No helium regulator anomalies were reported on STS-39.***	During leak check activities following the first STS-35 launch attempt, 1 of 2 20-psi helium regulators was found to have a 1 x 10^{-4} standard cubic centimeters per second (sccs) leak. The regulator was removed and returned to the vendor for evaluation. It was originally installed in OV-102 prior to its first flight and had experienced 9 missions. The 20-psi regulator fleet leader was on STS-41/OV-103 and had experienced 11 missions. Testing at the vendor identified an external helium leak greater than 18 standard cubic inches per minute (scim) at 285 pounds per square inch (psi). A 2-scim leak was observed at the maximum system operating pressure of 30 psi; the allowable leak rate at this pressure is 3 scim. Inspection and bubble leak checks identified 3 cracks in the sensor diaphragm. Wrinkles were also observed on the diaphragm, which is constructed of 2 plies of 347 stainless steel, each approximately 2 mils thick. The diaphragm is exposed to Gaseous Hydrogen (GH_2) sense line pressure. It exerts force on the Belleville springs that operate the regulator pilot poppet and regulate helium pressure. This was the first failure of this diaphragm type in the Program history. Materials and processing analysis at Rockwell International (RI) indicated that the failure was caused by fatigue cracking resulting from stress concentration at wrinkles. Possible causes of the wrinkles include reverse repressurization of the diaphragm and overstress during proof-pressure testing. Plastic deformation of the diaphragm is believed possible during proof-pressure testing. Because of this, the potential exists that all 20-psi regulator diaphragms are, at a minimum, wrinkled.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	OV-102 20-psi helium regulator leak.	Because one side of the diaphragm is exposed to GH_2, leaks through the diaphragm could result in Hydrogen (H_2) leakage into the aft compartment through the regulator ambient vent. Analysis indicated that a ruptured diaphragm could back-flow GH_2 at a maximum rate of 5,000 scim. This potential leak is detectable by the aft compartment Hazardous Gas Detection System (HGDS) and would have resulted in a scrub prior to launch. The regulator is used post-MECO to regulate the helium purge of H_2 lines in the Main Propulsion System (MPS). It is also used during reentry and landing to maintain positive pressure in the MPS line and eliminates the potential for drawing in contamination. A helium isolation valve is located upstream of the 20-psi regulator. The isolation valve could have been closed if the regulator failed open. A recent decision was made to pull all 20-psi helium regulators from the fleet to inspect the diaphragms for wrinkles or potential leaks. Replacement 20-psi helium regulators have new diaphragms installed. This removal and replacement effort was accomplished on OV-103.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1 (Continued)	OV-102 20-psi helium regulator leak.	Rationale for STS-39 flight was: • This was the first and only diaphragm failure in the Program history. • OV-103 20-psi regulators were replaced. • If GH_2 leaks through the diaphragm after launch, the maximum leak rate results in below allowable aft compartment H_2 concentrations and flammability limits. ***This risk factor was acceptable for STS-39.***
2	STS-39/OV-103 pilot-side Display Driver Unit (DDU) Attitude Direction Indicator (ADI) ball hesitation. ***No DDU ADI problems were reported on STS-39.***	During OPS-9 dedicated display dynamic drive testing on STS-39/OV-103, the pilot-side DDU ADI ball occasionally hesitated. Similar testing on the commander-side DDU found the ADI ball to work properly. The ADI from the commander-side was relocated to the pilot-side position, where it also was found to hesitate. The commander's ADI was returned to the commander-side position and was found to operate correctly. The identical OPS-9 testing was performed on OV-104 with the same results; ADI ball hesitation was experienced on the pilot side only. Subsequent testing at the SAIL determined that this anomalous condition was demonstrated in the pilot and aft ADIs. Testing with the new GPCs and new flight software, and with the old GPCs and STS-35 flight software, gave the same results.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	STS-39/OV-103 pilot-side DDU ADI ball hesitation.	This problem was traced to the DDU and the way it processes ADI data; the DDU does not process the ADI ball data for one update cycle on an intermittent basis. This phenomenon occurs both in the OPS-9 dynamic drive tests on the ground and when using the OPS-0, 1, 2, 3, and 6 flight software. Troubleshooting and SAIL testing continue to further isolate the cause of this problem. Testing is also planned for OV-105. The STS-39 crew was informed of the problem. It was demonstrated to the crew, who determined that the condition was acceptable for flight and would not adversely impede their performance. Rationale for STS-39 flight was: • ADI ball hesitation was demonstrated to the STS-39 crew and was determined not to impede performance. • There are no other known DDU processing anomalies in the new GPC OPS software. ***This risk factor was acceptable for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	STS-37/OV-104 Auxiliary Power Unit (APU) #3, Serial Number (S/N) 307, uncommanded Gas Generator Valve Module (GGVM) Shutoff Valve (SOV) motion. HR No. ORBI-031 {AR} ORBI-184 {AR} ***There were no APU SOV failures reported on STS-39. However, STS-39 APU #2 Fuel Pump/GGVM cooling system "A" failed to provide initial cooling after on-orbit APU shutdown (IFA-39-V-02). The problem was traced to a system "A" spray valve (LV25) which had received an exception (EV2123R1) to its 9-month life limit. This failure was not related to the STS-37 APU SOV anomaly.***	APU #3, S/N 307, was installed on STS-37/OV-104 since the last flight. During APU fuel line high-point bleed operations, the injector temperature indicated an unexpected increase of 60°F over a 24-minute (min) period. Review of the gas generator bed temperature data confirmed the temperature rise. Injector and gas generator bed temperature rise are a positive indication that fuel is reaching the bed. Troubleshooting isolated the problem to fuel leakage through the SOV. Movement of the SOV was confirmed by a fuel pump inlet pressure decrease and an exhaust duct pressure increase during the same 24-min period. A subsequent GGVM liquid leak check of the SOV was performed at 370 pounds per square inch absolute (psia), with no indicated leakage. This further confirmed that the SOV had moved slightly to the open position and had not leaked. Additional leak checks at pressures between 20 and 100 psia were performed to confirm low-pressure sealing. APU #3, S/N 307, was hot-fired at the pad prior to launch. Additionally, a final GGVM liquid leak check was performed at the pad to verify SOV integrity prior to launch. What caused the SOV to open is still unidentified. Kennedy Space Center (KSC) documented this problem as an unexplained anomaly; it was closed prior to STS-39 flight.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3 (Continued)	STS-37/OV-104 APU #3, S/N 307, uncommanded GGVM SOV motion.	There were no similar uncommanded APU GGVM valve movements during STS-39/OV-103 turnaround testing. All STS-39/OV-103 APUs flew on the previous mission with no similar anomalies. Rationale for STS-39 flight was: • There were no abnormal APU GGVM conditions identified for any APU installed on STS-39/OV-103. • Operational Maintenance Requirements and Specifications Document (OMRSD) liquid leak checks were performed on all STS-39/OV-103 APUs at the pad, following fuel loading, to verify GGVM valve integrity prior to launch. ***This risk factor was resolved for STS-39.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Lack of weld penetration on STS-39/OV-103 Reaction Control System (RCS) thruster R3A, S/N 218. HR No. ORBI-119 {AR} *There were no RCS thruster failures on STS-39 due to lack of weld penetration. There was, however, an RCS thruster and fuel injector temperature anomaly on STS-39 that was caused by a sensor or instrumentation error (IFA No. STS-39-V-03). This anomaly was not related to the weld defect issue.*	Ultrasonic inspection of all primary thrusters for weld defects was mandated following the thruster weld investigation in early 1990 [see STS-36 Mission Safety Evaluation (MSE), Postflight Edition, January 15, 1990, Section 4, Orbiter 14 for further details]. During planned ultrasonic inspection of RCS thruster R3A, S/N 218, a lack of weld penetration was discovered in the chamber-to-nozzle joint. The weld defect was over the entire circumference of the joint, approximately 0.120" in depth. Only 20%, or 0.013" weld penetration was estimated to exist. During the 1990 weld investigation, it was determined that, with the manufacturing process and controls in place, the worst-case weld penetration would be 30%. S/N 218 was removed from OV-103 and shipped to the vendor, Marquardt, for further investigation. Ultrasonic inspection of all other OV-103 thrusters was completed with no other weld penetration problems cited. Upon return of thruster S/N 218 to Marquardt, a double-wall x-ray was taken of the suspect joint and compared to an x-ray taken in 1981. Similar indications were found in both x-rays. The S/N 218 investigation at Marquardt also revealed that a Material Review Board (MRB) action was taken in 1981 due to porosity observed in the weld joint. MRB disposition directed removal of this porosity from the weldment. An engineering decision was made, following removal of the porosity, not to fill the resulting cavity because it was believed that sufficient material was left to ensure structural integrity. However, by not filling the cavity, raised areas were created on each side. During the recent investigation at Marquardt, the raised areas were machined flush with the weld joint. A subsequent ultrasonic test

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4 (Continued)	Lack of weld penetration on STS-39/OV-103 RCS thruster R3A, S/N 218.	revealed no discontinuity as previously seen. A piece was then cut from the weld area to determine the extent of weld penetration. This destructive examination determined the weld penetration to be within specification. Rationale for STS-39 flight was: • RCS thruster S/N 218 was replaced with a thruster that passed ultrasonic inspection. All other OV-103 primary thrusters passed the ultrasonic inspection criteria. • It was established during the 1990 weld investigation that the failure consequence of a 30% or greater weld penetration was a leak, not a total failure or burst. The thruster wire wrap would shut down a thruster with a weld leak in sufficient time to prevent further damage to the leaking thruster or adjacent thrusters. • Shutdown and deselection of a primary thruster were operationally acceptable due to RCS thruster redundancy. ***This risk factor was resolved for STS-39.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	Potential for lube oil leak on STS-39/OV-103 APU #2, S/N 301, gearbox. HR No. INTG-072A {C} INTG-149A {AR} ORBI-036 {AR} ***During STS-39 entry, APU #2, S/N 301, lube oil outlet pressure was low, reading at 25 psia; nominal outlet pressure is 40 to 50 psia (IFA No. STS-39-V-11). Visual inspection of the aft compartment at KSC showed no indication of a leak.*** ***This was the first APU lube oil pressure-low anomaly in the Program. Previous lube oil anomalies were related to high lube oil outlet pressure. It was concluded that the low oil pressure was due to low oil quantity, which occurred prior to flight during Orbiter Processing Facility (OPF) ullage check processing.***	Lube oil was observed around the gearbox seam on STS-39/OV-103 APU #2 during lube oil drain operations. The exact source of the oil was unknown; it was presumed to have leaked from the gearbox. A gearbox leak is a Crit 3 failure mode. There are 2 catastrophic failure modes attributed to gross leakage. The first, a Crit 1R2 failure mode, has the potential for loss of the vehicle if the grossly leaking oil ignites. The second, a Crit 1/1 failure mode, can occur when gross lube oil leakage leads to premature APU shutdown during an SSME-induced abort, resulting in loss of throttling capability. There were no signs of lube oil leakage on the other OV-103 APUs. Initial troubleshooting included wiping the gearbox seam clean and pressuring the gearbox to 50 psia (greater than 2 times the gearbox operating pressure). Post-pressurization inspection found slight traces of lube oil. The gearbox was serviced and again pressurized to 50 psia. Inspection found no new signs of lube oil leakage after 24 hours (hr) of pressurization. There was a minor pressure drop (less than 2 psia) during this period; however, this was probably due to pressure release through the seal cavity drain vent. After several days at 50 psia, there were no signs of gearbox leakage. The APU gearbox condition was reviewed by the MRB and approved for use "as is". Postflight inspection of the gearbox seam was required by the MRB. Gearbox pressure is continuously monitored prior to APU start in accordance with the Launch Commit Criteria (LCC). During this period, APU gearbox pressure must be less than 30 psia and greater than 6 psia to allow APU start.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	Potential for lube oil leak on STS-39/OV-103 APU #2, S/N 301, gearbox.	The probable cause of the visible lube oil leakage was spillage during lube oil drain operations. The APU lube oil fill and drain Quick Disconnect (QD) is located near the gearbox seam. Minor spillage during drain operations would result in the potential for depositing lube oil on the exterior of the gearbox; most likely in the proximity of the gearbox seam. Rationale for STS-39 flight was: • Extended pressure testing well above the gearbox operating pressure resulted in no signs of further lube oil leakage. • The potential exists for spillage during nominal lube oil drain operations. • The gearbox was approved for flight by MRB action. ***This risk factor was acceptable for STS-39.***
6	New Data Processing System (DPS) failure could lead to a Crit 1/1 condition. ***No similar anomalies were reported on STS-39.***	During testing in the SAIL on January 12, 1991, an operator noticed that the left keyboard entry was displayed simultaneously on Cathode Ray Tube (CRT) #1 and CRT #3. Because the select switch was in the CRT #3 position, the keyboard entry should only have been displayed on CRT #3. The operator cycled the select switch several times; the problem was corrected.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	New DPS failure could lead to a Crit 1/1 condition.	Troubleshooting determined that the anomaly could not be isolated to a hardware problem, because the anomaly repeated with different SAIL hardware. Potential hardware failure modes that could cause the same effects were identified: • CRT select switch, Part Number (P/N) ME452-0102-7201, contact short-to-ground or contact-to-contact short. • Data Entry Unit (DEU) receiver hybrid output fails low. There were no other relevant CRT select switch, P/N ME452-0102-7201, failures in the problem data base. There was a toggle switch anomaly similar to this failure on STS-41 (see Section 6, Orbiter 4); however, the switch was a different dash number. The STS-41 anomaly was attributed to a particle large enough to cause a contact-to-contact short. There have been 2 previous failures of DEU receiver hybrids, P/N 6088602; however, both were due to manufacturing defects and were considered isolated occurrences. The existing switch Failure Modes and Effects Analysis (FMEA) lists this type of failure as Crit 1R3. This is based on 2 keyboard entries causing an unpredictable BFS pre-engage response during ascent and entry. An assessment of displays presented during nominal ascent and entry revealed no SPEC/keystroke combinations that could result in a Crit 1/1 scenario; therefore, the current FMEA is applicable. However, further assessment of all available vehicle displays identified several SPEC and single- or double-keystroke execute combinations that could lead to a Crit 1/1 scenario. These displays would only be presented in

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)	New DPS failure could lead to a Crit 1/1 condition.	off-nominal conditions. Because of this potential, a CIL waiver for the Crit 1/1 conditions was prepared for STS-39. The CRT select switch FMEA was upgraded to a Crit 1/1 CIL for STS-37. Additionally, there is a potential for a new Hazard Report to baseline the associated risks. Rationale for STS-39 flight was: • The hardware involved, the CRT select switch and DEU receiver hybrid, had demonstrated high reliability. • The potential for a Crit 1/1 failure exists only in off-nominal scenarios. • All SPEC/keystroke combinations were identified. • The STS-39 crew was trained to check for keyboard entries displayed simultaneously on 2 CRTs when entering commands. ***This risk factor was acceptable for STS-39.***
7	Potential for new GPCs to erroneously overwrite memory. HR No. ORBI-194 {AR} ***No similar GPC anomalies were reported on STS-39.***	At the Level III Orbiter Flight Readiness Review, IBM reported a generic hardware problem with the new GPCs. The problem occurs when the transition is made from halt (also known as sleep or freeze-dried mode) to an operational mode. The GPC can cause random locations of memory to be overwritten. The problem was first seen with the Shuttle Mission Simulator (SMS); however, it was believed to be unique to the SMS. A hardware design change to a single page in the GPC is in process to fix this problem. Access to the GPCs in the horizontal position is required to implement this change.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)	Potential for new GPCs to erroneously overwrite memory.	The halt position on the AP-101S (new GPC) is known as the sleep mode. During the sleep mode, minimum power is drawn by the GPC to retain the memory (approximately 56 watts vs 560 watts nominal). Since the new GPC has a Complimentary Metal Oxide Silicon (CMOS) memory that would be lost if power is removed, Flight Rule 7-30 B was added for this first flight of the AP-101S GPCs. It states that "...Since removing power from the 101S GPC causes loss of memory integrity, a 101S GPC will only be powered off under extreme powerdown conditions." However, discovery of the new memory overwrite potential was a deterrent from putting 2 or 3 of the GPCs in the sleep or freeze-dried mode on orbit as was planned. The possible workarounds proposed for STS-39 included: • Keeping the GPCs in the standby mode, instead of the halt or sleep mode, so that a power transition is not required • Performing Initial Program Load (IPL) from the Mass Memory Unit (MMU) on any GPC undergoing power transition from the sleep mode (i.e., at power-up). The first workaround would eliminate the concern for memory overwrites but would consume more power than planned. It was planned to put 3 GPCs in the sleep mode when on orbit: the Backup Flight System (BFS) computer and 2 of the 4-GPC redundant set. Only 2 Primary Avionics Software System (PASS) computers were nominally planned to run on orbit to conserve power and enable the crew to freeze-dry the G2 [on orbit Guidance, Navigation and Control (GN&C)] on 1 GPC and the G3 (entry software) on the other sleeping GPC. In the process used to

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)	Potential for new GPCs to erroneously overwrite memory.	freeze-dry an AP-101S GPC, the GPC is loaded with the desired software configuration (G2 or G3 in this case) and put into the sleep mode. The reason for freeze-drying is to have a copy of critical software available without requiring access to MMUs. Human factors and timing are of concern to performing IPL at GPC power-up. While IPL is a viable workaround in any case, crew intervention is required for this workaround option. Because of this problem, a decision was made to configure the STS-39 GPCs on orbit as follows: • GPC 1 had the on-orbit GN&C software. • GPC 2 was put in the sleep mode when not in use. The crew powered-up GPC 2 and performed an IPL to load the redundant GN&C software for the planned rendezvous operations. • GPC 3 was freeze dried for G3 software and kept in standby mode. • GPC 4 had the Systems Management (SM) software. • GPC 5 was reserved for the BFS and was put in the sleep mode on orbit. The crew performed an IPL to load the BFS for deorbiting operations.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)	Potential for new GPCs to erroneously overwrite memory.	Rationale for STS-39 flight was: • There was a plan in place to alleviate the need to power-down certain GPCs on orbit. • If unplanned or inadvertent power loss occurred, the crew could perform the required IPL to reconfigure the GPCs. ***This risk factor was resolved for STS-39.***
8	Cracked lugs in STS-39/OV-103 Orbiter/External Tank (ET) umbilical door mechanism. HR No. ORBI-302A {AR} ***No anomalies attributed to the Orbiter/ET umbilical door mechanism were experienced on STS-39.***	During borescope inspection of STS-39/OV-103 Orbiter/ET umbilical door cavity thermal curtain installation, a crack was found on the Right-Hand (RH) door (LO_2-side) aft bellcrank hinge lug. This crack was 180° around the lug clevis, with an approximate displacement of 0.125". The crack was smeared from wiping, indicating that it may have existed for several door cycles. This finding led to borescope inspection of the RH forward lug and both Left-Hand (LH) lugs. The RH forward lug examination found a crack along the front surface, with no displacement. The LH forward lug was found to have a crack approximately 2" long, with an approximate displacement of 0.060". The LH aft lug showed no signs of cracking. Borescope and dye-penetrant inspection of OV-102 lug clevises found surface cracks (with no displacement) on the LH forward and aft lug clevises and on the RH aft; the RH forward lug showed no signs of cracks. Similar inspection of OV-104 revealed crack initiation on 3 of 4 lug clevises; the fourth lug clevis had indications of pitting.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)	Cracked lugs in STS-39/OV-103 Orbiter/ET umbilical door mechanism.	The concern was the potential for either Orbiter/ET door not completely closing for reentry due to failure of one or more hinge lugs. This could lead to thermal and structural damage to the aft compartment and result in loss of the crew and vehicle. This is a Crit 1/1 failure mode. Analysis determined that the initiating factor for this condition was Low-Cycle Fatigue (LCF). LCF, coupled with a stress riser at the lug clevis resulting from the original manufacturing process, can lead to initiation of small cracks in the clevis. However, it is not believed that these factors could result in the displacement observed on the OV-103 LH forward and RH aft lugs. Analysis determined that a single overload event would be required to cause lug displacement due to crack initiation. An overload event that would lead to displacement on the OV-103 RH aft lug could not be identified. There was the potential, however, that cracking could have resulted during final latch-up of the doors in the open position for flight. The procedure for latching the doors in the open position requires manual force to be applied to allow attachment to the centerline latches. This required force, not to exceed 210 pounds (lb), applies the highest tensile loads to the lug. Review of processing records and discussions with technicians did not uncover reports of anomalous noise or conditions that could have induced additional stress to the lug. It is not certain that a single overload event leading to displacement of the RH aft lug will ever be isolated.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)	Cracked lugs in STS-39/OV-103 Orbiter/ET umbilical door mechanism.	The cracking and displacement observed on the OV-103 LH forward lug was caused by an inadvertent action during processing. While the OV-103 ET doors were locked in place by the centerline latches, power was inadvertently applied to the LH ET door actuator causing it to momentarily apply increased stress on the LH forward lug. Interviews with technicians performing the work at that time have determined that they heard a loud "pop" shortly after power was applied. It was believed, therefore, that the additional stress induced to the lug caused the lug to crack and displace. In addition to inspecting all vehicles, a plan was formulated to perform tests on OV-102 with a modified bolt in the forward and aft lugs. The bolt was machined to reduce the diameter by 1/8" to simulate the condition that would result if the lug failed completely. Repeated cycling was performed, per normal OMRSD procedures, to determine the net effect. The results of the test indicated no adverse effects with the modified bolt installed, and all OMRSD requirements were met. Rerigging of the door actuator mechanism was not needed to meet the OMRSD requirements. The force applied to latch the door in the open position did not exceed 169 pounds force (lbf) during repeated attempts. Observation of the modified bolt during this test found minimum movement. After extensive review by the Space Shuttle Program community, and because the overload event leading to the displacement of the RH aft lug could not be identified, the decision was made to roll STS-39 back to the OPF and replace the ET door lug housing. Original plans called for replacing the OV-103 lug housings

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)	Cracked lugs in STS-39/OV-103 Orbiter/ET umbilical door mechanism.	with the OV-105 housings. However, because a starter crack was found on 1 of the 4 OV-105 lug clevises during receipt inspection at KSC, the decision was made to install the modified OV-102 lug housings which had recently been returned to KSC. While inspecting the OV-103 ET door assemblies at KSC, it was discovered that, on the RH aft housing, the splined shaft index mark was not correctly marked on the part, resulting in the push rod adjustment being adjusted to its full available travel. The spline alignments were off in 2 places by 1 tooth each. These misalignments compensated for each other and resulted in overall alignment within specification. Physical measurements of push rod "F" data indicated that the door was effectively aligned, and no action was taken prior to launch. Inspection of the other 3 OV-103 ET door assemblies was completed with no other index problems found. Subsequent to this inspection, the drawings and manufacturing paperwork for the OV-103 RH aft housing were reviewed. The original assembly drawing for the 2 aft ET housings was found to be missing a general note stating that the index marks on the bell cranks and the shaft must be aligned. The original assembly drawing for the forward ET housings contained such a statement. An engineering change was issued in June 1984 to add this statement to the aft housing drawing; however, the OV-103 aft housings were manufactured in October 1982.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8 (Continued)	Cracked lugs in STS-39/OV-103 Orbiter/ET umbilical door mechanism.	Manufacturing records for the RH aft housing were reviewed and found to be missing a sequence to align the index marks. Government inspection occurred only at the last manufacturing sequence. No in-process Government inspection was performed. The paperwork indicated that the RH housing was assembled on October 1982 and never disassembled. Therefore, the housing must have been assembled incorrectly at that time. Rationale for STS-39 flight was: • A modified ET door lug housing was installed on STS-39/OV-103. • The OMRSD adjustment, including ET door cycling, was completed with no anomalies. ***This risk factor was resolved for STS-39.***
9	Cabin pressure bleed valve anomalies on OV-105. HR No. ORBI-074 {C} ***No cabin pressure bleed valve anomalies were reported on STS-39.***	Three cabin pressure bleed valves installed on OV-105, S/Ns 5, 6, and 8, recently failed leak tests. A fourth, S/N 7, was inspected at the manufacturer, Charlton Technologies, Inc. (CTI), and was found to be leaking; however, the leak was within specification. All 4 valves exhibited signs of seal debonding. Valves S/N 1 through S/N 4 had no history of leakage or seal debond. The cause of the debonding was not determined. The bond could be affected by either a lack of primer on the valve or incorrect application of primer to the valve. Poor adhesion of the S/N 5 through S/N 8 molded seals, and seal leakage, was verified at CTI. Testing to date has

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
9 (Continued)	Cabin pressure bleed valve anomalies on OV-105.	been inconclusive in determining proper primer application. Records showed that the seals were molded in accordance with the applicable procedure; however, there was no specific inspection criteria for primer application. A historical comparison of seal material batches to seal leakage indicated no definitive correlation. All seal materials are batch tested for correct material properties. For flight vehicles, cabin pressure bleed valve leak tests at 15 pounds per square inch differential (psid) are performed every 5 flights. Maximum allowed leakage is 25 standard cubic centimeters per minute (sccm). During countdown, cabin pressure drop is monitored, with a 2-psid gross leak limit. Each valve is checked individually after venting the cabin; there is no reverification of leakage. OV-103 valves were last leak tested before STS-41, with no problems noted. OV-104 valves were checked during the STS-37 flow, with no anomalies. Rationale for STS-39 flight was: • OV-103 cabin pressure bleed valves passed all leak tests. • LCC and Flight Rules were in place to account for valve leakage after turnaround testing. ***This risk factor was resolved for STS-39.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10	Main Propulsion System (MPS) 3-way helium solenoid valve failed leak tests on OV-102. HR No. ORBI-108E {AR} ORBI-129A {C} ***No MPS helium leaks were experienced on STS-39.***	MPS leak tests on OV-102, after STS-35, isolated a leak to a 3-way helium solenoid valve (LV68) used in the 17-inch disconnect latch unlock. Additional tests confirmed the leak and further isolated it to a crack in the valve bellows. Similar 3-way helium solenoid valves are used in 46 locations in the MPS. The valves control helium pressure to open or close pneumatically-operated MPS valves. The concern was that worst-case failure of a helium valve could lead to helium leakage or valve rupture and result in a Crit 1 failure. This failure scenario was believed to potentially deplete the onboard helium supply, resulting in inability to close prevalves at MECO and the potential for main engine turbopump overspeed and explosion. Valve LV-68 was removed for evaluation and replaced. Retests indicated that the replacement valve was not leaking. Helium solenoid valve bellows are 2-ply (nickel and copper plies) and are fabricated by an electroforming/electroplating process. Convoluted plies are soldered to end fittings to complete the bellows assembly. In normal operation, the valve bellows assembly is pressurized by solenoid inlet pressure. Internal bellows pressure and spring rate provide the forces necessary to maintain the valve in the closed position when the solenoid is deenergized. Bellows assemblies are proofed at 1550 psi, more than twice the operating pressure. Bellows assemblies are reproofed after solder rework; therefore, a valve could be subjected to multiple proof-pressure tests. Initial teardown and inspection of the failed LV-68 valve identified a deformation, or squirm, in the bellows that was caused by buckling instability. Further examination found a circumferential crack on a convolute crown in the bellows.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10 (Continued)	MPS 3-way helium solenoid valve failed leak tests on OV-102.	The crack was approximately 0.150 inch long and was on the tension side of the squirm. Leak checks measured the leak rate to be 22 standard cubic feet per minute (scfm) at 300 psi. Extrapolation to the 750-psi operating pressure predicted the leak rate to be 54 scfm. Maximum leakage allowable is 10 sccs from the valve vent port. Metallurgical analysis determined that the crack was 80% through the inner bellows ply and was due to fatigue. Final separation in each ply was determined to be due to overload. The crack coincided with a small void in the copper strike. Other cracks were discovered in adjacent convolutes. These cracks propagated from the inner diameter toward similar voids in the copper strike. Stress analysis performed by the vendor indicated that the bellows design was marginal for the 1550-psi proof test. Predictions were that the internal pressure required to initiate squirm deformation is 1551 psi. A more conservative analysis indicated that the pressure needed to initiate squirm was much lower than 1500 psi. An evaluation of potential effects of a squirmed bellows on valve operation is underway. The investigation into this failure considered the potential for a lot-related failure mode. The failed bellows was manufactured in 1987 in a lot of 7 bellows; a startup lot after a 5-year layoff. Records indicated that the failed bellows, and all other bellows in the lot, were subjected to 3 proof tests because of required solder rework. A second bellows from the lot of 7 was installed in the LO_2-side outboard fill and drain valve on OV-103. The outboard fill and drain valve on OV-103 was

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10 (Continued)	MPS 3-way helium solenoid valve failed leak tests on OV-102.	replaced prior to STS-39 flight. Of the 5 remaining bellows from the lot, 4 were scrapped, and the last was rejected at KSC after the valve failed helium signature tests. This valve too showed signs of squirming. The investigation also determined that solenoid valve certification tests were inadequate for verifying bellows life. Certification testing included 12,000 valve operation cycles; however, only 50 pressure cycles were performed. Pressure cycling was believed to be the primary contributor to bellows squirming and fatigue. There were no other solenoid valve bellows failures recorded during flight or ground checkout. This history included a significant number of valve cycles and operations. Bellows problems were, however, encountered during the production of the OV-105 bellows assemblies in 1989. In this case, initial production runs were scrapped. OV-105 bellows failures were not the result of cracked bellows; however, the bellows were found squirmed and dimensionally unstable after repeated proof-pressure tests. All OV-105 bellows were reworked because of solder problems induced by numerous process and personnel deficiencies at the vendor. Prelaunch and operational procedures are in place to control potential helium leaks through the valve bellows. Prior to launch, excessive helium loss would be detected by the HGDS. If aft compartment helium concentrations exceeded the 10,000-parts per million (ppm) LCC limit, the launch would be scrubbed. After launch, helium tank pressures are monitored by the Caution and Warning System (CWS). An alarm would sound if helium tank pressures drop below 3,800 psi. If this was to occur during ascent, the crew would be required to manually close isolation valves

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
10 (Continued)	MPS 3-way helium solenoid valve failed leak tests on OV-102.	LV7 and LV8 to conserve helium. LV7 and LV8 would be reopened along with LV10 (engine crossover) at MECO minus 30 seconds (sec). These actions are intended to conserve sufficient helium stores to effect engine prevalve closure at MECO. JSC analysis determined that the worst-case leakage in flight due to a ruptured bellows is manageable. The leak rate, restricted by a 0.0930-inch diameter passage in the valve, was calculated to be 260 scfm. This leak rate would not deplete the helium supply during ascent; therefore, sufficient helium would remain in the system to shut engine prevalves at MECO. Additionally, a 260-scfm leak would not provide sufficient helium to overpressurize the aft compartment. Rationale for STS-39 flight was: • The 3-way helium solenoid valves had a highly-reliable history prior to the discovery of cracked bellows on OV-102. This history included 46 valves installed and operated on 37 Shuttle missions. • The bellows from the suspect lot installed on OV-103 were removed and replaced.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10 (Continued)	MPS 3-way helium solenoid valve failed leak tests on OV-102.	• Processing flow leak checks and prelaunch HGDS monitoring have the capability to identify leaks. If a leak were to occur after launch, CWS monitoring would alert the crew to take action to conserve the helium supply. ***This risk factor was acceptable for STS-39.***
11	Gaseous Hydrogen (GH_2) Flow Control Valve (FCV) weld crack found on OV-103. HR No. ORBI-306 {AR} ***No GH_2 FCV anomalies were reported on STS-39.***	During STS-39/OV-103 preparations, a small leak was detected at the engine #1 GH_2 FCV housing during mass spectrometer leak tests of the OV-103 GH_2 pressurization system. These tests were performed as part of the investigation into the high Hydrogen (H_2) concentration measured in the OV-103 aft compartment during STS-41 ascent. The leak was measured at 2.3×10^{-6} sccs, in excess of the 1×10^{-6} sccs specification limit. Initial calculations indicated that this leak rate was sufficient to account for the H_2 concentrations measured during STS-41; however, a more formal calculation of potential leak rates is in work. Analysis indicated that a worst-case leak through a circumferential crack would not provide sufficient H_2 to reach aft compartment flammability concentrations. Initial examination of the FCV found the leak source to be a 3/8" crack in the FCV housing outlet tube weld. This weld is a sealing weld only and provides no structural integrity. The FCV was removed and sent to Rockwell International (RI) for analysis. The replacement FCV was installed and passed all mass spectrometer leak tests.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)	GH_2 FCV weld crack found on OV-103.	Examination of the cracked FCV housing outlet tube weld at RI indicated the weld to be of good quality (good penetration, no evidence of material defect, and good weld blend). Structural analysis determined the failure mechanism to be High-Cycle Fatigue (HCF). The source of the fatigue was not determined; however, loads induced in the high-vibration environment was the leading candidate. Investigation into potential vibration sources is underway. Evaluation of GH_2 and Gaseous Oxygen (GO_2) qualification FCV housings for similar fatigue conditions is in work. Leak tests and Nondestructive Evaluation (NDE) methods will be employed on the qualification housings. Rationale for STS-39 flights was: • Mass spectrometer and visual inspections found no indication of leaks in OV-103 FCVs. • A worst-case circumferential crack would not provide sufficient H_2 to reach aft compartment flammability concentrations. ***This risk factor was resolved for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
12	New GPC, AP-101S, failure mode. ***No GPC sleep mode problems were reported on STS-39.***	GPC S/N 507 failed during operation in the Shuttle Avionics Integration Laboratory (SAIL) in June 1990. S/N 507 was reported to be unable to recover from the sleep mode. Power cycling restored GPC S/N 507 to normal operation. This anomaly was reported on Corrective Action Report (CAR) #7274. S/N 507 was removed from the SAIL and returned to IBM for troubleshooting. Troubleshooting at IBM could not repeat the failure during vibration and thermal testing. IBM was also unsuccessful in isolating the failure using Automatic Test Equipment (ATE). The sleep mode logic was analyzed for a potential design flaw. This analysis resulted in identification of the design problem addressed in Section 4, Orbiter 7. Because no failure could be isolated at IBM, S/N 507 was returned to the SAIL and operated for over 500 hr without a repeat of the earlier failure. A recent review of SAIL videotape showed that the S/N 507 mode switch was in the run position instead of the sleep position when the June 1990 failure occurred. This finding gave rise to the potential for a generic power supply problem in the AP-101S GPCs. S/N 507 will be returned to IBM for additional testing and troubleshooting that will focus on GPC power supply operation and functionality. If the failure cannot be duplicated or isolated at IBM, S/N 507 power supplies will be removed from flight status. There were 2 previous AP-101S power supply failures; both were in qualification units. These 2 qualification failures had different causes and effects; both resulted in design or procedural fixes.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
12 (Continued)	New GPC, AP-101S, failure mode.	Rationale for STS-39 flight with the new GPCs was: • There was no failure history of new, flight-unit GPC power supplies. • New GPCs installed on OV-103 were extensively tested with no similar problems experienced. ***This risk factor was resolved for STS-39.***
13	STS-37/OV-104 short touchdown on Edwards Air Force Base (EAFB) lakebed runway. HR No. ORBI-325B {AR} ***STS-39/OV-103 experienced no similar problems during landing at KSC.***	During the April 17, 1991 System Safety Review Panel (SSRP) telecon, STS-37/OV-104 *(Atlantis)* was reported to have touched down more than 600-ft short of the threshold on EAFB lakebed runway #33. Surface winds at touchdown were 8° at 17 knots (kt), gusting to 21 kt. The headwind component was 10 kt gusting to 13 kt, and a right crosswind component was 14 kt gusting to 18 kt. Touchdown velocity was 166-kt equivalent airspeed at 623 ft with a 5 ft/sec descent rate. During descent, the Orbiter encountered wind shear from 13,000 ft down to 9,000 ft. Wind shear was again encountered at approximately 1,000 ft. The second wind shear encounter resulted in an 850-ft decrease in range and an airspeed loss of approximately 20 kt. It is believed that both wind shears contributed to the short landing. Energy at the Terminal Area Energy Management (TAEM) interface was nominal. A high tailwind was experienced when entering the Heading Alignment Cone (HAC); however, this tailwind was well within acceptable criteria and vehicle performance parameters. The HAC displayed to the crew was automatically recomputed at 23,000 ft by onboard guidance, causing the HAC to shrink.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
13 (Continued)	STS-37/OV-104 short touchdown on EAFB lakebed runway.	The crew attempted to fly a 60° bank, 1.6-g turn in response to this HAC. However, the crew was not told of existing, known wind shear conditions between 20,000 ft and 7,000 ft, the altitudes at which wind conditions are provided to the crew in accordance with the entry cue cards. Not knowing the actual wind shear conditions led to the inability to precisely fly the HAC maneuver and resulted in a wider than computed HAC and lower energy at the transition to the approach and landing interface. At preflare, the vehicle was approximately 1,000 ft below the reference altitude, resulting in 3,200 ft of range error. Encountering the second wind shear at approximately 1,000 ft further reduced the range by 850 ft. The crew used all remaining energy to protect against a hard landing and high nose gear slapdown loads. Actual touchdown sink rate was 5 ft/sec, and nose gear slapdown was within the nominal range. Using a lightweight vehicle, high headwind, and a cold atmosphere, real-time landing support personnel predicted the OV-104 touchdown point to be 1,700 ft at 195 kt; nominal touchdown is 2,500 ft at 195 kt. Postflight simulation of the actual conditions indicated that the go/no-go landing redline was exceeded by a small amount. Postflight review revealed that information gathered by the Shuttle training aircraft could have mitigated wind shear effects. An effort is underway to review all communications procedures, including entry cue cards, to determine what improvements, if any, are required. This incident has elevated the awareness of potential wind shear conditions, and the potential effects wind shear has after passing the TAEM interface.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
13 (Continued)	STS-37/OV-104 short touchdown on EAFB lakebed runway.	Review of wind shear effects during other flight regimes, including heavyweight landings and abort sites, was performed. All primary abort sites have adequate runway margins (aprons), landing aids, and weather forecasting equipment to ensure safe landings. Current onboard guidance provides adequate landing margin for heavyweight vehicles, and flight rules are in place to ensure optimum margin on short runways. Underrun aprons on all primary landing and abort sites are a minimum of 1,000 ft in length. Transatlantic Abort Landing (TAL) site underrun aprons are walked down for debris inspection prior to potential use. STS-39/OV-103 was a heavyweight landing vehicle; 20,000-lb heavier than STS-37/OV-104. Simulation, using the actual parameters from STS-37, indicated that STS-39 could make the runway; however, the nose gear slapdown loads would be in the 3-sigma range. The STS-39 crew was briefed on the circumstances surrounding the STS-37 incident and performed simulated landings under the STS-37 conditions prior to launch. Additionally, calls were made to the crew concerning projected g-forces in the HAC and wind shear conditions.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
13 (Continued)	STS-37/OV-104 short touchdown at EAFB lakebed runway.	Rationale for STS-39 flight was: • Flight rules and current landing criteria were found to be adequate, and no changes were made for STS-39. • Communications procedures were updated prior to launch. • Crew training emphasized the importance of following command steering cues from the auto-guidance computer under marginal weather conditions. • For additional awareness, the STS-39 crew was briefed concerning the circumstances surrounding the STS-37 incident, and the corrective action to take should these circumstances recur. ***This risk factor was acceptable for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	HPOTP first-stage turbine disc cracking. HR No. ME-C1 (All Phases) {AR} ***No SSME anomalies attributed to HPOTP turbine disc cracking were reported on STS-39.***	Dye-penetrant inspection of 4 HPOTP first-stage turbine discs identified radial cracks in the interstage pilot rib. Gradient oxide discoloration was found in 2 cracks. These cracks were not obvious or detected prior to removal of the gold plating. The high-time HPOTP, where these cracks were first found, was the fleet leader with 21,908 sec and 52 starts; it had been removed from the flight program for a long time. Sixteen turbine discs were inspected to date; 7 were found with radial cracks in the interstage pilot rib. Materials and processing analysis determined that the cracks initiated midspan in the disc and extended either to the outboard or inboard corner of the pilot rib. Scanning Electron Microscope (SEM) inspection of the fractures indicated a brittle crystallographic appearance. The fracture mode showed the effects of H_2 influence, indicating probable LCF or sustained load crack propagation. Structural analysis indicated a cyclic strain range, overwhelmingly dominated by thermal shock at shutdown, caused by H_2 cooling of the hot disc. Peak strain was determined through tests to follow a minimum of 40 to 100 sec of operation, or when the disc reached steady-state high operational temperature. Evaluation of the correlation of LCF analysis to this failure mode indicated that the worst-case thermal shock strain range is insufficient to result in cracking without H_2 embrittlement. Deviation Approval Request (DAR) #2474 for fatigue damage ratio was reevaluated for STS-39/OV-103 HPOTP first-stage turbine discs. To date, the lowest damage ratio for a disc found with radial cracks is 1.0. All previously-flown discs had a damage ratio margin >4. This was true for 2 of 3 discs on STS-39. However, the damage ratio of turbine disc S/N 2702270, engine #2026,

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1 (Continued)	HPOTP first-stage turbine disc cracking.	HPOTP 2226R3, was calculated to be 0.258 post-STS-39 flight. Because the damage ratio margin was less than 4, the DAR #2474 limit was increased to allow a postflight damage ratio of 0.27. This action cleared turbine disc S/N 2702270 for flight. Continued evaluation of the DAR limit for turbine discs on future missions led to an additional change in the criteria. Rocketdyne updated the previous DAR to deal with HPOTP turbine disc cracking by utilizing a statistical analysis method vice the damage fraction method. This revision changed the DAR criteria from 0.27 damage fraction in the interstage seal pilot rib location to a life-limit of 14 starts (the HPOTP/turbine disc must be operated in excess of 20 sec to be counted as a start). This DAR life-limit was based on a fleet leader approach used for other SSME components. The 14-start life-limit is half the lowest number of starts (29) for a turbine disc found with cracks. Fleet history showed 35 discs with greater than 14 starts, and 20 discs with greater than 28 starts. Discs examined to date with 12, 15, 18, and 19 starts showed no cracks. Turbine discs in STS-39/OV-103 HPOTPs had no more than 9 starts. Rationale for STS-39 flight was: • HPOTPs on OV-103 did not exceed the previously approved 0.27 damage factor DAR limit after STS-39. Additionally, OV-103 HPOTPs did not exceed the 14-start DAR limit for STS-39. • Fleet history encompassed all significant LCF variables. ***This risk factor was acceptable for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
2	High-pressure fuel duct flange radius cracking. HR No. ME-D3 (All Phases) {AR} ***No SSME anomalies associated with high-pressure fuel duct flange cracks were reported on STS-39.***	Sustained load cracking in the high-pressure fuel duct flange fillet radii was first identified in 1986. There are 2 flanges on the duct: at the F-4 joint, the interface between the duct and the HPFTP, and at the F-5 joint between the duct and the main fuel valve interface. The basic cracking mechanism was attributed to hydride formation from hydrogen diffusion within the basic 5Al-2.5Sn titanium material in regions of high sustained tensile stress. Sustained tensile stress was reduced by lowering the preload at each joint. Recently, cracks were found in a high-pressure fuel duct flange that was installed with lower preload. These cracks were found during post-proof-test inspection after the duct insulation system was repaired. OMRSD requirements direct preflight and postflight dye-penetrant examination to determine if high-pressure fuel duct flange cracking exists. Preflight examination must occur within 45 days of flight. The 45-day interval is based on maximum sustained crack growth for 90 days and engine testing with cracked ducts. Crack propagation was not demonstrated during hot-fire testing of 4 high-pressure fuel ducts with flange cracks. Ducts with cracks identified during the preflight dye-penetrant examination are removed prior to flight. Based on the 1986 findings, the high-pressure fuel duct flange was redesigned to incorporate Inconel-718 material. High-pressure fuel ducts on engines #2026 and #2030 are Inconel-718. The duct on engine #2029 was of the old design and was dye-penetrant inspected prior to launch. No cracks were found.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
2 (Continued)	High-pressure fuel duct flange radius cracking.	Rationale for STS-39 flight was: • The new Inconel-718 high-pressure fuel ducts were installed on engines #2026 and #2030. • The 45-day dye-penetrant examination was adequate to detect cracks that could affect flange preload or duct performance. This examination was performed on the engine #2029 duct prior to launch; no cracks were found. • No adverse affects were anticipated based on hot-fire tests with cracked duct flanges, even if cracks had gone undetected. ***This risk factor was acceptable for STS-39.***
3	Suspected contamination of engine #2029 HPOTP. HR No. ME-C1 (All Phases) {AR} ***No SSME anomalies attributed to contamination of the engine #2029 HPOTP were reported on STS-39.***	During leak test operations of engine #2029 in the OPF, the potential existed for introducing contamination into the HPOTP of that engine. It was determined that an improper Ground Support Equipment (GSE) configuration was used for that test. As part of the MPS leak tests, tygon tubing from the Captive Air Vent (CAV) system is attached to the HPOTP intermediate seal drain line to vent helium outside the OPF in an effort to lower the helium background. Upon disconnection of the tygon tube, positive pressure was observed flowing from the CAV tube in the form of a mist (visible air). The visible air was originally reported to be a hydraulic fluid mist. Engines #2026 and #2030 were not connected to the CAV at any time since installation.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
3 (Continued)	Suspected contamination of engine #2029 HPOTP.	Swab samples taken from all 4 Liquid Oxygen (LOX) drain lines for Main Engine (ME)-3, engine #2029, did not indicate any hydrocarbon contamination (hydraulic fluid). All LOX drain lines on ME-3 were replaced. Rationale for STS-39 flight was: • Any contamination introduced into the HPOTP or engine would have been detected in the swab sample testing; none was detected. ***This risk factor was resolved for STS-39.***
4	Pogo standpipe thermal insulator may have been inadvertently removed during final engine assembly. HR No. INTG-005 {C} ***No SSME anomalies associated with the Pogo system were reported on STS-39.***	Rocketdyne recently disclosed that an engine assembly mechanic stated that he removed the thermal insulator (teflon sleeve) from the Pogo standpipe on engines #2028 and #0215. The mechanic stated that he thought the sleeve was a shipping protector. Subsequent inspection of the Pogo standpipe on engine #0215 confirmed that the teflon sleeve was not present. Early Pogo system development tests demonstrated the need for the standpipe thermal insulator to prevent Gaseous Oxygen (GOX) from condensing on the cold standpipe surface. Because of this finding, inspection of the STS-39 engine pogo standpipes was required. This inspection was completed; standpipe insulation was found to be in place on all OV-103 engines.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
4 (Continued)	Pogo standpipe thermal insulator may have been inadvertently removed during final engine assembly.	Rationale for STS-39 flight was: • Inspection verified that Pogo standpipe thermal insulation was properly installed on all STS-39/OV-103 engines. ***This risk factor was resolved for STS-39.***
5	HPOTP coolant weld leak. HR No. ME-C1 (All Phases) {AR} ***No SSME HPOTP coolant weld leaks were reported on STS-39.***	A 10-scim leak was discovered during an encapsulation leak test post-hot-fire on engine #2015. Investigation found a 0.25" crack in a coolant plug weld in the housing of HPOTP #4301R2. This was 1 of 4 plugs in the coolant circuit (mixed gas) for the downstream second-stage disc. HPOTP #4301R2 had accumulated 41 starts >4.2 sec and 21,628 sec. Flange stresses are predominantly driven by engine installation and engine mainstage cycles (pressure and external loads). The disc coolant circuit has 4 redundant paths. The second-stage disc has margin even with no coolant. The shaft coolant circuit is a single path. Exceeding the turbine seal redline is classified as Crit 1R. Mission duration hot-fire testing with no coolant was safely conducted. The 10-scim helium leakage due to the crack defect resulted in 15-scim hot-gas leakage to the Orbiter aft boattail (this does not violate the interface control specifications).

RESOLVED STS-39 SAFETY RISK FACTORS

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

SSME

5 (Continued) — HPOTP coolant weld leak

HPOTP #4301R2 main housing hot-fire history included 41 starts with operation >4.2 sec, and a total operating time of 21,628 sec. Hot-fire history for the HPOTPs installed on STS-39/OV-103 SSMEs was as follows:

Pump	Starts (>4.2 sec)	Time (sec)
2226R3	9	3,490
4506R1	11	4,474
4008R3	3	897

The coolant plug weld is a Class II weld; shrinkage is common to these welds. HPOTP #4301R2 housing plug welds 11 and 17 were reworked during fabrication. There was exposure to LCF, and the flange was exposed to high stresses. The plug environment also rendered it susceptible to H_2 embrittlement.

Rationale for STS-39 flight was:

- All STS-39/OV-103 HPOTPs were leak tested prior to flight (encapsulation); no leaks were found.
- STS-39/OV-103 HPOTPs had a safety factor >2 in starts and seconds relative to the failed unit.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
5 (Continued)	HPOTP coolant weld leak	• A leak of the size experienced, if developed in flight, has no significant effects. It is within the interface control specification requirements for aft leakage. Turbopump coolant loss would be negligible. ***This risk factor was resolved for STS-39.***
6	Engine #2011 found with a leak in the Main Combustion Chamber (MCC) Augmented Spark Igniter (ASI). HR No. ORBI-306 {AR} ME-B5 (All Phases) {C} ***No SSME MCC ASI anomalies were reported on STS-39.***	Postflight encapsulation tests on engine #2011 resulted in identification of a leak in the MCC ASI (MCC #4005). Encapsulation testing of all STS-41 SSMEs, including engine #2011, was directed due to the high H_2 concentrations in the aft compartment recorded by catch bottle during STS-41 ascent (see Section 6, Integration 2 for further details). Using sniffer and soap solution tests, the leak was isolated to braze joint #1 of the main injector ASI. A record review found that this braze joint was previously repaired during original MCC #4004 ASI manufacturing. The brazing process was improved after this original problem. The leak was measured to be 2.56 scim, below the specification limit of 6 scim. Calculations determined that this leak would result in a 1,000-scim leak during SSME operation. This operational leak rate was not large enough to provide the leak rate recorded during STS-41 ascent.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
6 (Continued)	Engine #2011 found with a leak in the MCC ASI.	Rationale for STS-39 flight was: • STS-39/OV-103 SSME MCC ASIs had no record of repair, and all were manufactured after brazing process improvements. • OV-103 SSMEs passed the encapsulation leak tests, with no greater than a 0.4-scim leak recorded. These SSMEs also passed all helium signature tests with no measurable leakage detected. ***This risk factor was resolved for STS-39.***
7	Potential for pad abort caused by purge check valve seat leakage. HR No. ME-A1S Rev. F {C} ***No SSME check valve anomalies were reported on STS-39.***	During a test firing of engine #0213 at Stennis Space Center (SSC), excessive MCC dome purge check valve leakage resulted in engine shutdown due to violation of the 50-psia ignition confirmation redline. Post-test investigation determined that the MCC dome purge check valve leakage was caused by an Inco-718 particle lodging on the valve seat. The particle, 0.054" x 0.040" x 0.02", resulted in a leak through the valve of 11.5 - 12.5 scfm at 25 pounds per square inch gage (psig). There were no similar leakages recorded in the flight program (132 engine starts), including 108 oxidizer purge check valves. However, this was the sixth occurrence during the ground test program (1808 tests). Records indicated that 2 cases were caused by Kel-F contamination from the MCC dome purge pressure-actuated valve (led to a pressure-actuated valve redesign), 2 were caused by contamination

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
7 (Continued)	Potential for pad abort caused by purge check valve seat leakage.	originating external to the engine, and the last 2 from contamination of an undetermined source. In the most recent case, the source of the Inco-718 particle is as yet unknown. Inco-718 is used in the helium precharge valve housing and in the fuel preburner purge pressure-actuated valve. The particle found did not resemble a machining chip or burr. The helium precharge valve was installed on test 904-087, 3 tests prior to the most recent engine #0213 test. The purge line was disconnected at the helium precharge valve during HPOTP replacement. Rationale for STS-39 flight was: • Existing SSME assembly/disassembly procedures at KSC protect the engines from contamination entry. All joints are cleaned before disassembly, and open ports are covered with Aclar film and polyethylene foam. • OV-103 SSME check valves successfully passed reverse seat leak checks after the last engine hot-fire. • There were no similar occurrences affecting the redline margins. • Worst-case failure is a pad abort. *This risk factor was resolved for STS-39.*

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
8	Potential for loss of SSME controller redundancy. HR No. INTG-165 {C} ***No SSME controller anomalies were reported on STS-39.***	During flight readiness testing of STS-37/OV-104, engine #2031, SSME controller Unit Number (U/N) F27, channel B halted when powered up. Analysis of the controller memory dump indicated an echo check failure after loading of the channel B output electronics storage register. A change in the storage register is indicative of a hardware failure. Repeated attempts to repeat this failure were unsuccessful. U/N F27 was removed from engine #2031 and replaced with U/N F29. This was the fifth storage register failure detected during ground checkout. There were no storage register-related failures during SSME starts for flight. The worst-case failure of this type results in loss of SSME controller redundancy. If this failure were to occur between SSME start and Solid Rocket Booster (SRB) ignition, the result would be a pad abort. SSME Controller U/N F27 was returned to Honeywell, the vendor, on January 31, 1991. Upon initial power-up, the failure repeated. The failure, however, did not repeat during subsequent cold starts, thermal cycles, or vibration testing. Additional destructive failure analysis was planned. Rationale for STS-39 flight was: • Checks at T–34 hours prior to launch verified all SSME controller hardware functions. • There were no related failures during SSME starts at launch. • Worst-case failure results in loss of SSME controller redundancy. ***This risk factor was resolved for STS-39.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
9	HPOTP second-stage turbine blades found rubbing coolant jet tubes on U/N 6009. HR No. ME-C1A Rev. F {AR} (All Phases) ***No SSME HPOTP anomalies attributed to rubbing of turbine blades were reported on STS-39.***	Normal disassembly of HPOTP U/N 6009 found that the second-stage blades rubbed 3 of 19 coolant jet tubes. Approximately 0.012" was removed from the rubbed tubes. There was believed to be no structural damage, and no effect on performance was noted. Only minor damage to the blade shank was noted. Scanning Electron Microscope (SEM) inspection of the blade found random surface cracks up to 0.005" long. Comparable cracks are usually experienced with the normal casting process. These cracks were not detectable with optics or IVc penetrant. There were multiple cases of second-stage turbine blade rubbing. Minimum clearance occurs between the disc stiffening rib and the turbine housing during the shutdown transient. The original height of the jets was a maximum of 0.012" above the print dimension. Worst-case interference with the jet is 0.014" and would be detected by turbine shaft micro-travel inspection. Structural analysis indicated adequate margin with the presence of rubbed blades. Rationale for STS-39 flight was: • Blade rubbing was considered benign; damage to the second-stage turbine blades was acceptable. • Cracks observed on U/N 6009 were less than the dynamic threshold flaw size. ***This risk factor was acceptable for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
10	Engine #0213 G-15 seal failed during ground testing at SSC. HR No. ME-D3C Rev. F {AR} ME-D3M Rev. F {AR} ***No SSME G-15 seal anomalies were reported on STS-39.***	During scheduled inspection of the Main Combustion Chamber (MCC) and nozzle on development engine #0213 at SSC, the G-15 seal was found cracked and buckled. This seal was in place for 34 tests and 15,114 sec of operation. Nozzle 4011 on engine #0213 encompasses the maximum effective protrusion in the fleet, with a nozzle protrusion of 0.100". The maximum acceptable protrusion limit for flight hardware is 0.077". It was removed from flight status because it exceeded the DAR limit. No leaks through the G-15 seal were evident during pre- and post-test leak checks. Bluing and cracking of the G-15 seal were expected prior to nozzle removal based on extended monitoring of deteriorating or missing Flow Recirculation Inhibitor (FRI), but not to the extent witnessed on disassembly. Multiple adjacent leaks caused hydraulic protrusion and core gas ingestion. G-15 seal cracking and erosion are the result of hot-gas ingestion into the G-15 cavity. Diversion of hot-gas flow into the G-15 cavity requires either large, physical tube protrusions or hydraulic protrusion from multiple vented tubes or eroded tube crowns. These protrusions are considered outside of the DAR limit. All STS-39/OV-103 SSMEs were verified not to have coolant tube erosion or vented tubes. There was no degradation or missing FRI on any OV-103 nozzle. Tube protrusions and hot-fire time on STS-39/OV-103 engines were well within DAR limits. The G-15 seals were leak checked since the last engine hot-fire.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
10 (Continued)	Engine #0213 G-15 seal failed during ground testing at SSC.	Rationale for STS-39 flight was:

Rationale for STS-39 flight was:

- Conditions surrounding the G-15 seal on engine #0213 were known and were outside established DAR limits for flight-worthy SSME components.
- STS-39/OV-103 G-15 seals were leak checked, and the FRI was intact on all engines.
- There were no vented tubes in the areas of positive protrusion, and no nozzle tube crown erosion was found.
- Components on STS-39/OV-103 SSMEs were well within DAR limits, and all engines had acceptable protrusion (0.077" maximum acceptable).

Engine	Protrusion
2026	0.066"
2030	0.020"
2029	0.070"

This risk factor was resolved for STS-39.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Test and Evaluation Motor (TEM)-7 fixed-housing ablative liner debond. HR No. BN-08 Rev. C {C} ***No fixed housing ablative liner anomalies were reported on STS-39.***	Post-test examination of the TEM-7 nozzle revealed 100% debond of the fixed-housing ablative liner from the metal housing. Erosion of the 4 ground-test pressure transducer metal fittings was also witnessed. The investigation into this problem revealed that the debond was related to the ground-test configuration (pressure transducer fittings); these fittings are not used on flight nozzles. Oval-shaped sooted flow areas were found around all 4 pressure transducer ports. There was no evidence, however, of hot-gas flow entering the liner/metal housing interface at the ends of the phenolic liner. There was no flight history of fixed-housing ablative liner-to-metal housing debonds. Rationale for STS-39 flight was: • The debonds experienced were unique to the test configuration. • Flight and static test history revealed no evidence of the TEM-7 type debond. ***This risk factor was resolved for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2	Inner Boot Ring (IBR) bond line separation on STS-38 RH Solid Rocket Motor (SRM). HR No. BN-08 Rev. C. {C} ***No SRM IBR anomalies were reported on STS-39. Postflight inspection of STS-39 did, however, identify 11 circumferential areas of erosion/washing and wedgeouts in the nozzle cowl/Outer Boot Ring (OBR) (IFA-39-M-01). A trend assessment performed using Redesigned Solid Rocket Motor (RSRM) historical flight data indicated that there is no general trend of degraded performance or increase in wedgeout or washouts. The trend assessment determined that the STS-39 RH cowl wedgeouts represent the worst case to date for a flight motor. Relative to the OBR, the trend assessment also illustrated a small decrease of the mean margin of safety; however, there was no violation of the safety factor requirements indicated.***	The IBR fixed-housing bond line on the STS-38 RH SRM was found to be approximately 80% separated during a special investigation conducted at postflight disassembly. There were splashdown-related impact marks made by snubber retainer bolts on the bearing end ring in the 270° region. This is indicative of high water-impact loads. The IBR and fixed-housing insulation remained in position; no displacement or edge separations were observed. Removal of the IBR and fixed-housing insulation by machining revealed the IBR bond line separation. The IBR remained bonded to the metal housing in the 90° region. The IBR remained bonded to the fixed-housing insulation, and the fixed-housing insulation was well bonded to the metal housing. No soot or combustion products were found in the separated bond line; it remained intact and/or the ends sealed throughout motor operation. The physical evidence was consistent with splashdown-induced damage; however, there was no previous documentation of this type of separation. Detailed investigation of STS-35 IBRs found bond lines completely intact. Structural analysis predicted the IBR fixed-housing bond line exceeds a 2.0 Factor of Safety (FOS) during motor operation. The analysis indicated post-burn separation was possible due to the splashdown environment; the potential existed for a contaminated and thermally-degraded bond line at the time of splashdown. There were also splashdown-induced tensile loads.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
2 (Continued)	IBR bond line separation on STS-38 RH SRM.	Rationale for STS-39 flight was: • IBR failure was due to post-motor burn events. Snubber damage indicated a high-impact loading event. High water-impact loads, in conjunction with elevated bond line temperatures and contamination, can cause bond line failure. • IBR processing was consistent with previous successfully-flown parts. • The flight FOS is >2.0 at the maximum potential contamination level (60 milligrams/square foot). ***This risk factor was resolved for STS-39.***
3	Room-Temperature Vulcanizing (RTV) was found past the primary O-ring of nozzle joint #5 on disassembly of TEM-7. HR No. BN-03 Rev. C {AR} ***No SRM nozzle joint sealing anomalies were reported on STS-39.***	Upon disassembly of TEM-7, RTV was found past the primary O-ring of nozzle joint #5 (fixed housing-to-aft end ring). The concern was that the RTV could prevent proper sealing of the joint #5 O-ring, allowing a gas path. This was the first time that this condition was seen on joint #5 of a High Performance Motor (HPM); similar conditions were seen at joints #3 and #4 on previous HPM nozzle configurations. There were no similar occurrences on flight motors. While the HPM configuration only has the primary O-ring, flight motors have a secondary O-ring and bolt Stat-O-Seals. Leak tests verify the integrity of the secondary seals after assembly of a flight motor.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
3 (Continued)	RTV was found past the primary O-ring of nozzle joint #5 on disassembly of TEM-7.	RTV typically extends partially down the axial section of the joint, and sometimes reaches the primary O-ring. The O-ring forces uncured RTV away from the seal footprint, maintaining the sealing interface. Tests have demonstrated that RTV and the fluorocarbon O-ring are compatible, with no adverse affects on O-ring sealing properties. Under normal flight operating conditions, nozzle joint #5 primary and secondary seal gaps close or remain stationary. The thermal environment witnessed at joint #5 is reduced by its location behind the flex boot. Thermal analysis predicted that there would be no heat effects to the O-ring if there were blowpaths to the primary O-ring. Rationale for STS-39 flight was: • There was no history of similar conditions on flight motors. • The flight motor configuration provides a secondary O-ring and Stat-O-Seals for further protection. • Leak tests verified the secondary seal integrity. ***This risk factor was resolved for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
4	Thermal Protection System (TPS) damage on STS-39 RH SRB forward assembly. HR No. B-60-24 Rev. C DCN-5 {C} C-60-03 Rev. B DCN-5 {C} ***No SRM anomalies were reported on STS-39 that could be attributed to the preflight contact between the RH SRB forward assembly and the access platform.***	During access platform repositioning in preparation for STS-39 rollout from the Vehicle Assembly Building (VAB), a portion of the platform contacted the RH SRB forward assembly. The Marshall Spray-On Ablative (MSA) TPS was damaged. The MSA in the damaged area was removed to inspect the substrate for damage; no damage was witnessed. K5NA was installed as a substrate for the removed MSA, and MSA was reapplied. Pull tests were performed in the repair area. There was no concern that the damaged would become a debris source because it was on the outboard side of the SRM, away from the Orbiter. Rationale for STS-39 flight was: • No damage to the substrate was witnessed. Appropriate repairs were made. • The damaged area was not in the debris zone. ***This risk factor was resolved for STS-39.***

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
5	STS-39 RH SRB Thrust Vector Control (TVC) hydraulic reservoirs bottomed out in fill direction during Shuttle Interface Test (SIT) hydraulic power-up. HR No. B-20-20 Rev. C {C} B-20-21 Rev. B-DCN6 {C} ***No anomalies in the SRB TVC systems were experienced on STS-39.***	During power-up of SRB hydraulics for the STS-39 SIT, the RH SRB TVC reservoirs were observed to bottom out in the fill direction. Technicians immediately shut down the test and hydraulic system. The overfill condition was caused by a valve not being locked in the operate position. By not being locked, the valve switched to the flush position when hydraulic pressure was applied to the system from the ground support equipment. Repetition of the test verified that the valve could be set in the operate position and not locked in place as required by the SIT procedure. The overfill condition could lead to malfunctioning of the RH rock actuator switching and prefiltration valve, resulting in loss of hydraulic pressure. This action also has the potential for overpressurization of the low-pressure side of the TVC system. Review of the test data confirmed that overpressurization of the RH TVC system did not occur and the low-pressure relief valves did not open. Proper functioning of the low-pressure relief valves was verified during the STS-39 SRB build process. The SIT procedure was modified to verify that the valve was in the proper, operate position and locked in place. Successful completion of the SIT verified proper SRB hydraulic system operation and proper valve position set and lock for flight. There was no additional operation of this valve prior to flight.

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
5 (Continued)	STS-39 RH SRB TVC hydraulic reservoirs bottomed out in fill direction during SIT hydraulic power-up.	Rationale for STS-39 flight was: • The RH TVC hydraulic system was not overpressurized during the SIT. • The anomalous valve was locked in position for flight. ***This risk factor was resolved for STS-39.***
6	STS-37 RH cowl bond line blowpaths. HR No. BN-08 Rev. C {C} ***No cowl bond line blowpaths were reported on STS-39. Postflight inspection of STS-39 did, however, identify 11 circumferential areas of erosion/washing and wedgeouts in the nozzle cowl/OBR (IFA-39-M-01). (See this section, SRM 2, for additional details.)***	The STS-37 RSRM underwent a special detailed postflight disassembly inspection as a result of the TEM-7 IBR and fixed-housing ablative liner debond issue (see Section 4, SRM 1). STS-37 nozzle examination revealed 2 blowpaths on nozzle joint 2 cowl-to-insulation axial bond line, at the 286° and 295° locations. The 2 blowpaths occurred on the RH RSRM nozzle. Examination revealed no heat effects on the phenolic, adhesive, metal, or paint. The inspection also revealed no structural effects, since the cowl assembly is structurally maintained by the cowl bond line and pinned in place with 36 0.375"-diameter steel pins. In the event of a cowl bond line failure, the cowl will remain attached to the cowl housing. The FOS without the bond is 1.62, well above the 1.4 requirement. Cowl bond lines were examined on both RSRM nozzles, and typical soot patterns were noted on the nozzle joint 2 interface. Seventeen of 34 post-fired RSRM inspected nozzles displayed RTV blowpaths in nozzle joint 2. Soot to the primary

RESOLVED STS-39 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
6 (Continued)	STS-37 RH cowl bond line blowpaths.	seal is typical, and seal damage was never observed. Cowl housing thermal damage was also never observed. Rationale for STS-39 flight was: • The observed STS-37 cowl bond line condition resulted in no adverse thermal, structural, or sealing effects. • The FOS without the bond is 1.62, exceeding the 1.4 requirement. ***This risk factor was resolved for STS-39.***

SECTION 5

STS-37 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-37/OV-104 mission, the previous Space Shuttle flight. Each anomaly is briefly described, and risk acceptance information and rationale are provided. Orbital Manuvering System (OMS) pod RP03, last used on STS-38/OV-104, has been installed on STS-39/OV-103. The STS-38 IFA relating to the thruster anomaly on RP03 is addressed in this section as Orbiter 1.

Hazard Report (HR) numbers associated with each risk factor in this section are listed beneath the anomaly title. Where there is no baselined HR associated with the anomaly, or if the associated HR has been eliminated, none is listed. Hazard closure classification, either Accepted Risk {AR} or Controlled {C}, is included for each HR listed.

SECTION 5 INDEX

STS-37 INFLIGHT ANOMALIES

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1	Thruster R1U showed low Chamber Pressure (P_c) on RP03/STS-38. IFA No. STS-38-V-07 HR No. ORBI-056 {C} ***No low thruster Pc anomalies were reported on STS-39.***	Thruster R1U indicated degraded P_c by approximately 20 pounds per square inch absolute (psia) during reentry. Normal operating pressure is 150 psia. Thruster R1U worked properly. R1U was fired normally, but was lowered to last priority; it was not deselected. For previous low P_c failure, the thruster failed "off" when approximately 10-psia degradation was indicated. In addition, 3 other thrusters, R3D, RF3L, and R4U, showed low P_c indications. It was believed that a partially-clogged P_c sensor tube or pressure transducer caused the degradation. Visual inspection of thruster R1U revealed no anomalous conditions. Chamber decay tests performed on November 27, 1990 found 6-8 pounds per square inch (psi) pressure decay in a 2-hour (hr) period. Post-decay mass spectrometer testing identified no leakage. R1U was cleared for flight. Troubleshooting of thrusters R1U, R3D, RF3L, and R4U was completed with no further problems identified. Analysis determined that R1U, R3D, RF3L, and R4U all indicated low P_c during interconnect operations; the right pod thruster manifold was interconnected to the left Orbital Maneuvering System (OMS) propellant tanks. When the RP03 propellant source was switched from the right OMS propellant tanks back to the straight feed configuration, thruster P_c in R1U, R3D, RF3L, and R4U returned to nominal. This finding led to the decision to perform thruster firings on STS-37/OV-104 in the interconnect configuration. When this was performed, OV-104 RCS thrusters L1U and L1L showed degraded P_c; approximately 130 psia instead of 150 psia nominal. P_c in L1U and L1L returned to nominal after reconfiguration to straight feed. Contamination was believed to be in the oxidizer interconnect line on OV-104. (See this section, Orbiter 5, for further details).

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1 (Continued)	Thruster R1U showed low P_c on RP03/STS38.	Rationale for STS-39 flight was: • R1U, R3D, RF3L, and R4U were functional even if this condition occurred on orbit. • There is multiple redundancy: the system can tolerate 2 failures (Crit 1R3) during normal flight and 1 failure during abort modes (Crit 1R2). ***This anomaly was not a safety concern for STS-39.***

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2	Reaction Control System (RCS) primary thruster R1U failed off during External Tank (ET) separation maneuver. IFA No. STS-37-V-01 HR No. ORBI-056 {AR} ***There were no RCS thruster failures on STS-39 attributed to valve contamination. There was, however, a thruster fuel injector temperature anomaly on STS-39 that was caused by a sensor or instrumentation error (STS-39-V-03). There was no indication that this temperature anomaly was related to iron nitrate contamination.***	RCS primary thruster R1U failed off during the STS-37 ET separation maneuver. Recorded pressure data indicated that the thruster P_c was 10 psia; nominal P_c is 150 psia at thruster firing. Thruster injector temperatures indicated some oxidizer and fuel flow in R1U when firing was attempted. R1U was deselected by Redundancy Management (RM) and remained off for the remainder of the mission. There were several similar thruster anomalies of this type on previous missions. The pressure traces from the STS-37 R1U failure were similar to those seen on R3D and R4R on STS-36. Iron nitrate contamination of the oxidizer valve poppet was determined to be the cause of both STS-36 thruster failures. Iron nitrate contamination forms when the oxidizer, Nitrogen Tetroxide (N_2O_4), is allowed to contact moisture in ambient air. This contamination prevented the oxidizer valves from opening in the time allotted to achieve proper P_c. It is believed that iron nitrate contamination may also have caused R1U to fail. Thruster R1U was removed from OV-104 and sent to the White Sands Test Facility (WSTF) for further analysis.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	RCS primary thruster R1U failed off during ET separation maneuver.	Rationale for STS-39 flight was: • STS-39/OV-104 RCS thrusters passed all Operational Maintenance Requirements and Specifications Document (OMRSD) tests. • RCS primary thrusters are Criticality 1R2 during a Return-To-Launch Site (RTLS) abort and 1R3 during normal flight. ***This anomaly/risk factor was acceptable for STS-39.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Water Spray Boiler (WSB) #2 did not cool Auxiliary Power Unit (APU) lube oil while under operation of controller "A". IFA No. STS-37-V-02A HR No. ORBI-036 {AR} ***There were no WSB APU lube cooling anomalies reported on STS-39. However, during STS-39 entry, APU #2, S/N 301, lube oil outlet pressure was low, reading 25 psia; nominal outlet pressure is 40 to 50 psia (IFA No. STS-39-V-11). This was the first APU lube oil pressure-low anomaly in the Program. Previous lube oil anomalies were related to high lube oil outlet pressure.***	WSB #2 failed to cool APU #2 lube oil after the end of the pool boiling period during ascent. WSB #2 was under the operation of controller "A". The crew was directed to switch to controller "B" when lube oil temperature reached 280° Fahrenheit (F); nominal cooling begins at 250°F. Lube oil temperatures began to drop just prior to the crew action. The same anomaly occurred during STS-38, the previous flight of OV-104. Both controller "A" and WSB #2 were removed and replaced after STS-38. The most probable cause of this problem was freezing of the spray bar due to wax buildup in the WSB. Wax, in the form of Pentaerythritol, is formed when hydrazine, the APU fuel, is allowed to mix with lube oil. Pentaerythritol will begin to melt when the lube oil temperature exceeds 200°F. There were no preflight indications of wax buildup in the APU #2 lube oil or in WSB #2. Research of WSB cooling problems indicated an emerging trend when WSBs are paired with APU Serial Number (S/N) 208. Further investigation into other APU/WSB combinations is in work. Not finding high APU lube oil pressures on OV-103 was considered a good indication that wax had not formed in the lube oil. Rationale for STS-39 flight was: • For nominal missions, analysis indicated adequate cooling capability to prevent pre-Main Engine Cutoff (MECO) APU shutdown due to high APU lube oil temperatures. • Previous OV-103 flights showed no signs of WSB cooling problems. ***This anomaly was resolved for STS-39.***

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Power Reactant Supply and Distribution (PRSD) Oxygen (O_2) manifold valve #2 failed to close when commanded. IFA No. STS-37-V-03 HR No. ORBI-094 {AR} ***No PRSD anomalies were reported on STS-39.***	PRSD O_2 manifold valve #2 failed to close when commanded the first 2 times on orbit. The valve, S/N 28, finally closed on the third command and was left closed for the remainder of the STS-37 mission. O_2 manifold valve #2 is 1 of 2 redundant valves used to isolate the manifold or PRSD supply tank from a system leak. Failure of both manifold valves to close would result in depletion of fuel cell O_2 reactants and potential loss of the 3 fuel cell power plants. PRSD O_2 manifold valve #2, S/N 28, previously failed to close when first commanded on OV-104 during STS-34. Postflight troubleshooting found no problems with the valve operation, and the anomaly was closed as unexplained. S/N 28 worked properly on the 2 OV-104 flights between STS-34 and STS-37: STS-36 and STS-38. S/N 28 was removed and replaced when OV-104 returned to KSC. There were no similar PRSD manifold valve problems on OV-103. Rationale for STS-39 flight was: • OV-103 PRSD manifold valves checked out satisfactorily during the STS-39 flow processing; there was no history of previous failures. • Loss of the vehicle and crew would require failure of redundant valves and loss of 3 fuel cells. ***This anomaly was resolved for STS-39.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Indications of low P_c on primary thrusters L1U and L1L during interconnect operations. IFA No. STS-37-V-08 HR No. ORBI-056 {C} ***No low thruster P_c anomalies were reported on STS-39.***	During STS-38/OV-104, low P_c was experienced on 4 primary thrusters: R1U, R3D, RF3L, and R4U. The related IFA is found in this section, Orbiter 1, because the right pod on STS-38/OV-104, RP03, was installed on STS-39/OV-103. Postflight troubleshooting did not reveal any thruster leaks or other anomalies which might lead to low thruster P_c. Recent analysis determined that STS-38 thrusters R1U, R3D, RF3L, and R4U all indicated low P_c during interconnect operations; the right pod thruster manifold was interconnected to the left Orbital Maneuvering System (OMS) propellant tanks. When the RP03 propellant source was switched from the right OMS propellant tanks back to the straight feed configuration, thruster P_c in R1U, R3D, RF3L, and R4U returned to nominal. This finding led to the decision to perform thruster firings on STS-37/OV-104 in the interconnect configuration. When this was performed, thrusters L1U and L1L showed degraded P_c, approximately 130 psia instead of 150 psia nominal. P_c in L1U and L1L returned to nominal after reconfiguration to straight feed. It is believed that there was contamination in the oxidizer interconnect line. No further action was taken at KSC. All thrusters on STS-39/OV-103, including those that showed low P_c in RP03, passed all OMRSD checkouts. The STS-39 mission profile required multiple thruster firings in the interconnect configuration. This was a good test to determine if: (1) there is a problem with the interconnect configuration on OV-104, and (2) if there is a problem with low thruster P_c associated with OMS pod RP03. If the anomaly had repeated on STS-39, it would have reduced mission capability and would be a safety-of-flight issue.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	Indications of low P_c on primary thrusters L1U and L1L during interconnect operations.	Rationale for STS-39 flight was: • The low thruster P_c problem, if related to the interconnect issue and experienced on STS-39/OV-103, would be a mission impact issue and not a safety concern. • RCS thrusters are Criticality 1R2 for RTLS abort and 1R3 for other mission phases. ***This anomaly was not a safety concern for STS-39.***
6	Backup Flight Software (BFS) navigation initialization anomaly. IFA No. STS-37-V-09 HR No. ORBI-066 {AR} ***No BFS anomalies were reported on STS-39.***	Postlaunch data analysis of the BFS telemetry indicated that, from prelaunch BFS Operational Sequence (OPS)-1 transition until the T–8 second (sec) BFS reinitialization, the Z-component (altitude) of the BFS state vector increased at a rate of approximately 1 foot per second (ft/sec) to approximately 7700 feet (ft). The BFS navigation errors were cleared at the T–8 sec point in the launch countdown when the BFS was reinitialized to the pad B position. Ascent telemetry review indicated that both the BFS and the Primary Avionics Software System (PASS) performed nominally. Data review found that the Z-component was approximately 250 ft at the OPS-1 transition plus 3-minute (min) point; previous flight data indicated the Z-component to be 6-30 ft during the same time in the launch countdown. During previous launch countdowns, Z-component errors of 1500 to 3000 ft were observed up to the T–8 sec period. Recent pre-STS-37 tests conducted in the Shuttle Avionics

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)	BFS navigation initialization anomaly.	Integration Laboratory (SAIL) with the new General Purpose Computers (GPCs) demonstrated errors of 4000 to 4500 ft. Errors are believed to be caused by gravity feedback and eventually lead to error growth in the Z-component. The investigation included a code audit of the BFS and PASS. This audit determined that the problem was only in the first initialization of the navigation function in the BFS. The PASS navigation function did not have the same problems. Testing of the BFS at the SAIL repeated the STS-37 prelaunch anomaly on each attempt and demonstrated that the problem was not from the Inertial Measurement Unit (IMU) input. It is believed that this anomaly was caused by the increased processing time of the new GPCs combined with the way BFS sequences the initialization of the navigation function. This software problem was determined to be limited to the prelaunch initialization of the BFS. Reinitialization of the BFS at T−8 sec clears any problems created during this first BFS initialization. Discrepancy Report (DR) 106197 was generated to identify this BFS navigation initialization problem for resolution. Rationale for STS-39 flight was: • The anomaly was isolated to prelaunch initialization of the BFS navigation function only and is cleared at the T−8 sec BFS reinitialization. • Analysis and SAIL testing has demonstrated no similar anomalies with the PASS. ***The potential for repetition of this anomaly was recognized. If repeated, this anomaly was acceptable for STS-39.***

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Left-Hand (LH) forward skirt skin panel buckling. IFA No. STS-37-B-01 ***No Solid Rocket Booster (SRB) anomalies were reported on STS-39.***	During postflight inspection, the aft end of the LH forward skirt skin panel was found to have buckled on either side of the system tunnel. This forward skirt, S/N 22, had flown on STS-27R (BIO-030) and STS-33R (BIO-034). The STS-27R and STS-33 postflight inspection reports indicated no abnormal conditions of the LH forward skirt. STS-37 postflight Orbiter data, reviewed through prelaunch, launch and separation, showed nominal flight conditions. Marshall Space Flight Center (MSFC) concluded that buckling was most likely caused by the slapdown load during water impact. The slapdown load on this booster was reported to be the highest ever recorded (92 g versus the 12 to 40 g history). The dynamics of the SRB were likely affected by wave height, period, wave length, etc., associated with a sea state 5 condition. Rationale for STS-39 flight was: • This was a descent-related occurrence which impacts attrition only; not a safety-of-flight issue. • Forward skirt refurbishment inspection occurs between flights and includes visual inspection, critical dimensions checks, corrosion inspection, and nondestructive testing. • No STS-39/BI-043 forward skirt anomalies were reported prior to launch. ***This anomaly was resolved for STS-39.***

SECTION 6

STS-41 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-41/OV-103 mission, the previous flight of the Orbiter vehicle. Each anomaly is briefly described, and risk acceptance information and rationale are provided.

Hazard Report (HR) numbers associated with each anomaly in this section are listed beneath the anomaly title. Where there is no baselined HR associated with the anomaly, or if the associated HR has been eliminated, none is listed. Hazard closure classification, either Accepted Risk {AR} or Controlled {C}, is included for each HR listed.

SECTION 6 INDEX

STS-41 INFLIGHT ANOMALIES

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1	System Management (SM) Nominal Bus Assignment Table (NBAT) General Purpose Computer (GPC) #2 assignment anomaly on STS-41. IFA No. STS-41-I-01 HR No. ORBI-066 {AR} ORBI-194 {AR} ***No similar GPC assignment anomalies were reported on STS-39.***	During performance of STS-41 post-insertion procedures, the crew discovered that GPC #2 was assigned to string 3 according to SM2 NBAT; GPC #2 should have been unassigned. Investigation determined that this condition existed prior to launch. Further investigation found that an error was made during S0007 troubleshooting of an Inertial Upper Stage (IUS) valve configuration problem. Review of Launch Processing Set (LPS) retrievals found that Data Entry Unit (DEU) equivalent commands were issued on October 5, 1990, to Specification (Spec) 0, GPC Memory. DEU equivalent commands should have been sent to Spec 62, Payload Communications Display, per the IUS telemetry configuration. The resulting NBAT anomaly would not have affected the actual bus assignments because flight software would not have accepted this configuration. The crew reworked the NBAT to the proper configuration. A review of software change procedures at Kennedy Space Center (KSC) was undertaken to determine if acceptable validation procedures existed. In addition, a review of all potential issuances of DEU equivalent commands by the LPS was completed. It was determined that, in all cases except IUS Telemetry Format Load (TFL) lockup, all LPS Ground Operations Aerospace Language (GOAL) programs that issue DEU equivalent commands verify proper Specs prior to releasing the command. Efforts are underway to include the IUS TFL lockup in the GOAL Spec verification process prior to DEU equivalent command issuance. Evaluation of software-driven operations at KSC determined that the potential exposure to similar problems was limited to S0007 operations.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1 (Continued)	SM NBAT GPC #2 assignment anomaly on STS-41.	Rationale for STS-39 flight was: • The IUS TFL lockup procedure is only used for nonclassified IUS flights; corrective actions are in work to eliminate this exposure for future IUS missions. • Other software change procedures already incorporated verification techniques to protect against this type of error. • Nominal crew procedures verified the correct NBAT configuration. ***This anomaly was resolved for STS-39.***
2	Aft compartment Hydrogen (H_2) concentration high during ascent. IFA No. STS-41-I-03 HR No. ORBI-306 {AR} ***No similar aft compartment H_2 concentration anomalies were reported on STS-39.***	Postflight analysis of STS-41/OV-103 aft compartment catch bottle contents indicated the highest ascent H_2 concentrations of any Shuttle mission. Leak rate calculations based on H_2 concentrations in 3 of 6 STS-41 catch bottles ranged from 16,500 standard cubic inches per minute (scim) to 37,000 scim. Average H_2 leakage during ascent for the fleet is less than 10,000 scim. Prior to STS-41, the maximum catch bottle H_2 concentration was on STS-31, the last OV-103 mission. Leak calculations based on the STS-31 sample resulted in an estimated leak rate of 30,000 scim. A leak greater than 59,000 scim, coupled with a sufficient amount of Oxygen (O_2), is considered to be the minimum flammability limit. Through the 11 OV-103 flights, there is a trend of increasing H_2 concentrations in the catch bottles.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)	Aft compartment H_2 concentration high during ascent.	A possible leak source is the Space Shuttle Main Engines (SSMEs). Prior to engine start, only approximately 5% of the SSME high-pressure H_2 joints are wetted. The Launch Commit Criteria (LCC) was changed for STS-39 and subsequent missions to only allow an aft compartment H_2 concentration of 150 parts per million (ppm) during stable replenish (i.e., when the high-pressure joints are wetted prior to launch). The 3 STS-41 SSMEs were on OV-103 for 3 flights; all were removed. Catch bottles are used to periodically sample aft compartment atmosphere for H_2, O_2, and other potentially hazardous elements. Samples are analyzed at KSC using a gas chromatograph. There are 6 catch bottles in each aft compartment, and it is not unusual to have only 1-3 good samples. Catch bottle samples with argon present, or catch bottles with higher than expected pressure, are discarded because they indicate atmospheric leakage into the bottles after landing. There is nominally a leak rate variation between the left and right catch bottles. Variability has ranged from 2,500 scim on STS-29 to 19,500 scim on STS-41. This variability is common to all vehicles and is attributed to the Liquid Hydrogen (LH_2) 17-inch disconnect being located on the left side of the aft compartment. Post-STS-41, SSMEs and GH_2-side Main Propulsion Systems (MPSs) were extensively leak checked with helium at 900 pounds per square inch gage (psig). The engine #1 MPS GH_2 Flow Control Valve (FCV) outlet tube weld was found to

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)	Aft compartment H_2 concentration high during ascent.	be leaking during ambient helium mass spectrometer leak checks. This leak was small: 2.38 x 10^{-6} standard cubic centimeters per second (sccs) versus the specification limit of 1 x 10^{-6} sccs. This FCV was replaced. All 3 SSMEs were encapsulated, and leak checked upon removal, with no leaks found. The encapsulated leak test was also performed on STS-39 SSMEs prior to installation, with no leaks identified. Post SSME removal inspection of Orbiter interface flanges found warped GH_2 engine isolation check valve flanges on engine #2 and #3 interfaces. Vendor inspection indicated that one of these check valves had a 0.0038" out-of-flatness condition; the other was within drawing flatness tolerance of 0.001". Both isolation check valves were replaced. Rationale for STS-39 flight was: • All SSMEs flown on STS-41 were removed. The STS-39 SSMEs passed the encapsulated leak checks. • All MPS decay and helium signature tests were performed on STS-39/OV-103 with only minor anomalies identified. These tests were more stringent than those performed on STS-41.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)	Aft compartment H_2 concentration high during ascent.	• Ascent aft compartment H_2 concentrations measured to date were within flammability limits. • The LCC was revised to allow stable replenish aft compartment H_2 concentrations up to 150 ppm. ***This anomaly/risk factor was acceptable for STS-39.***
3	Left-Hand (LH) Solid Rocket Booster (SRB) aft strut separation device NASA Standard Initiator (NSI) detonator separated from the pressure cartridge. IFA No. STS-41-I-03 HR No. INTG-135 {C} ***No NSI anomalies were reported on STS-39.***	Postflight inspection of STS-41 Solid Rocket Motors (SRMs) found that NSI, lot number MPX, Serial Number (S/N) 1193, was ejected from the pressure cartridge, Part Number (P/N) 10303-0001-801, lot number AAP, S/N 2,003,371. This ejection occurred after the proper functioning of the NSI during SRB/External Tank (ET) aft strut separation. The resulting debris was contained by the surrounding foam insulation, and there was no debris concern raised by this anomaly. This was the first occurrence of this failure mode in flight; several failures of this type were experienced during SRB/ET aft strut separation device qualification tests (in excess of 74% of the 35 devices tested). It is, therefore, believed that this type of failure was expected to occur during flight during the life of the Program. In all cases, this was a post-function failure of the NSI in the aft strut separation device; in no case did the separation device fail to separate. Orbiter separation hardware and other NSI applications have had no history of NSI ejections, either during qualification

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
3 (Continued)	LH SRB aft strut separation device NSI detonator separated from the pressure cartridge.	testing or flight. NSI applications in SRM Igniter Initiators (SIIs) have had no similar ejection problems, either in qualification tests or in flight. Initial hardness measurements taken on NSI S/N 1193 at United Space Boosters, Inc. (USBI) indicated that it was below the minimum Rockwell hardness of 36. Subsequent hardness measurements of S/N 1193 at Rockwell International (RI) determined that the actual hardness was in excess of the requirement. Original coupon hardness data for lot MPX also indicated hardness measurements in excess of the requirement. The USBI measurements, therefore, were determined to be in error. Each NSI is subjected to proof-pressure testing at 15,000 psig prior to propellant loading during fabrication. Proof-pressure tests of 2% of each lot subject NSIs to 40,000 pounds per square inch (psi). Independent testing at Marshall Space Flight Center (MSFC) demonstrated that the NSI will rupture at the thread relief when subjected to hydrostatic pressure of 66,000 psi. The resulting safety margin, based on NSI static test pressures, is greater than 4.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
3 (Continued)	LH SRB aft strut separation device NSI detonator separated from the pressure cartridge.	Rationale for STS-39 flight was: • NSI blowout can be expected to occur during the life of the Program based on the aft strut separation device qualification test experience. • NSI blowout is a post-functional aft strut separation device failure. • There is no history of similar NSI ejections in other applications on the Orbiter. ***This anomaly was resolved for STS-39.***

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Auxiliary Power Unit (APU) #1 Gas Generator (GG)/Fuel Pump (FP) heater system "B" failed "on" during STS-41. IFA No. STS-41-V-03 HR No. ORBI-104A {C} ***There were no STS-39 APU heater anomalies attributed to improper heater wiring.***	During Flight Day (FD) 4 checkout of the Flight Control System (FCS), normal switchover of the APU #1 heaters from system "A" to "B" was performed. Upon switchover, the "B" heater failed to cycle off. APU heater cycling is thermostatically controlled, cycling on at 73°F and off at 100°F. A Fault Detection and Annunciator (FDA) alert sounded when the temperature in the APU fuel bypass line reached 180°F. This occurred 2 minutes (min) after switchover from system "A" to system "B" heaters. The bypass line temperature rose at a rate of 40°F/min versus the 6°F/min nominal rate. APU fuel bypass line temperature peaked at 258°F 3 min following switchover. The crew immediately switched back to the system "A" heaters, and normal GG/FP heater cycling resumed. This failure mode is indicative of a short in the heater string with possible thermostat failure. The worst-case effect would be a failed "on" heater. If the failed "on" heater is not detected, the fuel lines would overheat (Crit 1R2). Hydrazine, with a detonation temperature of 350°F, would detonate and result in APU fuel line rupture, hydrazine leakage, fire, and potential loss of crew and vehicle. Cycling APU GG/FP heaters off at 100°F is designed to protect against fuel line overtemperature. Troubleshooting at KSC found a short circuit to ground between the fuel line heater and the water valve heater wires. This short was believed to be at a location where a clamp is used to secure the wiring to the fuel line. Further troubleshooting is in work to isolate the short. It is believed that activity associated with the changeout of the system "A" thermostat during STS-41 flow processing resulted in damage to the system "B" wiring. Both system "A" and "B" wiring run through the same cable.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	APU #1 GG/FP heater system "B" failed "on" during STS-41.	Retest of system "A" was performed after the thermostat changeout with no anomalies noted; however, no tests were performed on system "B". Heater system "B" was replaced. Both "A" and "B" heater systems were checked out in the Orbiter Processing Facility (OPF) and were cycled during tanking. Action was assigned to the Orbiter Project at the STS-38 Flight Readiness Review to determine the acceptability for future flights with a potential "smart" APU heater circuit failure on orbit, as experienced on STS-41. The response to this action is summarized as follows: • Prelaunch tests will be conducted during tanking for flight on all APU heaters. This test will include switching between "A" and "B" heaters. • There will be a new APU high-temperature FDA limit set at 150°F. The present limit is 180°F. This will provide an additional minute of response time to the crew. The ground monitoring system was changed to alert the APU console operator when temperatures reach 130°F to enhance response awareness. • All APU reconfigurations will be performed in Acquisition of Signal (AOS) conditions only, to allow ground monitoring of any failure. • On FD 1, heater reconfiguration will be performed early, at 6-hr Mission Elapsed Time (MET), to allow verification of system "B" heaters.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	APU #1 GG/FP heater system "B" failed "on" during STS-41.	• Any APU heater anomalies detected during Loss of Signal (LOS) conditions will result in the crew powering down all APU GG/FP heaters. Heater reconfiguration will follow for the failed heater, and the remaining heater strings will be reactivated. • To enhance response time, "booties" will be installed on APU heater switches for quick recognition. Additionally, the crew's orbit pocket checklist was updated to reflect crew response procedures in the event of an APU heater FDA. Rationale for STS-39 flight was: • The procedural changes described above were made in the event of an APU heater/thermostat failure. • All STS-39/OV-103 APU heater circuits were successfully checked out in the OPF with no anomalies.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	APU #1 GG/FP heater system "B" failed "on" during STS-41.	• OV-103 troubleshooting and flight history indicated a unique failure on STS-41. • APU heaters are redundant and are monitored by the onboard FDA and the Mission Control Center (MCC) during operation. High-temperature limits were lowered to enhance response time. ***This anomaly was resolved for STS-39.***
2	Inertial Measurement Unit (IMU) #1 experienced Z-axis accelerometer transients. IFA No. STS-41-V-04 HR No. ORBI-051 {C} ***No IMU accelerometer anomalies were reported on STS-39.***	STS-41/OV-103 IMU #1, S/N 007, was deselected by Redundancy Management (RM) because it was experiencing Z-axis accelerometer transients. Problems occurred several times with the transient lasting from 5-15 min at a time. IMU #1 did, however, track the redundant IMU set following deselection. A similar problem was experienced on STS-32, when IMU #1, S/N 024, was deselected for Y-axis transients. There was no indication that this was a generic IMU problem; however, there have now been 2 flight and ground-based test transient accelerometer failures. IMU #1, S/N 007, was removed and replaced at KSC. The concern with an IMU failure is: first or second failures may require crew action to downmode IMUs to standby if in OPS-2 or OPS-3, or to downmode to off in all OPS modes if IMUs are clearly degraded beyond use. This could result in

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	IMU #1 experienced Z-axis accelerometer transients.	possible crew/vehicle loss due to multiple-axis second failure. If the second failure is multiple axis, it can be detected but not isolated. There is a possibility of RM prime selecting the failed IMU; the crew would be required to select the good IMU if it can determine from ground track information that the failed IMU had been selected by RM prime. Failure of the navigation base could result in loss of all 3 IMUs, and subsequent loss of the crew/vehicle. Rationale for the STS-39 flight was: • The IMU system has triple redundancy. Flight rules are in place to deal with single or double loss of redundancy. • This was not considered a generic IMU problem. ***This anomaly was resolved for STS-39.***

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Backup Flight Software (BFS) backup cabin delta pressure/delta temperature alarm was triggered at Main Engine Cutoff (MECO). IFA No. STS-41-V-05 ***There were no cabin pressure anomalies reported on STS-39.***	The BFS backup delta pressure/delta temperature calculation at MECO indicated a cabin pressure leak rate in excess of 0.14 psi/min. This calculation generated an alarm to the crew to identify the apparent condition. After silencing the alarm and checking alternate readouts, it was determined that there was no cabin leak. Failure analysis and data review found no problem with either the cabin pressure sensor or the BFS. An actual hardware or software failure would have been detected by ground monitoring. This was the first failure occurrence of this type, and there were no other delta pressure alarms generated for the remainder of the flight. Continued evaluation determined that a 2 data bit step response by the new transducer in the cabin pressure sensor caused the delta pressure calculation to trigger the alarm. Transducers previously used in the cabin pressure sensor only generated a 1 data bit response. The BFS design group is currently reviewing this anomaly for a potential software change. The potential existed for a repeat of this anomaly on STS-39 because similar cabin pressure transducers were installed. The first experience on STS-41 resulted in a one-time alarm that was easily silenced. The STS-39 crew was briefed on the potential for this alarm prior to launch.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)	BFS backup cabin delta pressure/delta temperature alarm was triggered at MECO.	Rationale for STS-39 flight was: • This anomaly between the BFS and delta pressure transducer results in a non-safety-critical, false delta cabin pressure alarm. There are alternate ways to determine the validity of a similar false alarm. • The STS-39 crew was briefed on the potential for a repeat of this anomaly. ***This anomaly was not a safety concern for STS-39.***
4	STS-41 Commander's Left-Hand (LH) Attitude Direction Indicator (ADI) rate scale switch failure. IFA No. STS-41-V-06 ***There were no similar anomalies reported on STS-39.***	During FD 4 deorbit preparations, a failure message was presented that identified that the Commander's LH ADI rate scale switch showed both "HI" and "MED" simultaneously. Postflight data analysis confirmed that both signals were active simultaneously for 26 seconds (sec). There had been no prior failures of this switch type without switch operation. This failure mode could not be reproduced during troubleshooting and examination at KSC. The switch was removed and sent to the vendor for x-ray analysis. Some contamination was found; however, it was not of sufficient size to cause a contact-to-contact short. This problem was closed as an unexplained anomaly. The switch, P/N ME452-0102-7101, is a single-contact, triple-pole, hermetically-sealed toggle switch. This switch has a Crit 3/3 assessment in the ADI circuitry. RI identified 196 similar type switches (7106) at other locations in the Orbiter.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4 (Continued)	STS-41 Commander's LH ADI rate scale switch failure.	Two applications are Crit 1R2, and 2 applications are Crit 1R3. There are no Crit 1/1 applications in the Orbiter. There had been a total of 11,400,000 actuations of this switch (pre- and post-51-L), and 1 failed closed. The probability of this failure is very low (probability = 8.77×10^{-8}). Rationale for STS-39 flight was: • The LH ADI rate scale switch was replaced. The failure mode could not be reproduced, and this problem was closed as an unexplained anomaly. • This failure was not considered to be generic and was a first-time failure occurrence with a very low failure rate (probability = 8.77×10^{-8}). • Similar switches with Crit 1R applications are tested before each flight per the Operational Maintenance Requirements and Specifications Document (OMRSD). ***This anomaly was resolved for STS-39.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Orbiter/ET LH_2 aft attach/separation hole plugger failed. IFA No. STS-41-V-07 HR No. ORBI-302A {AR} ***No similar Orbiter/ET hole plugger anomalies were reported on STS-39.***	The Orbiter/ET LH_2 aft attach/separation hole plugger failed to close fully. Postlanding inspection found that the plugger failed to seat properly due to debris obstruction. Debris also fell to the runway when the ET umbilical door was opened. This debris apparently escaped from the debris container after the ET umbilical door closed on orbit. Similar hole plugger failures occurred on STS-29, STS-34, and STS-35. There was concern that loose debris could block the ET umbilical door from closing fully, resulting in potential loss of the crew and vehicle during reentry. The likelihood of escaping fragments preventing the ET umbilical door from closing was determined to be remote. The ET doors may be recycled in flight if closing or latching is impeded. The Orbiter performs a maneuver at ET separation, moving away from the ET and escaping potential debris prior to ET umbilical door closure. Rationale for STS-39 flight was: • The likelihood of debris jamming the ET umbilical door was remote. • Doors may be recycled in flight if closing or latching is impeded. • The ET separation burn moves the Orbiter away from any potentially escaping debris. ***This anomaly/risk factor was acceptable for STS-39.***

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	STS-41 LH Rotation Hand Controller (RHC) trim inhibit switch indicated a contact miscompare. IFA No. STS-41-V-08 ***There were no similar anomalies reported on STS-39.***	During FD 4 operations, a failure flag indicated that the LH RHC trim inhibit switch "enable" and "inhibit" contacts were made simultaneously. Data review confirmed that both signals were present for a 15-sec period. The problem disappeared after this 15-sec period and did not repeat. The RHC trim inhibit switch, P/N ME452-0102-7201, is a Crit 3/3 application. This anomaly could not be repeated during troubleshooting at KSC (unexplained anomaly). There are a total of 274 P/N ME452-0102-7201 switches per Orbiter. Two applications, Cathode Ray Tube (CRT) Select (SEL) switches #7 and #8, are located in the Data Processing Software System (DPS) and have been identified as Crit 1/1 failure modes. Switch #7, the left-side CRT SEL switch, provides the means for switching the left keyboard from the left CRT to the center CRT, or vice versa; switch #8, the right-side CRT SEL switch, provides the means for switching the right keyboard from the right CRT to the center CRT, or vice versa. During a Crit 1/1 failure mode (fails closed, premature closed, or contact-to-contact short), both the center and either the left or right Display Electronic Units (DEUs) will respond to the same keyboard entry due to switch failure. If this occurs in a critical situation, the results could be catastrophic. However, the Redundancy Management System (RMS) would recognize the opposing commands, vote out the input, and disregard the entry. For this application, actions taken by the RMS are not considered as a backup to switch failure.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	STS-41 LH RHC trim inhibit switch indicated a contact miscompare.	There were 3 previous failures of this switch type in different applications: 1 in flight on a flight-deck speaker microphone unit, and 2 during testing at the Shuttle Avionics Integration Laboratory (SAIL). Of the 3 previously-recorded failures, only 1 was considered to be a hardware failure and was attributed to wear. The switch was operated in excess of its certified life. Flight and test history data indicated no failures related to age or number of cycles within the qualified lifetime. Ground turnaround tests, including keyboard testing, verify that all Crit 1/1 and 1R2 switches are functional. Rationale for STS-39 flight was: • Ground turnaround test verified that all Crit 1/1 and 1R2 switches were functional. • Flight and test history indicated that there were no failures related to age or number of cycles within the qualified lifetime. • The RMS would recognize opposing commands and vote out an erroneous input caused by switch failure. ***This anomaly was resolved for STS-39.***

SECTION 7

STS-39 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-39/OV-103 mission. Each anomaly is briefly described, and risk acceptance information and rationale are provided.

Hazard Report (HR) numbers associated with each risk factor in this section are listed beneath the anomaly title. Where there is no baselined HR associated with the anomaly, or if the associated HR has been eliminated, none is listed. Hazard closure classification, either Accepted Risk {AR} or Controlled {C}, is included for each HR listed.

SECTION 7 INDEX

STS-39 INFLIGHT ANOMALIES

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Abort Region Determinator (ARD) indicated different abort boundary times than expected. IFA No. STS-39-I-01	Postflight data evaluation determined that STS-39 achieved Main Engine Cutoff (MECO) 1.1 seconds (sec) later than predicted. ARD calls for uphill aborts were also later than predicted, and the 3-g throttle started 5 sec later than predicted. The real-time estimate was that the Space Shuttle Vehicle (SSV) was approximately 4,000 pounds (lb) heavier than calculated prior to flight. This early estimate was based on typical propellant flow (approximately 3,600 lb/sec x 1.1 sec = 3,960 lb) and a 1-to-1 relationship between excess propellant and inert weight. Reconstruction of the trajectory determined that there was a delta Space Shuttle Main Engine (SSME) specific impulse (isp) of +0.86 sec, a delta SSME thrust of -2,050 lb, and a delta inert weight of -500 lb. Analysis results demonstrated that the combination of higher-than-predicted isp and lower-than-predicted thrust will force the ARD to be overly conservative. This was considered preferable to having the ARD be non-conservative. Investigation into this anomaly found that the reconstructed trajectory matched the ARD predictions for STS-39. The higher-than-predicted isp and lower-than-predicted thrust were the result of conservative ARD predictions based on STS-39 engine test performance data. The low isp experienced on STS-39 was within the program experience base; since STS-2, the isp range has been ±0.9 sec.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Flash Evaporator System (FES) feedline "A", system #2, heater failure. IFA No. STS-39-V-01 HR No. INTG-164 {C} ORBI-276B {C}	Prior to entering the tanking phase of the second launch countdown, FES feedline "A", system #2, heater failed. Review of data traces indicated that the cause of the heater failure was a short-to-ground, evidenced by a 20-24 ampere (amp) current spike for 5-10 milliseconds on main bus "B" prior to removal of heater power by a 10-amp line fuse. Although the FES feedline heaters are Criticality (Crit) 2R3, there was concern that the short could be in a cable bundle that also included Crit 1/1 functions. There was also concern for the potential of arc-tracking of the Kapton insulation, leading to a potential fire. It was believed that a technician inadvertently stepped on the feedline heater wire harness during repair of the secondary seal cavity pressure sensor that caused the first launch attempt to be scrubbed. The prelaunch investigation into this problem determined that the signature of the current spike, being short in duration, did not show signs of arc-tracking. This investigation also included identifying all functions routed through the cable bundle in question and testing the functionality of all critical command paths. Nearly all critical command paths were verified; however, the commands for Solid Rocket Booster (SRB) holddown post release systems, that were routed through the suspect cable bundle, could not be verified until the actual command was generated at T–0. Redundant command paths for the SRB holddown post release were available. The decision was made to accept the risk of this condition and proceed with the launch of STS-39. Further investigation into this anomaly will continue as access to the cable bundle and FES feedline "A", system #2, becomes available.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	FES feedline "A", system #2, heater failure.	Initial troubleshooting identified the potential for this anomaly to have originated in the Aft Load Controller Assembly (ALCA) #2. Removal and replacement of ALCA #2 did not totally alleviate the problem. Megger checks isolated a short circuit in the wire harness between ALCA #2 and FES #2. This wire harness was removed and replaced, and retest of FES #2 heaters was successfully completed.
2	Auxiliary Power Unit (APU) #2 Fuel Pump (FP)/Gas Generator Valve Module (GGVM) coolant system "A" valve did not open. IFA No. STS-39-V-02 HR No. ORBI-265A {AR}	After on-orbit APU shutdown, FP/GGVM coolant systems are automatically activated to cool the FP/GGVM. On STS-39/OV-103, APU #2 FP/GGVM cooling system "A" failed to initiate cooling. Cooling system "B" was successfully activated to perform this cooling function. It was determined that the cooling system "A" spray valve LV25 had failed closed. FP/GGVM cooling is needed in the contingency that an abort from orbit is required soon after APU shutdown. Without additional cooling, the FP/GGVM takes approximately 6 hours to cool beyond the point of potential hydrazine detonation and subsequent fire or explosion in the APU, possibly causing loss of the vehicle and crew. APU FP/GGVM cooling spray valves are life-limited to 9 months due to susceptibility to nickel-hydroxide contamination. Valve LV25 on cooling system "A" had exceeded this life-limit by 30 days at the time of the STS-39 launch. A time/life exception (EV 2123R1) was approved prior to launch to allow LV25 to fly on STS-39. This valve was removed, and a replacement valve will be installed prior to the next OV-103 mission.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Reaction Control System (RCS) vernier thruster F5R fuel injector temperature biased low. IFA No. STS-39-V-03 HR No. ORBI-056 {C}	During firing of RCS vernier thruster F5R, the fuel injector temperature read 30-40°F lower than the oxidizer injector temperature. Both the oxidizer and fuel injector temperatures should not vary more than 10°F. Because there were no apparent thruster heater failures or leaks detected, it is believed that this anomaly was caused by a sensor or instrumentation error. Loss of a vernier thruster is a Crit 2/2 failure mode, loss of mission capability. There have been 2 previous vernier thruster oxidizer/fuel injector temperature anomalies similar to this occurrence; 1 on STS-3 and 1 on STS-4. Both anomalies were determined to be the result of poor sensor thermal conductivity in the vacuum of space. The corrective action taken to overcome thermal conductivity problems was to add thermal grease to the exterior of the sensor probe and the sensor injector well. It was subsequently determined that the sensor in F5R was of the old configuration and did not have the added thermal grease. Postflight troubleshooting determined that this anomaly, as in the 2 previous cases, was caused by poor contact between the sensor and the thruster. There was not a true temperature degradation in thruster F5R on STS-39.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Operations (OPS) recorder #2 uncommanded configuration before launch [potential Multiplexer-Demultiplexer (MDM) hybrid circuit failure]. IFA No. STS-39-V-04 HR No. ORBI-038A {AR}	During the second launch attempt, OPS recorder #2 experienced uncommanded operation; the recorder was discovered "on" and had changed tracks. Data review indicated that OPS recorder #2 operated in the following sequence prior to being found "on" and commanded "off": approximately 4 seconds (sec) "on", 1 sec "off", and 3 minutes (min), 20 sec "on". Preliminary engineering analysis prior to launch indicated that the anomaly was in the recorder. Testing of OPS recorder #2 was prescribed and performed prior to launch, with no further anomalies identified. OPS recorder #2 was cleared for launch and operated nominally through most of the mission. On Flight Day (FD) 7, OPS recorder #2 was witnessed to repeat the prelaunch anomaly. While "on", OPS recorder #2 changed tracks, speed, and mode without the required commands. OPS recorder #2 was subsequently reconfigured from the ground, and it operated nominally for the remainder of the mission. Investigation into the prelaunch OPS recorder #2 anomaly continued throughout the mission. Several different scenarios were identified that could recreate the prelaunch anomaly. Consideration of these scenarios led to the preliminary determination that MDM Payload Forward 2 (PF2), Serial Number (S/N) 72, the MDM between the General Purpose Computers (GPCs) and OPS recorder #2, could be the cause of the anomaly. It was believed that PF2 generated erroneous output to OPS recorder #2, causing the track, speed, and mode change experienced prior to and during the STS-39 mission. The potential for erroneous MDM output, if generic, was a concern in the cases where MDMs are used in Crit 1 proximity/rendezvous operations. MDM PF2 performs only Crit 3/3 functions.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4 (Continued)	OPS recorder #2 uncommanded configuration before launch [potential MDM hybrid circuit failure].	Review of MDM failure history indicated 2 failures in MDM hybrid circuits. One of these 2 failures was in 1985 and was with MDM PF2, S/N 72, the unit on STS-39/OV-103. The 1985 anomaly had characteristics similar to the STS-39 OPS recorder #2 incident. At that time, the anomaly was attributed to the MDM control hybrid circuit on card 10. The control hybrid circuit was replaced, and MDM S/N 72 was installed on OV-103 in the PF2 position. There were no further anomalies with this MDM until STS-39. Analysis indicated that the shift hybrid circuit may have caused both the 1985 anomaly and the STS-39 anomaly. MDM PF2, S/N 72, was removed from OV-103 at Kennedy Space Center (KSC) and sent to the vendor for further failure analysis. There was no indication of a generic MDM problem.
5	Supply water dump nozzle temperature drop. IFA No. STS-39-V-08	Approximately 20 min into supply water dump #5, the water dump nozzle temperature rapidly decreased 30°F, from 163°F to 133°F, over a 14-min period. Nozzle temperatures normally remain around 170°F. After this period, the nozzle temperature recovered to normal. With the supply water dump valve closed prior to a subsequent dump, a rapid 5°F drop in nozzle temperature was observed. Nozzle heaters were "on" when this event occurred. Data review from the last 2 OV-103 flights indicated that the nozzle temperatures rose while heaters were "on" and the supply water dump valve was closed. At no time was the supply water dump function inhibited by the fluctuation in nozzle temperature during STS-39.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	Supply water dump nozzle temperature drop.	Failure to melt ice in the nozzle would prevent the use of the primary supply water dump method, a Crit 1R3 condition. There are alternate supply water dump methods available, including routing supply water through the wastewater dump lines. Troubleshooting at KSC was unsuccessful in reproducing this anomaly. The anomaly was considered to be caused by a transient effect.
6	Auxiliary Power Unit (APU) #2 lube oil outlet pressure was low. IFA No. STS-39-V-11 (STS-39-K-01)	During entry, APU #2, S/N 301, lube oil outlet pressure was low, reading 25 pounds per square inch absolute (psia); nominal outlet pressure is 40-50 psia. Additionally, the minimum delta pressure (ΔP) across the pump was 20 psia during entry; nominal is 25-30 psia. The pump was certified to a ΔP of 25-30 psia and qualified to a minimum ΔP of 23 psia. During the pressure anomalies, APU #2 gearbox bearing temperatures #1 and #2 were within limits at 308°F maximum. Flight Rules require APU shutdown if the lube oil outlet temperature or gearbox bearing temperatures exceed 425°F. Loss of lube oil flow can result in high gearbox temperatures, leading to APU shutdown and loss of critical APU function. This was the first APU lube oil outlet pressure-low anomaly in the program. Previous lube oil anomalies were related to high lube oil outlet pressures. Troubleshooting at KSC determined that APU #2 was not properly serviced prior to STS-39 launch, resulting in a low quantity of lube oil in APU #2. For this reason, APU #2, S/N 301, was not removed from OV-103 prior to the next flight of this Orbiter vehicle on STS-48. This Orbiter IFA was subsequently transferred to a KSC IFA (STS-39-K-01) for procedural corrective action.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	Right-Hand (RH) outboard Main Landing Gear (MLG) tire excessive wear. IFA No. STS-39-V-12 HR No. ORBI-021 {AR}	Postflight inspection of the MLG tires found that the RH outboard tire showed signs of significant wear. The outer 3 plies were worn excessively; MLG tires have 16 plies. The RH outboard tire did not lose tire pressure as a result of the excessive wear. The data indicated touchdown occurred at 210 knots; nominal touchdown speed is between 185 knots and 210 knots. Orbiter sink rate was nominal at 2 ft/sec. The RH MLG tires contacted the runway approximately 216 ft earlier than the Left-Hand (LH) MLG tires. At initial touchdown, the vehicle centerline was 10 ft to the left of the runway centerline, drifting left at a rate of 3 ft/sec. The Commander initiated a right roll command and applied right rudder to correct this drift prior to nose landing gear touchdown. It is believed that this action resulted in shifting the vehicle weight to the RH outboard tire, contributing to the excessive wear. There was a 12-knot headwind and a 1-knot crosswind at the time of the landing. The roughness of the KSC Shuttle Landing Facility (SLF) runway may also have contributed to the excessive tire wear. Previous experience with tire wear has been limited to localized spin-up spots in the MLG tires; there has been no similar, uniform wear to this extent in the history of the Space Shuttle Program. The worst-case spin-up spot tire wear led to the failure of a MLG tire and cessation of the use of KSC as a planned end-of-mission landing site. The SLF is available for all missions as a primary, backup, or contingency landing site. Investigation into this anomaly concluded that the excessive tire wear was the result of environmental and crew performance dispersions. All MLG tires are replaced between flights.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8	Loss of communications during entry. IFA No. STS-39-V-13	Communications were lost during entry; this occurred ahead of schedule and for a longer period of time than predicted and normal. Postflight assessment of this anomaly indicated that the communications dropout was a result of the Orbiter's attitude during the high-inclination entry and the relative position of the Tracking and Data Relay Satellite (TDRS) during that portion of the reentry; there were no hardware or software problems found. Because of these conditions, the S-band antenna used for 2-way data and voice communications between the Orbiter and the ground was not in the required line-of-sight with the TDRS. Onboard navigation control would have provided sufficient data to the Commander and Pilot to achieve a safe landing if a hardware or software problem had caused the communications loss. Data dropouts were typical during Space Shuttle missions prior to the Tracking and Data Relay Satellite System (TDRSS) availability and use. Mission planning can minimize or preclude data and voice dropouts through attitude control. When attitude control is not available, mission planning can accurately predict the periods of communications dropouts.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	The Pilot's Rotation Hand Controller (RHC) bottomed-out during entry. IFA No. STS-39-V-14 HR No. ORBI-152 {C}	During entry, the Pilot was in control of the vehicle until approximately 3 min prior to touchdown. At that time, the Pilot turned control of the vehicle over to the Commander, who completed the landing. When the Pilot took his hand off the RHC, it dropped into the slot and bottomed-out. The Pilot indicated that he tried to pull it back up and attempted to lock it in place, but was unable to lock it because the adjustment knob was jammed. Postlanding inspection found that the adjustment knob was in the full counterclockwise, or loose position. The Commander and Pilot RHCs are adjustable in the up or down direction, as well as in the fore and aft direction. The RHC is normally locked into the desired position with 2 adjustment knobs. The lower knob, used for up and down adjustment, is a standard friction-type knob; turning it counterclockwise to loosen, clockwise to tighten. The Commander and Pilot nominally adjust their respective RHC prior to entry, with the 2 knobs, to best fit their relative hand position. The STS-39 pilot would have been able to use his RHC, if required, even though it would not have been adjusted to the optimum height.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1	High-Pressure Oxidizer Turbopump (HPOTP) secondary seal cavity pressure sensor failure on STS-39, engine #2029. IFA No. STS-39-E-01 HR No. ME-C1 (All Phases) {AR}	During the initial stages of STS-39 tanking, engine #2029 posted a Fault Identification (FID) during Purge Sequence Number (PSN)-3. The FID indicated that the HPOTP secondary seal cavity pressure, Channel "A", violated Launch Commit Criteria (LCC) sensor qualification limits (4 psia minimum, 20 psia maximum during PSN-1 through PSN-4) and was disqualified. Channel "A" indicated that the secondary seal cavity pressure had risen to 330 psia, while Channel "B" on the same sensor indicated the pressure to be 16 psia. The secondary seal cavity pressure sensor channels on the other SSME HPOTPs also read 16 psia. The disqualification of Channel "A" led to the scrub of the first STS-39 launch attempt. The LCC requires both secondary seal cavity pressure sensor channels to be within qualification limits prior to launch to protect redundancy should 1 channel fail during ascent. If disqualification of Channel "A" had been waived for launch and Channel "B" had failed high during ascent and was disqualified, the result would have been the loss of redline protection for that engine. In the event of loss of redline protection and exceedance of the redline limit, the result would be catastrophic loss of the engine and potential loss of the vehicle and crew. The calculated probability of the loss of HPOTP secondary seal cavity pressure redline protection, coupled with exceedance of the redline limit, is 1 in 213,000 launches for any 1 of 3 SSMEs. The failed pressure sensor and associated wire harness were removed from STS-39/OV-103 and sent to the Marshall Space Flight Center (MSFC) Huntsville Simulation Laboratory (HSL) for failure analysis. The sensor was installed in the HSL in the flight configuration and checked out under ambient conditions; no

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1 (Continued)	HPOTP secondary seal cavity pressure sensor failure on STS-39, engine #2029.	problems were found. The sensor was chilled using liquid nitrogen, and Channel "A" failed high. Harness connectors on Channel "A" and "B" were reversed at the sensor, and the sensor was again chilled. The same failure signature was identified on Channel "B", the expected result. A new sensor was installed, connected to the harness from STS-39, and chilled with liquid nitrogen. No failure was found, thus isolating the failure mode to the sensor. Following failure analysis at the HSL, the sensor was returned to the vendor for teardown inspection. Resistance and calibration checks performed at the vendor indicated that the fault was downstream of the bridge circuit at ambient temperature. Teardown inspection isolated the fault to the impedance board within the sensor. A fracture in the impedance board was visually evident after removal of the Room-Temperature Vulcanizing (RTV). The strain-gage grid network was found to be lifted in the vicinity of the fracture. The straight surface feature of the fracture was indicative of pre-existent damage. Surface "rounding" suggested the presence of the fracture before strain-gage grid etching. The epoxy overcoat was also found to be missing in the local area of the fracture. This was the first impedance board failure in the SSME Program; there was no evidence of a generic design problem. Hot-fire experience of 19 units with impedance boards from the same lot is in excess of 580 starts and 229,313 seconds (sec). Nine of these units have experienced a total of 43 flights. Impedance boards in the SSME Program have witnessed over 3,250,000 sec of hot-fire exposure; exposure in flight units has been documented to be in excess of 750,000 sec.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	RH Solid Rocket Motor (SRM) nozzle cowl/Outer Boot Ring (OBR) erosion/washing. IFA No. STS-39-M-01 HR No. BC-10 Rev. C {C}	Postflight inspection of the STS-39 RH nozzle cowl/OBR Carbon Cloth Phenolic (CCP) identified 11 circumferential areas of erosion/washing and wedgeouts. Erosion/washing and wedgeouts are typical in this region; however, the STS-39 RH cowl showed unusual, erratic erosion, ply lifting, and atypical short-ply wedgeouts. The investigation determined that these conditions occurred during motor operation. Based on preliminary visual inspection, the condition of the RH nozzle cowl/OBR was thought to be outside the SRM experience base. However, closer inspection at Thiokol determined that the total heat-affected depth and associated margins of safety comply with the Contractor End-Item (CEI) Specification char and erosion criteria. Measurements indicated that the wedgeouts were of similar size as those seen on previous flight motors for erosion and char depth. Additionally, the calculated margin of safety for the worst-case region was positive (15%). Review of postflight inspection of nozzle cowls/OBRs determined that material-affected depth is not greatly influenced by wedgeout or deeper erosion areas. Postflight assessments did, however, demonstrate that higher erosion corresponds with lower char depth. Conversely, areas with less erosion show more char. Statistical assessment of the Redesigned Solid Rocket Motor (RSRM) flight data base concluded that there was a high probability of meeting the 1.5 Factor of Safety (FOS) requirement. For the OBR, a 96.0% chance of meeting or exceeding the 1.5 FOS requirement was calculated. For the cowl, the probability of meeting the 1.5 FOS requirement was 99.2%. It was also determined statistically that the probability of violating a 1.0 FOS is less than 0.01% for both the OBR and cowl.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)	RH SRM nozzle cowl/OBR erosion/washing.	Trend assessment performed using RSRM historical flight data indicated no general trend of degraded performance or increased wedgeouts or washouts. The trend assessment determined that the STS-39 RH cowl wedgeouts represented the worst case seen to date for a flight motor. Relative to the OBR, a small decrease in the mean margin of safety was found; however, no violation of safety factor requirements was indicated. In the course of the STS-39 investigation, it was determined that the CCP used on the STS-39 RH cowl and OBR was from the same lot of material as both aft exit cones used on the STS-40 SRMs. For this reason, an investigation into the history of similar occurrences in aft exit cones was performed. It was learned that washout areas and CCP ply lifting had been observed on several static test nozzle aft exit cones (DM-7, ETM-1A, TEM-6, QM-7, PV-1, and FSM-1). Flight SRM exit cones are jettisoned after SRM separation and are not recovered. All static SRM aft exit cones exhibiting similar anomalies have performed successfully. A worst-on-worst case assessment, based on the anomalies found on the STS-39 RH cowl/OBR, indicated a 1.35 resulting safety factor. This was calculated with the assumption that CCP heat-affected depth was increased by 26%.

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

ΔP	Delta Pressure
ADI	Attitude Direction Indicator
AFB	Air Force Base
AFP-675	Air Force Program-675
AGC	Automatic Gain Control
AIS	Arizona Imager/Spectrograph
ALCA	Aft Load Controller Assembly
ALFE	Advanced Liquid Feed Experiment
amp	Ampere
AOS	Acquisition of Signal
APM	Ascent Particle Monitor
APU	Auxiliary Power Unit
AR	Accepted Risk
ARD	Abort Region Determinator
ASI	Augmented Spark Igniter
ATE	Automatic Test Equipment
BFS	Backup Flight Software
	Backup Flight System
C	Controlled
CA	California
CAR	Corrective Action Report
CAV	Captive Air Vent
CCP	Carbon Cloth Phenolic
CEI	Contractor End-Item
CIL	Critical Items List
CIRRIS	Cryogenic Infrared Radiance Instrumentation for Shuttle
CIV	Critical Ionization Velocity
CLOUDS	Cloud Logic to Optimize Use of Defense Systems
Crit	Criticality
CRO	Chemical Release Observation
CRT	Cathode Ray Tube

APPENDIX A

LIST OF ACRONYMS - CONTINUED

CTI	Charlton Technologies, Inc.
CVF	Circular Variable Filter
CWS	Caution and Warning System
DAR	Deviation Approval Request
dB	Decibel
DDU	Display Driver Unit
DEU	Data Entry Unit
	Display Electronic Unit
DoD	Department of Defense
DPS	Data Processing Software System
	Data Processing System
DR	Discrepancy Report
DSE	Data System Experiment
EAFB	Edwards Air Force Base
EDT	Eastern Daylight Time
EIU	Engine Interface Unit
EOD	Erasable Optical Disk
ET	External Tank
F	Fahrenheit
FAR UV	FAR Ultraviolet Camera
FASCOS	Flight Acceleration Safety Cutoff System
FCS	Flight Control System
FCV	Flow Control Valve
FD	Flight Day
FDA	Fault Detection and Annunciator
FES	Flash Evaporator System
FID	Fault Identification
FMEA	Failure Modes and Effects Analysis
FMEA/CIL	Failure Modes and Effects Analysis/Critical Items List
FOS	Factor of Safety
FP	Fuel Pump
FR	Flight Rule
FRI	Flow Recirculation Inhibitor
FRR	Flight Readiness Review
ft	Feet
ft/sec	Foot Per Second

APPENDIX A

LIST OF ACRONYMS - CONTINUED

GAS	Get Away Special
GG	Gas Generator
GGVM	Gas Generator Valve Module
GH_2	Gaseous Hydrogen
GLOS	Gaseous Luminosity of Optical Surface
GN&C	Guidance, Navigation and Control
GO_2	Gaseous Oxygen
GOAL	Ground Operations Aerospace Language
GOX	Gaseous Oxygen
GPC	General Purpose Computer
GSE	Ground Support Equipment
H_2	Hydrogen
HAC	Heading Alignment Cone
HCF	High-Cycle Fatigue
HGDS	Hazardous Gas Detection System
HPFTP	High-Pressure Fuel Turbopump
HPM	High Performance Motor
HPOTP	High-Pressure Oxidizer Turbopump
HPU	Hydraulic Power Unit
HR	Hazard Report
hr	Hour
HSL	Huntsville Simulation Laboratory
HUP	Horizon Ultraviolet Program
I/O	Input/Output
IBR	Inner Boot Ring
IBSS	Infrared Background Signature Survey
ICD	Interface Control Document
IFA	Inflight Anomaly
IMU	Inertial Measurement Unit
INTG	Integration
IPL	Initial Program Load
isp	Specific Impulse
IUS	Inertial Upper Stage
KSC	Kennedy Space Center
kt	Knot

APPENDIX A

LIST OF ACRONYMS - CONTINUED

L-2	Launch Minus 2 Day
lb	Pound
lbf	Pounds Force
LCC	Launch Commit Criteria
LCF	Low-Cycle Fatigue
LH	Left-Hand
LH_2	Liquid Hydrogen
LLCO	Low-Level Cutoff
LO_2	Liquid Oxygen
LOS	Loss of Signal
LOX	Liquid Oxygen
LPFTP	Low Pressure Fuel Turbopump
LPOTP	Low-Pressure Oxidizer Turbopump
LPS	Launch Processing Set
LSFR	Launch Site Flow Review
MCC	Main Combustion Chamber
	Mission Control Center
MDM	Multiplexer-Demultiplexer
ME	Main Engine
MEC	Master Event Controller
	Main Engine Controller
MECO	Main Engine Cutoff
MET	Mission Elapsed Time
min	Minute
MLG	Main Landing Gear
MLP	Mobile Launch Platform
MMH	Monomethyl Hydrazine
MMU	Mass Memory Unit
MPEC	Multi-Purpose Experiment Canister
MPS	Main Propulsion System
MRB	Material Review Board
ms	millisecond
MSA	Marshall Spray-On Ablative
MSE	Mission Safety Evaluation
MSFC	Marshall Space Flight Center
N_2O_4	Nitrogen Tetroxide
NASA	National Aeronautics and Space Administration

APPENDIX A

LIST OF ACRONYMS - CONTINUED

NBAT	Nominal Bus Assignment Table
NDE	Nondestructive Evaluation
NSI	NASA Standard Initiator
NSRS	NASA Safety Reporting System
O_2	Oxygen
OBR	Outer Boot Ring
OMRSD	Operational Maintenance Requirements and Specifications Document
OMS	Orbital Maneuvering System
OPF	Orbiter Processing Facility
OPS	Operational Sequence
	Operations
OSMQ	Office of Safety and Mission Quality
OV	Orbiter Vehicle
P/N	Part Number
PAR	Prelaunch Assessment Review
PASS	Primary Avionics Software System
P_c	Chamber Pressure
PF2	Payload Forward 2
PFS	Primary Flight System
POR	Power-On Reset
ppm	Parts Per Million
PRCB	Program Requirements Control Board
PRSD	Power Reactant Supply and Distribution
psi	Pounds Per Square Inch
psia	Pounds Per Square Inch Absolute
psid	Pounds Per Square Inch Differential
psig	Pounds Per Square Inch Gage
PSN	Purge Sequence Number
QD	Quick Disconnect
QINMS	Quadruple Ion-Neutral Mass Spectrometer
RCS	Reaction Control System
RGA	Rate Gyro Assembly
RH	Right-Hand
RHC	Rotation Hand Controller
RI	Rockwell International

APPENDIX A

LIST OF ACRONYMS - CONTINUED

RM	Redundancy Management
RME	Radiation Monitoring Equipment
RMS	Redundancy Management System
	Remote Manipulator System
ROEU	Remotely Operated Electrical Umbilical
RSRM	Redesigned Solid Rocket Motor
RSS	Range Safety System
RTLS	Return-to-Launch Site
RTV	Room-Temperature Vulcanizing
S/N	Serial Number
SAIL	Shuttle Avionics Integration Laboratory
sccm	Standard Cubic Centimeters Per Minute
sccs	Standard Cubic Centimeters Per Second
scfm	Standard Cubic Feet Per Minute
scim	Standard Cubic Inches Per Minute
SDI	Strategic Defense Initiative
sec	Second
SEL	Select
SEM	Scanning Electron Microscope
SII	Solid Rocket Motor Ignition Initiator
SIT	Shuttle Interface Test
SKIRT	Spacecraft Kinetic Infrared Test
SLF	Shuttle Landing Facility
SM	System Management
	Systems Management
SMS	Shuttle Mission Simulator
SOV	Shutoff Valve
SPAS-II	Shuttle Pallet Satellite-II
Spec	Specification
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSC	Stennis Space Center
SSME	Space Shuttle Main Engine
SSRP	System Safety Review Panel
SSV	Space Shuttle Vehicle
STP-01	Space Test Program-01

APPENDIX A

LIST OF ACRONYMS - CONTINUED

TAEM	Terminal Area Energy Management
TAL	Transatlantic Abort Landing
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TEM	Test and Evaluation Motor
TFL	Telemetry Format Load
TPS	Thermal Protection System
TVC	Thrust Vector Control
U/N	Unit Number
UDMH	Unsymmetric Dimethyl Hydrazine
URA	Uniformly Redundant Array
USBI	United Space Boosters, Inc.
UVLIM	Ultraviolet Limb Imaging Experiment
UVPI	Ultraviolet Plume Instrument
VAB	Vehicle Assembly Building
WSB	Water Spray Boiler
WSTF	White Sands Test Facility