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MISSION SAFETY EVALUATION REPORT FOR STS-38

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-38

Postflight Edition: December 21, 1990

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EXECUTIVE SUMMARY

After several months delay due to Liquid Hydrogen (LH_2) leaks in the External Tank (ET) 17" LH_2 umbilical disconnect, *Atlantis* was launched at 6:48 p.m. Eastern Standard Time (EST) on November 15, 1990. *Atlantis* was cleared for launch after the replacement of the ET 17" umbilical disconnect and a successful tanking test on October 24, 1990. The STS-38 Department of Defense mission was the seventh mission for *Atlantis* and the fifth night launch in the Space Shuttle Program history. No technical concerns were encountered during the countdown.

The end of the STS-38 mission was extended by one day because of high crosswinds at the primary landing site, Edwards Air Force Base (EAFB), California. On the next day, November 20, 1990, the winds were too high for a safe landing at EAFB, and a decision was made to select the Shuttle Landing Facility (SLF) at the Kennedy Space Center (KSC) as the primary landing site. The deorbit burn was performed on orbit 79 of the STS-38 mission. Reentry and descent were effected with no problems. *Atlantis* touched down on SLF runway 33 at 4:42 p.m. EST on November 20, 1990. This was the sixth Space Shuttle landing at KSC.

Postlanding inspection of *Atlantis'* exterior found little evidence of damage. There has been a concern about returning an Orbiter to KSC, because the last mission to do so experienced tire failure. *Atlantis'* tires and brakes were in excellent condition with no indication of excessive wear.

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 STS-38 flight were also documented prior to the STS-38 Flight Readiness Review (FRR) (FRR Edition) and the STS-38 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-38 safety risk factors that represent a change from previous flights, factors from previous flights that had an impact on this flight, and factors that were 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-38 mission, including launch data, crew size, mission duration, launch and landing sites, and other mission-related information.

Section 3 - Contains a list of safety risk factors/issues, considered resolved or not a safety concern prior to STS-38 launch, that were impacted or repeated by anomalies reported for the STS-38 flight.

Section 4 - Contains a list of safety risk factors that were considered resolved for STS-38.

Section 5 - Contains a list of Inflight Anomalies (IFAs) that developed during the STS-41 mission, the previous Space Shuttle flight.

Section 6 - Contains a list of IFAs that developed during the STS-36 mission, the previous flight of the Orbiter Vehicle (OV-104).

Section 7 - Contains a list of IFAs that developed during the STS-38 mission. Those IFAs that are 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 (in notebook format) presentation data, white papers, and other documentation. These data were used to support the resolution rationale or retention of open status for each item discussed in the MSE.

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

SECTION 2

STS-38 MISSION SUMMARY

2.1 Summary Description of STS-38 Mission

After several months delay due to Liquid Hydrogen (LH_2) leaks in the External Tank (ET) 17" LH_2 umbilical disconnect, *Atlantis* was launched at 6:48 p.m. Eastern Standard Time (EST) on November 15, 1990. *Atlantis* was cleared for launch after the replacement of the ET 17" umbilical disconnect and a successful tanking test on October 24, 1990. The STS-38 Department of Defense (DoD) mission was the seventh mission for *Atlantis* and the fifth night launch in the Space Shuttle Program history. No technical concerns were encountered during the countdown.

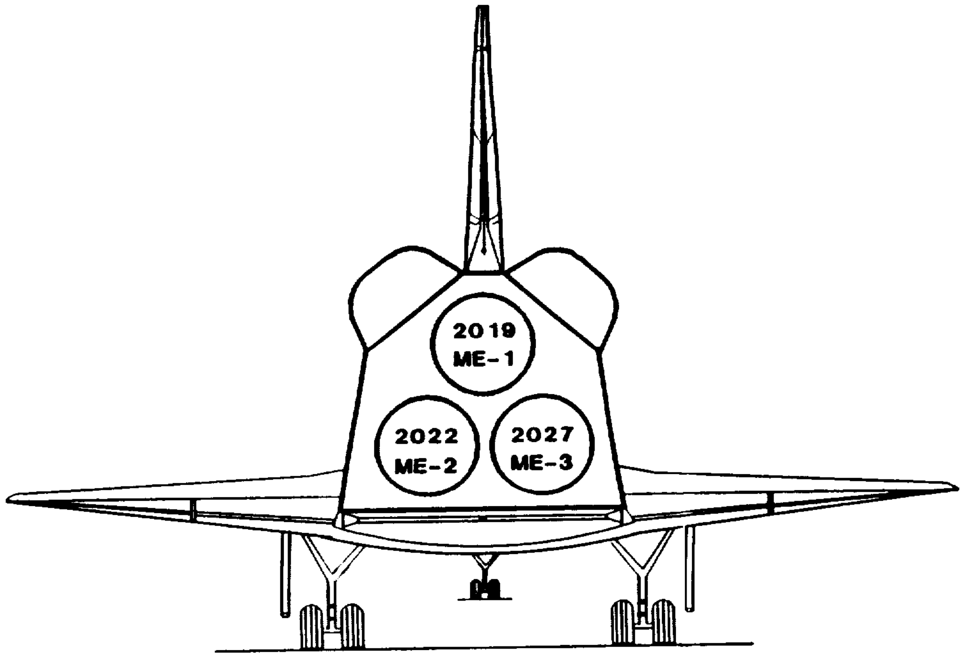
During this classified mission, no reports or accounts were given relative to mission operations. Based on the limited number and relatively benign Inflight Anomalies (IFAs) recorded, the mission was considered extremely successful. Some unidentified debris was observed falling from the base of both Solid Rocket Boosters (SRBs) during ascent; however, it is believed that this debris was aluminized tape used during prelaunch SRB closeout operations. Postflight inspection found no indication of adverse heat effects in the region of the Thrust Vector Control systems components. Disassembly of both Solid Rocket Motor (SRM) igniter joints found putty blowholes and resulting sooting. Damage to the Gask-O-Seal retainer cadmium plating was also reported. The sooting and damage to the cadmium plating were considered to be within the experience base of recent missions and, therefore, was not recorded as an IFA. Putty blowholes are an expected occurrence and will be the norm on future SRMs until a redesign of the igniter joint, which includes removing the putty, is implemented in late 1991.

The end of the STS-38 mission was extended by one day because of high crosswinds at the primary landing site, Edwards Air Force Base (EAFB), California. On the next day, November 20, 1990, the winds were too high for a safe landing at EAFB, and a decision was made to select the Shuttle Landing Facility (SLF) at the Kennedy Space Center (KSC) as the primary landing site. The deorbit burn was performed on orbit 79 of the STS-38 mission. Reentry and descent was effected with no problems. *Atlantis* touched down on SLF runway 33 at 4:42 p.m. EST on November 20, 1990. This was the sixth Space Shuttle landing at KSC.

Postlanding inspection of *Atlantis'* exterior found little evidence of damage. There has been a concern about returning an Orbiter to KSC, because the last mission to do so experienced tire failure. *Atlantis'* tires and brakes were in excellent condition with no indication of excessive wear. *Atlantis* rolled approximately 8,900 feet after touchdown.

2.2 Flight/Vehicle Data

- Launch Date: November 15, 1990
- Launch Time: 6:48 p.m. EST
- Launch Site: KSC Pad 39A
- RTLS: Kennedy Space Center, Shuttle Landing Facility
- Landing Site: Kennedy Space Center, Shuttle Landing Facility
- Landing Date: November 20, 1990
- Landing Time: 4:42 p.m. EST
- Mission Duration: 4 days, 21 hours, 54 minutes, 28 seconds
- Crew Size: 5
- Orbiter: OV-104 (7) *Atlantis*
- SSMEs: (1) #2019, (2) #2022, (3) #2027
- ET: ET-40
- SRBs: BI-039
- SRMs: RSRM Flight Set #12
- MLP: MLP #1



ENGINE	#2019	#2022	#2027
POWERHEAD	#2020	#2022	#4004
MCC*	#2023	#4006	#2024
NOZZLE	#2024	#2023	#2027
CONTROLLER	F4	F7	F23
FASCOS*	#23	#22	#20
HPFTP*	#6008	#6007	#4109R1
LPFTP*	#2022R1	#2024	#4005
HPOTP*	#2323R3	#4107R3	#9309R1
LPOTP*	#2025R1	#2104R1	#4302

* Acronyms can be found in Appendix A.

2.3 Partially Fixed Orifice Flow Control Valves

STS-38/OV-104 is the second Space Shuttle that flew with partially fixed orifice Gaseous Oxygen (GO_2) Flow Control Valves (FCVs). The current active GO_2 FCVs used in the External Tank (ET) Liquid Oxygen (LO_2) tank pressurization system will be replaced by Main Propulsion System (MPS) GO_2 fixed orifices. The GO_2 FCVs are used to return gas from the Main Engines back to the ET to maintain ullage pressure. The fixed orifice has been approved for Space Shuttle fleet implementation pending operational assessment and performance verification [Program Requirements Control Board Document (PRCBD) S50509R2]. Approval was based on results of the feasibility assessment performed by Space Shuttle Engineering/Level II and the supporting contributions of the Marshall Space Flight Center (MSFC) ET Project. Incorporation of the fixed orifice hardware and procedures into the MPS eliminates some of the criticality (Crit) 1 and 1R failure modes associated with the current FCVs which include the valves, the associated electronics, and the pressure-sensing transducers.

Flight tests with the FCVs shimmed for intermediate flow rates will be used to verify the analytical model and the final orifice size selection. The FCV stroke between high-flow and low-flow positions will be gradually reduced over the course of 3 flights until the valves are fixed in one position. The ET GO_2 vent/relief valve pressure was increased from 24 pounds per square inch gage (psig) to 31 psig on STS-38 by replacing the spherical spring sensing assembly. This reduced the minimum ullage pressure from 16.0 psig to 14.2 psig, reduced 4 Crit 1R failure modes to Crit 3, and eliminated the effects of 15 other failure modes. The 31-psig GO_2 vent/relief valve was test qualified. The current schedule leading to the implementation of fixed FCVs is as follows:

Flight	Vehicle		High Flow	Low Flow
STS-31	OV-103		100%	42%
STS-41	OV-103	**Step 1**	93%	55%
STS-38	OV-104	**Step 2**	85%	66%
STS-35	OV-102	**Step 1**	93%	55%
TBD	TBD	**Fixed**	78%	---

STS-41 was the first of the series of flights with partially fixed orifices that will lead to implementation of a fixed orifice FCV. Final implementation of GO_2 fixed orifices has not been scheduled. The current selection for the final fixed setting is at 78% flow; however, this value is subject to review and change in accordance with the test results from the previous flights with partially-shimmed FCVs.

2.4 STS-38 Booster Assemblies

The Solid Rocket Boosters (SRBs) for STS-38, BI-039, flew with the Ordnance Ring Pin and Pin Retainer modification. This is the first step in resolving the loss of Ordnance Ring Pins experienced on many flights, the last on STS-36 (see Section 6, SRB 1 for details). The modification incorporates a deeper groove in the pin that results in a tighter fit between the pin and the outer retainer legs. The center tang on the retainer was modified to fit over the pin head. Six new design retainers and pins were installed on each SRB.

The propellent cast in the right-hand aft Solid Rocket Motor (SRM) segment contained aluminum from the Reynolds Aluminum Company. Reynolds was qualified as 1 of 3 aluminum suppliers in 1985; however, their product has not yet flown. Aluminum powder from Reynolds was used on all segments of the Development Motor (DM)-7 static test motor in 1985. Post-test analysis of the demonstrated ballistic parameters showed that all requirements were achieved and the motor performance was close to predicted. Reynolds aluminum powder was also used to cast segments used in Qualification Motors (QM)-6, DM-9, QM-7, and QM-8.

2.5 Payload Data

The payload is classified.

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

SECTION 3

SAFETY RISK FACTORS/ISSUES IMPACTED BY STS-38 ANOMALIES

This section lists safety risk factors/issues, considered resolved (or not a safety concern) for STS-38 prior to launch (see Sections 4, 5, and 6), that were repeated or related to anomalies that occurred during the STS-38 flight (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-38 Safety Risk Factors		
Orbiter 14	STS-38/OV-104 Reaction Control System (RCS) thruster oxidizer leaks.	RCS thruster R1U exhibited low Chamber Pressure (P_c) on first firing during STS-38 operations. Performance during this firing was nominal, except for the low P_c. The crew manually transitioned R1U to the lowest priority, but did not deselect it. Postlanding R1U decay tests determined a pressure drop of 6 to 8 pounds per square inch (psi) in a 2-hour period. Sniff checks have indicated no leak. (See Section 7, Orbiter 7 for further details). (IFA No. STS-38-07)
SSME 3	POGO precharge pressure software anomaly.	A post-Main Engine Cutoff (MECO) Fault Identification (FID) on engine #1 was presented during STS-38. This FID indicated that the POGO precharge pressure transducer failed qualification limits on the low side. POGO precharge pressure limits were implemented in AR01, a recent Space Shuttle Main Engine (SSME) controller software upgrade, and used on STS-38; however, because this FID was presented after nominal engine performance and shutdown, it was not considered a problem. For STS-39 and subsequent flights, POGO precharge pressure qualification limits will be inhibited after MECO +16 seconds to preclude recurrence of this problem.

ITEM		COMMENT

Section 4: Resolved STS-38 Safety Risk Factors

SRM 6	Solid Rocket Motor (SRM) Ignition Initiator (SII) leak test.	Disassembly and inspection of STS-38 SIIs did not identify grease in the leak check ports. An indentation was found on the left-hand SII O-ring; however, there was no resulting blowby. On the right-hand SII, raised metal (described as a bubble) was found in the leak check port; it was believed to be the result of the original machining process. Neither of these findings inhibited SII performance and were not considered safety concerns.

Section 6: STS-36 Inflight Anomalies

Orbiter 3	RCS thruster R3D failed "off" during External Tank separation. IFA No. STS-36-04	RCS thruster R1U exhibited low P_c on first firing during STS-38 operations. Performance during this firing was nominal, except for the low P_c. The crew manually transitioned R1U to the lowest priority, but did not deselect it. Postlanding R1U decay tests determined a pressure drop of 6 to 8 psi in a 2-hr period. Sniff checks have indicated no leak. (See Section 7, Orbiter 7 for further details). (IFA No. STS-38-07)
Orbiter 12	Thruster R4R failed "off" during pre-entry hot-fire test. IFA No. STS-36-12	RCS thruster R1U exhibited low P_c on first firing during STS-38 operations. Performance during this firing was nominal, except for the low P_c. The crew manually transitioned R1U to the lowest priority, but did not deselect it. Postlanding R1U decay tests determined a pressure drop of 6 to 8 psi in a 2-hr period. Sniff checks have indicated no leak. (See Section 7, Orbiter 7 for further details). (IFA No. STS-38-07)

SECTION 4

RESOLVED STS-38 SAFETY RISK FACTORS

This section contains a summary of the safety risk factors that were considered resolved for STS-38. 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, contained 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.

Orbiter 7	Auxiliary Power Unit Gas Generator Valve Module issue.
Orbiter 10	OV-102 20-psi helium regulator leak.
Orbiter 11	Fuel Cell separator plate plating defects.
SRM 2	Putty on igniter inner gasket of test motors.
SRM 3	STS-31 right Solid Rocket Motor igniter adapter-to-forward dome joint putty blowhole.

SECTION 4 INDEX

RESOLVED STS-38 SAFETY RISK FACTORS

SECTION 4 INDEX - CONTINUED

RESOLVED STS-38 SAFETY RISK FACTORS

SECTION 4 INDEX - CONTINUED

RESOLVED STS-38 SAFETY RISK FACTORS

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Liquid Hydrogen (LH_2) leaks on STS-35/OV-102 and STS-38/OV-104. HR No. INTG-006A {AR} INTG-071 {AR} ORBI-306 {AR} ***There were no anomalies associated with Hydrogen (H_2) leaks previously experienced on STS-38/OV-104. H_2 levels at the LH_2 umbilical and in the aft compartment were within the OV-104 experience base.***	Significant LH_2 leaks were experienced on STS-35/OV-102 and STS-38/OV-104. Extensive investigation of these leaks was performed, and a summary of associated activities is found in the STS-35 MSE, L–1 Update Edition, September 17, 1990. Preparations for the launch of STS-38/OV-104 included additional measurements and inspections to define the condition of the mated External Tank (ET) and Orbiter LH_2 disconnect. Each side of the disconnect interface was examined and measured; no anomalies were reported. The primary and secondary interface seals were examined under 10-power magnification for contamination prior to installation. The Orbiter disconnect had flown previously. A new, 6000-series ET disconnect, Serial Number (S/N) 6812, was installed on ET-40. S/N 6812 was acceptance tested with the Orbiter disconnect simulator and exhibited no H_2 leakage during the test. The ET disconnect, 17" flange bolts were torqued to 660 inch-pound (in-lb). A tanking test was performed on STS-38/OV-103 on October 24, 1990. No significant H_2 concentration was noted either at the 17" disconnect or in the aft compartment. The Launch Commit Criteria (LCC) for Orbiter/ET 17" umbilical H_2 concentrations was readdressed for STS-35 and was also in effect for STS-38. After considering the results of the LH_2 leak investigation and extensive review of past and recent test data, the following maximum redline requirements were established in the STS-38 LCC: • No presence of unusual vapors and liquid droplets. The term "unusual vapors and liquid droplets" was defined as: – An obvious blowing leak or a vapor cloud which obscures the disconnect or feedline region for an extended period [>5 minute (min)]. – Consistent frequent liquid drops falling or flowing with identifiable vapor trails.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)		• No H_2 concentration greater than 40,000 parts per million (ppm) (4%) on both external sensors [Leak Detectors (LDs) 54 and 55]. If a sensor is declared failed, the remaining sensor could not exceed 40,000 ppm (4%). • No H_2 concentration greater than 20,000 ppm (2%) on 1 of 2 sensors (LDs 54 and 55) without evaluation of available data by the Mission Management Team (MMT) and MMT approval to continue the launch countdown. • If intermittent or erratic readings occur, the data would be evaluated over a 10-min period to determine the actual H_2 concentration. The LCC for aft compartment H_2 concentration remained the same for STS-38/OV-104; at 500 ppm maximum during fast fill and 300 ppm maximum during stable replenish. Safety concurred with this LCC for STS-38. Rationale for STS-38 flight was: • The ET 17" disconnect installed on ET-40 was new and passed acceptance testing with the Orbiter simulator. No anomalies were reported relative to the Orbiter disconnect. • LCC requirements protected against launching with an excessive H_2 leak. • STS-38/OV-104 successfully passed a tanking test. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2	Holddown Post (HDP) cracks. HR No. INTG-158B {AR} ***No anomalies were attributed to the crack found on HDP #7. Postflight inspection of HDP #7 determined that the identified crack did not grow and there were no cracks found on the other HDPs.***	During strain gage replacement on the Mobile Launch Platform (MLP) #2 HDPs after STS-41, a crack was found on HDP #3. The crack was determined to be 2" long x 1/4" deep. Based on this finding, inspection of all MLP HDPs was directed. MLP #1 (STS-35 at pad A) inspection revealed a crack in HDP #7, 3/8" long, 11" from the base, and 3" clockwise from the setscrew. This was the only crack found on MLP #1 HDPs. MLP #3 (STS-38 at pad B) inspection found a similar crack on HDP #7, 1/2" long, 4" from the base, and 2" counterclockwise from the setscrew. No other HDPs on MLP #3 were found with cracks. Cracks found on MLP #1 and MLP #3 were considered surface cracks, with no appreciable depth. Cracks of this type were previously experienced with the old HDP design. The new HDP design is currently in use and includes the 3 HDPs with cracks; it is made from the same material, but provides better protection from plume impingement. Analysis determined that cracking was due to thermal loads applied to the HDPs during launch; not mechanical loading. The thermal loads are the direct result of Solid Rocket Booster (SRB) plume impingement on the MLP deck surfaces, followed immediately by quenching with water from the deluge system. HDPs #3, #4, #7, and #8 experienced the highest concentration of plume impingement of the 8 MLP HDPs. Stress analysis indicated that the cracks are in a low stress area; average stress in the crack vicinity is 6,000 pounds per square inch (psi). Review of data from strain gages located 24" above the HDP base showed stress values of 8400 psi. On the average, stress increases with distance from the HDP base and, therefore, stress at the identified crack locations were estimated to be less. HDPs were designed for a minimum stress of 85,000 psi. Based on stress calculations at the crack site, the resulting Factor of Safety (FOS) was above 10. Critical flaw size was determined by analysis to be 40" in length and completely through the 5" thick HDP. It was not believed, therefore, that there was an appreciable reduction in this FOS due to the observed surface cracking.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
2 (Continued)		Rationale for STS-38 flight was: • The crack on MLP #3 (STS-38) is in a low stress area of HDP #7. • Cracks are a result of thermal cycling; not mechanical loading. • Similar cracks were experienced on older design HDPs; however, the new design HDP, currently in use, provides increased plume impingement protection. • A calculated FOS greater than 10 was assured, even with existing cracks. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1	Nose Landing Gear (NLG) axle bearing nut. HR No. ORBI-179 {AR} ***The NLG axle bearing nuts were found properly in place during postflight inspection.***	During installation of the nose wheel assembly on STS-35/OV-102 at Kennedy Space Center (KSC), a quality inspector noticed wheel assembly free play of approximately 0.007". Further inspection with Menasco representatives (nose and main landing gear assembly vendor) found that the nose wheel bearing retainer nuts on each side of the bearing housing were cocked. Discoloration was also noted on the axle and bearing housing. The bearing retainer nut is an aluminum alloy, male-threaded nut used to retain the landing gear axle roller bearing. There is a retainer nut on each end of the housing. Inspection of other NLG assemblies, including the OV-104 NLG, was undertaken to determine if similar problems existed. OV-104 inspection found the bearing retainer nuts to be cocked and evidence of free play similar to that seen on OV-102. No evidence of discoloration of the axle or housing was found. New retainer nuts were installed on the OV-104 axle; inspection showed that the housing threads were unaffected. Stress analysis of the nut cracking mechanism and possible loads, coupled with visual inspection of the retainer nut, led to the determination that the nut would stay in a wedged-in position after a "1-thread" lateral displacement without further radial motion. The nut geometry limits rotation for all design cases. (See STS-31 MSE, L–1 Update, April 23, 1990, Section 4, Orbiter 8, for further details.) Rationale for STS-38 flight was: • New retainer nuts were installed on the STS-38/OV-104 NLG axle. ***This risk factor was resolved for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2	Untempered martensite in the NLG. HR No. ORBI-179 {AR} ***There were no NLG anomalies reported on STS-38/OV-104.***	During the inspection of the NLG retainer nut problem at Menasco, teardown and inspection of the OV-102 NLG were performed. (See this section, Orbiter 1.) During the inspection, a 2" long x 0.22" wide x 0.41" deep area of untempered martensite was found on the inside of the 300M steel axle housing. The formation of the untempered martensite was determined to be caused by contact of the rotating Inconel 718 axle with the housing during touchdown. There was a raised area in the center of the axle used for centering the axle in the housing. Menasco determined the depth of the untempered martensite to be 0.41" by successively removing a thin layer, polishing to 125 rms, and performing an etch inspection. The total depth removed was 0.061" that included the overtempered (softened) zone surrounding the untempered martensite. The reworked axle, with the centerline ridge removed, will not contact the housing under anticipated slapdown load conditions. Inspection of the OV-104 NLG housing found no untempered martensite. Contact was made with the housing; however, this contact did not penetrate the housing paint. Corrective action was taken to remove the raised area and replace it with a painted strip for use in centering the axle. Rationale for STS-38 flight was: • No untempered martensite was found during NLG housing inspection. • The raised area on the axle was removed to preclude contact with the housing. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3	OV-105 Gaseous Hydrogen (GH_2) Flow Control Valve (FCV) broken poppet. HR No. INTG-151 {AR} ET S.09 {AR} ***There were no GH_2 FCV anomalies reported on STS-38/OV-104.***	During acceptance testing at Wyle Laboratories in May 1990, an OV-105 GH_2 FCV failed. This failure occurred after approximately 11 seconds (sec) of low GH_2 flow through the FCV and prior to the first valve cycle. Failure investigation determined that the valve poppet was broken. The poppet is made from Corrosion Resistant Steel (CRES) 440C. Only 4 poppets were manufactured in the same lot: 3 for OV-105 FCVs and 1 spare. During acceptance testing of another OV-105 GH_2 FCV on June 18, 1990, a valve poppet broke in a manner similar to the previous failure. This failure occurred 5 sec into the test with the FCV in the high-flow position. Of the 4 FCVs with poppets made from the same lot: 2 failed; 1 passed; and the 1 spare has yet to be tested. The concern was that the failed valves were of the same configuration as those in the Orbiter fleet. Visual inspection of both failed valves revealed similar fractures; both had a crescent-shaped section of poppet outer rim material missing. The first failed poppet had a greater amount of material missing than the second. The lost poppet material was recovered from the test stand debris trap downstream. Inspection of the test stand upstream filter verified it as clean and undamaged in both cases. Rationale for STS-38 flight was: • This type of fracture would not result in total loss of the poppet. Downstream impact of poppet material would not be a problem because of low flow rate, unless the total poppet came loose. A fractured poppet has the same effect as a valve which failed open (Crit 1R2). • OV-105 FCV poppet failures appeared to be material or process related. • There were no similar FCV poppet failures during testing or flight. ***This risk factor was acceptable for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	External leak of STS-35/OV-102 1/4" Liquid Oxygen (LO_2) flex hose. HR No. ORBI-306 {AR} INTG-151 {AR} ***There were no flex hose anomalies reported on STS-38/OV-104.***	During Operational Maintenance Requirements and Specifications Document (OMRSD) helium signature leak check on STS-35/OV-102 flex hoses, a 32.5-standard cubic inch per minute (scim) external leak was found in the LO_2 1/4" 20-psi regulator sense flex hose. The specification leak rate limit is less than 3.7×10^{-5} scim. This leak was not detected during Orbiter Processing Facility (OPF) Oxygen (O_2) system decay checks. The leaking flex hose and the LH_2 1/4" flex hose were replaced. Details of the 1/4" flex hose failure analysis are addressed in Section 4, Orbiter 12, STS-35 MSE L-1 Update Edition, September 17, 1990. Other flex hose sense lines, the OV-102 LH_2, the OV-104 LO_2, and the OV-103 LO_2, were returned to Rockwell International (RI) for examination prior to the STS-35 launch attempt. No problems were identified with these lines. The OV-104 LO_2 sense line was replaced with a new hose. All other OV-104 flex hoses were visually inspected for external damage and bird-caging and were found acceptable. Helium leak checks were performed with no anomalies reported. Another visual inspection for external damage was conducted on the STS-38/OV-104 flex hoses prior to aft compartment closeout. Rationale for STS-38 flight was: • The 1/4" LO_2 sense line on OV-104 was new; the LH_2 sense line was inspected with no apparent bird-caging. Leak and decay checks were successfully performed prior to the launch. • Inspection of the other flex lines was completed with no bird-caging found. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Outer ring movement on Preload Indicating (PLI) washers on LH_2 Orbiter/Space Shuttle Main Engine (SSME) interface on OV-103. HR No. ORBI-306 {AR} ***There were no anomalies attributed to this interface on STS-38/OV-104.***	During STS-31/OV-103 postflight inspection, outer rings of several PLI washers were found loose on the LH_2 Orbiter/SSME interface joint F.1. There are 36 3/8" bolts and PLIs on each flange, 1 per engine. Six outer rings were found loose on the engine #2011 LH_2 line, 7 each on engines #2031 and #2107. Leak checks at joint F.1 were performed on OV-103 engines with no leakage found. Gaps on loose outer rings were measured to be less than 0.0015". Installation torque was verified to be 360 in-lb or greater, and outer rings on the engine #2107 line were still loose. Breakaway torque was also verified on all engines to be 300-600 in-lb. OV-104 was also examined during this investigation. Engine #2022 F.1 joint was found with 1 loose outer ring, engine #2027 with 2. All joints were previously leak checked with zero leakage reported. Stress analysis performed by RI/Downey determined that the calculated load for an outer ring gap of 0.0015" was greater than 2500 pounds (lb). Cryogenic effects were determined to be insignificant. Calculated bolt load at 360 in-lb torque was approximately 6000 lb. The investigation concluded that the OV-103 and OV-104 installations were acceptable. Minimum preload was maintained with the PLI washer outer ring free to rotate with less than 0.0015" gap. All F.1 joints on OV-104 passed leak checks. PLI washer outer rings were verified not to rotate freely at installation, and the preload was maintained. Rationale for STS-38 flight was: • All F.1 joints on STS-38/OV-104 passed leak checks. • Minimum preload was maintained while allowing the PLI washer outer ring to rotate. ***This risk factor was resolved for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6	Wing struts below-minimum wall thickness. HR No. ORBI-277 {C} ***There was no indication of structural degradation of the OV-104 wings during STS-38 postflight inspection.***	Numerous wing strut tubes in OV-103 and OV-104 were found with below-minimum wall thickness. The concern was that reduced strut wall thickness may lead to structural failure. Postflight inspection of OV-103 after STS-31 revealed a damaged strut tube in the Left-Hand (LH) wing. This particular strut was replaced. Failure analysis showed that the buckled area occurred in a portion of the tube with below-minimum wall thickness (0.012" instead of the required 0.018" minimum). RI analysis also concluded that the damage was not caused by flight loads, but most likely was caused by work-related damage. There are 240 wing struts on each Orbiter wing. The tubing is purchased as seamless drawn 2024 aluminum with a 0.095" ± 10% wall thickness. The formed tube is chemically milled on the outer diameter to the specified wall thickness. The completed tube is ultrasonically inspected for wall thickness at 3 axial locations, 4 circumferential points per location. RI ultrasonically measured wall thicknesses of all wing struts in OV-103 and OV-104 with a margin of safety of 0.35 or less. OV-103 had 5 struts below the minimum in the LH wing and 10 in the right. OV-104 had 2 in the LH wing and 2 in the right; 2 of the 4 were determined to have a negative margin of safety. Doublers were installed on these 2 struts, thereby increasing the margin of safety to a positive level. These struts will eventually be replaced. The maximum below-minimum wall thickness condition is 0.006" that was found on the original strut. Other struts were 0.001-0.003" below the minimum. OV-103 and OV-104 struts were ultrasonically inspected at 8 circumferential locations at approximately mid-length. If they were not acceptable, both ends were also checked. Ultrasonic inspection was performed through a coating using a digital readout to give actual tube thickness. Individual tube thickness was not uniform. RI performed a stress analysis to determine the margin of safety for each strut in OV-104 that was undersized, based upon design loads. If the struts having a margin of safety of 0.10 or less were assumed to be the worst-case condition found so far (i.e., 0.006" under the

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6 (Continued)		minimum), then their margins of safety were negative. The thin wall areas are believed to be due to improper chemical milling. Rationale for STS-38 flight was: • STS-38/OV-104 wing inspection provided the location of all undersized struts. Doublers were installed on the 2 struts with negative margin, thereby increasing the margin of safety to a positive level. No further damage was noted. • Detailed stress analysis verified a positive margin of safety on all installed tubes. • With the exception of the 2 struts with doublers installed, all STS-38/OV-104 wing strut tubes were determined to be structurally acceptable for unrestricted use. ***This risk factor was resolved for STS-38.***
7	Auxiliary Power Unit (APU) Gas Generator Valve Module (GGVM) issue. HR No. ORBI-031 {AR} ORBI-184 {AR} ***OV-104 APUs performed nominally during STS-38.***	During the STS-31 launch attempt on April 10, 1990, APU #1, S/N 305 with GGVM S/N 3002, exhibited speed control problems shortly after startup. Failure analysis concluded that the speed control problem was caused by missing tungsten carbide material from the GGVM Pulse Control Valve (PCV) seat, allowing a fuel leak path. At the time, this failure mode was not considered generic (see STS-41 MSE, L–2 Edition, October 4, 1990, Section 5, Orbiter 1, for further details of the STS-31 failure). Recent testing and inspection of other GGVMs for this phenomenon found additional anomalies related to the valves. The resulting failure analyses concluded that GGVM PCV and Shutoff Valve (SOV) failures were generic in nature. Original conclusions drawn from the failure analysis indicated that PCV failures were related to high operating cycles and SOV failures were attributed to leaching of the valve seat. However, the most recent PCV failure on GGVM S/N 4003 illustrated that PCV failure may not be cycle related as previously believed. Details of the

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7 (Continued)		GGVM investigation are found in the STS-41 MSE, Flight Readiness Review (FRR) Edition, September 24, 1990, Section 4, Orbiter 3. The Orbiter Project Office (OPO) specified interim GGVM life limits to protect against the inability to shut down an APU. Discussion at the STS-35 Delta FRR led to a reevaluation of the life limit requirements. GGVM life limits are now bounded by 25-1/2 months of SOV wet time and 72,000 PCV cycles. However, because of the recent PCV failure on GGVM S/N 4003, life limit based on PCV cycles may not be credible. In addition to the life limit criteria, the OPO also specified a requirement for a liquid leak test on all APU/GGVMs prior to launch. This test verifies the integrity of the PCV seat. Passing this test provides some confidence that no PCV seat material is missing. Leak tests of STS-38/OV-104 APU/GGVMs were performed at the pad with no problems noted. APU #3, S/N 311, was hot fired at the pad. APU #1 and APU #2 were hot fired during the first STS-38 pad process. Rationale for STS-38 flight was: • STS-38/OV-104 APU/GGVMs limited life criteria were not exceeded by the launch date. • Liquid leak tests were successfully performed and provided additional confidence that no PCV seat material was missing. • LCC would not allow a launch with an APU in high speed. • APUs #1, #2, and #3 were successfully hot fired prior to launch. *This risk factor was acceptable for STS-38.*

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ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8	Main Propulsion System (MPS) joint weld issue. HR No. ORBI-306 {AR} ***There were no anomalies attributed to welded MPS joints reported on STS-38/OV-104.***	H_2 leaks on OV-102 and OV-104 raised concern for the potential of undetected weld defects in MPS joints that could lead to joint failure and potential leak paths. The majority of LH_2 MPS welds are inside vacuum jacketed lines and are not presently considered suspect. A Johnson Space Center (JSC)/RI investigation focused on non-vacuum jacketed lines, 2" diameter or less, that were manufactured by RI. X-ray records of 58 LH_2 and LO_2 welded MPS lines were reexamined. No out-of-specification conditions were found in 40 of the 58 lines. The remaining 18 lines were identified for further engineering evaluation. Fourteen lines were determined to be borderline interpretation cases and were accepted by engineering disposition. Detailed engineering evaluation was required on the x-rays of 4 others. Three of the 4 joints in question are on OV-102 [a GH_2 line, a Gaseous Oxygen (GOX)/LO_2 line, and a LO_2 bleed line]; one line (a GH_2 line) is a spare. Conservative review of the x-ray images of the 4 joints in question led to the following interpretations: • The OV-102 GH_2 line showed a lack of weld penetration, approximately 0.030" long. • Lack of weld penetration, approximately 0.250" long, was found in the OV-102 GOX/LO_2 line. • A crack-like indication, 0.055" in length, was found in the OV-102 LO_2 bleed line. • Lack of weld penetration in 60% of the circumference was identified in the spare GH_2 line.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		Given the above interpretations, fracture analysis was performed to determine the magnitude of the potential for flaw growth. The analysis assumed nominal flaw depth of 90% of 0.035" wall thickness. The analysis determined that flaw growth is not predicted for line exposures to proof-pressure tests and subsequent mission environments. The JSC/RI investigation found no indication of generic concerns with welded MPS lines. Lines on STS-38/OV-104 were considered cleared for flight due to similarity in welding process and inspection criteria. Rationale for STS-38 flight was: • There was no indication of a generic problem. • Flaw growth was not predicted based on fracture analysis. ***This risk factor was resolved for STS-38.***
9	Cracks in APU dynatube fittings. HR No. ORBI-103 {AR} ***There were no APU dynatube fitting leaks reported during OV-104 postflight inspection.***	During acceptance testing of APU S/N 307, helium leak checks were performed. A leak, 2 x 10^{-6} standard cubic centimeters per second (sccs), was found at the GGVM bypass dynatube fitting. Acceptance test allowable leakage is 1 x 10^{-6} sccs. Dye penetrant inspection of the dynatube fitting identified a 90° circumferential crack. Inspection of the reference and inlet dynatubes found that 2 of 4 fittings also had cracks; 1 circumferential and several radial. All APUs at Sundstrand were inspected based on the S/N 307 findings. A dynatube fitting on APU S/N 310 was found with a 300° circumferential crack. No cracks were found on 2 other APUs, S/N 312 and S/N 207.

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ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9 (Continued)		GGVM dynatube fittings are made from 17-7 PH which is susceptible to stress corrosion. The presence of a contaminant is required to promote stress corrosion cracking. The stress corrosion problem was identified early in the program at the GGVM supplier. Dynatubes on improved APU GGVMs will be made from 17-4 PH. APUs are helium leak checked prior to and after acceptance testing, as well as after initial installation in an Orbiter. Critical flaws are considered screenable by these helium leak checks. STS-38/OV-104 APUs did not experience similar leaks during acceptance and post-installation testing. GGVM dynatube fittings are lockwired on installation. Rationale for STS-38 flight was: • The APUs are helium leak checked before and after the Acceptance Test Procedure (ATP). • The APUs are leak checked after initial installation in the vehicle. • No leakage was detected on OV-104 APUs, and there was no leakage history. • The design is failure tolerant; i.e., the dynatube fittings have a sealing surface outboard of the stress corrosion cracking area, and the fittings are lockwired in place. ***This risk factor was resolved for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
10	OV-102 20-psi helium regulator leak. HR No. ORBI-306 {AR} ***There were no anomalies attributed to the 20-psi helium regulators on STS-38/ OV-104.***	During leak check activities following the first STS-35 launch attempt, one of two 20-psi helium regulators was found to have a leak of 1 x 10^{-4} sccs. The regulator was removed after this finding and returned to the vendor for evaluation. This regulator 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 scim at 285 psi. A 2-scim leak was observed at the maximum system operating pressure of 30 psi; 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. The diaphragm is constructed of 2 plys of 347 stainless steel, approximately 2 mils thick each. The sensor diaphragm is exposed to GH_2 sense line pressure. The diaphragm exerts forces on the Belleville springs that operate the regulator pilot poppet and regulate helium pressure. This was the first diaphragm failure in the program history. Materials and processing analysis at RI indicated that the diaphragm failed by fatigue cracking resulting from stress concentrations at wrinkles. Possible causes of the wrinkles included 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. Because one side of the diaphragm is exposed to GH_2, leaks through the diaphragm could lead to 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 5000 scim. This potential leak is detectable by the aft compartment Hazardous Gas Detection System (HGDS) and would result in a scrub prior to launch.

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<u>ORBITER</u>		
10 (Continued)		The regulator is used post-Main Engine Cutoff (MECO) to regulate the helium purge of the H_2 lines in the MPS. It is also used during reentry and landing to maintain positive pressure in the MPS lines and eliminates the potential for drawing in contamination. A helium isolation valve is upstream of the 20-psi regulator. The isolation valve could be closed if the regulator failed open. Rationale for STS-38 flight was: • This was the first diaphragm failure in the program history. • OV-104 20-psi regulators were verified by helium signature leak checks and functional tests. • If GH_2 leaks through the diaphragm after launch, maximum leak rates result in below-allowable aft compartment H_2 concentrations and flammability limits. *This risk factor was acceptable for STS-38.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11	Fuel Cell (FC) separator plate plating defects. HR No. ORBI-282A {C} ***There were no FC anomalies reported on STS-38/OV-104.***	During recent refurbishment of FC S/N 109, plating blisters were found on 6 separator plates. These blisters were similar to those observed on separator plates from S/N 104 and S/N 115 in September 1989, which led to an indepth investigation. That investigation determined that all suspect separator plates were from the same manufacturing lot. The blistered separator plates found in S/N 109 were not from this original suspect lot and, therefore, gave rise to the potential for a generic problem with all FC separator plates. Magnesium corrosion was recently found to be attributed to the presence of microcracks in the nickel layer. Corrosion through one H_2/O_2 plate (not part of the S/N 109 problem) was found during a visual inspection in September 1990. Pressure tests resulted in no leakage, indicating that the corrosion product did not degrade plate integrity. This corrosion site developed over a 31-month exposure period to potassium hydroxide. The previous investigation found that the blister failure mechanism was separation of the gold and nickel layers from the magnesium base material. At that time, no corrosion was observed through to the magnesium base. Potassium hydroxide, used as an electrolyte with water, was determined to passivate the bare magnesium so that corrosion could not occur. If material/magnesium impurities are present at the blister site, corrosion pits could potentially develop. The concern associated with blistering is based on the potential for explosive mixing of H_2 and O_2 through the separator plates, resulting in loss of vehicle and crew. Indication of H_2 and O_2 mixing requires immediate FC shutdown and safing. Leakage in the H_2-to-O_2 separator plate can be detected by the FC performance monitor. If leakage is detected, procedures call for the crew to shut down the indicated FC. Loss of 1 FC results in a minimum-duration flight; loss of a second FC requires emergency powerdown and landing at the next primary landing site. Mixing of H_2 and coolant is more benign, resulting in slow degradation in FC performance. Turnaround testing also checks for potential leakage through the use of Nitrogen (N_2) diagnostics and coolant leak checks to verify FC integrity. STS-38/OV-104 FCs passed all of these tests.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)		FCs used for the qualification test program operated for 2000 hours (hr) with no problems. A historical review found that FCs operated successfully with blistered plates for up to 3500 hr. FCs S/N 104 and S/N 115, where blisters were initially found, had over 1000 hr of operating time. FC S/N 109 had 411 hr of operation prior to the refurbishment effort that identified the 6 blistered plates. OV-104 FCs had less than 850 hr of total operation prior to STS-38. Investigation is ongoing to determine a solution to the blister failure mode. Turnaround testing and inflight FC performance monitoring will continue to be the mitigating control against catastrophic H_2-to-O_2 separator plate leakage until a solution is found. Rationale for STS-38 flight was: • No corrosion was detected during failure analysis of all blistered separator plates. Corrosion was found in microcracks through the nickel layer; however, plate integrity was maintained. • Leakage is detectable by existing instrumentation monitored in the Mission Control Center (MCC); a detected problem in flight results in a ground call for the crew to shut down and safe the problem FC. • Turnaround testing was completed with no identified problems. ***This risk factor was acceptable for STS-38.***

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ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
12	OV-104 GH_2 FCV power "on" violation. HR No. INTG-151 {AR} ***The GH_2 FCVs performed nominally on STS-38/OV-104.***	Data review at KSC showed that, during post-tanking checkout, the STS-38/OV-104 GH_2 FCVs were energized longer than the OMRSD limit (energized approximately 3.25 hr on 2 separate occasions). This occurred during the 2 tanking tests at the pad prior to rollback of STS-38/OV-104 due to the LH_2 leak problem. The OMRSD requirements for leaving the valve "power" on are: • No valve solenoid can experience more than 20 min of cumulative on-time in any 3-hr period. • There cannot be simultaneous energizing for more than 10 min of 2 or more valves in any 3-hr period. The concern is for the health of the valve solenoid and the nitrite rubber O-ring. Continuous FCV O-ring operating temperatures range from −65 to 275°F. RI analysis indicated that the O-ring temperature would reach 259°F after 1 hr of operation, 280°F after 1.5 hr, and stabilize at 282°F after 2 hr. A similar situation occurred on STS-35/OV-102. The OV-102 LH_2 FCVs successfully passed resistance tests and leak checks with no anomalies; however, the FCV valve solenoids and O-rings were replaced because of the concern regarding the time period that the O-rings were subjected to temperatures above the operating service temperature. The OV-102 removed hardware was analyzed to determine if the unplanned energizing adversely affected the valve components. Results of the analysis indicated no adverse effects on the solenoids; however, O-ring resiliency was reduced. STS-38/OV-104 LH_2 FCVs also passed resistance tests and leak checks with no anomalies. However, because of the OV-102 O-ring degradation findings, STS-38/OV-104 LH_2 FCV solenoids and O-rings were replaced prior to launch. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
13	Freon leaks in Flash Evaporator Subsystem (FES) on STS-38/OV-104. HR No. ORBI-275 {C} ORBI-276B {C} ORBI-300 {C} ORBI-321A {C} ***There were no further FES freon leaks reported on STS-38/OV-104 after the prelaunch repair activities.***	Freon Coolant Loops (FCLs) are monitored continuously during launch preparation for potential freon leaks. During STS-38 operations, FCL #1 freon accumulator quantity was found to have decreased approximately 6% in a period of 6 weeks while in the vertical position. OV-104 troubleshooting with a halogen leak detector found no leak in either the midbody or aft. Continued troubleshooting detected a leak in the area of the FES topping ducts; >200 ppm in the left topping duct and 80-90 ppm in the right duct. Because of the concern for a large freon leak leading to frozen freon in the interchanger and the potential loss of both freon loops and water loops, the decision was made to remove and replace the FES. Upon FCL #1 deservicing, Gaseous Nitrogen (GN_2) and helium leak checks were performed to determine if there were any other leaks in the system. Additional leaks were isolated at the freon pump package, a radiator flange, 1 midbody cold plate, and 3 aft cold plates. All leaks were repaired. Retest of the installed FES was completed with no anomalies reported. Materials and processing analysis of the removed FES found small amounts of contamination. Previous analysis indicated that corrosion of the kind found could not be produced in a freon environment; however, that conclusion is now in question. Analysis of the origin of the contamination is underway. Rationale for STS-38 flight was: • The FES was replaced and passed all leak tests prior to launch. • Other leaks found in FCL #1 were repaired. ***This risk factor was resolved for STS-38.***

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ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
14	STS-38/OV-104 Reaction Control System (RCS) thruster oxidizer leaks. HR No. ORBI-119 {AR} ORBI-203 {C} ***OV-104 RCS thruster R1U exhibited low Chamber Pressure (P_c) during STS-38 operations. R1U performance, when required, was nominal. The crew manually transitioned R1U to the lowest priority; however, it was not deselected. See Section 7, Orbiter 7 for further details.***	During the post-rollback inspection of STS-38/OV-104, 11 RCS thrusters, 7 aft and 4 forward, were found to exhibit oxidizer leakage. The condition worsened after the manifold isolation valves were closed to protect the RCS against potential GN_2 bubble intrusion following a left Orbital Maneuvering System (OMS) GN_2 injector problem. Oxidizer leakage was attributed to manifold temperature and pressure changes caused by minor purge temperature variations. Ambient temperature fluctuations caused manifold oxidizer pressure to decrease below the OMRSD requirement of 40 pounds per square inch absolute (psia) to 30 psia. This decrease in manifold pressure allowed oxidizer leakage through the thruster valve seats. All thrusters stopped leaking after ambient temperature was restored, and thruster heaters were powered on except R4D and L3D. Monitoring of R4D and L3D determined that the oxidizer leak was not subsiding and, therefore, both were replaced. No further oxidizer leaks were experienced after thruster replacement. Rationale for STS-38 flight was: • All 11 RCS thruster leaks were stopped. Thrusters R4D and L3D were replaced. ***This risk factor was resolved for STS-38.***

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ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
15	Thermal Protection System (TPS) damage as a result of thruster R4D oxidizer leak. HR No. ORBI-047A {AR} ***There were no further TPS problems reported as a result of thruster oxidizer leaks.***	Due to the substantial tile damage incurred on OV-103 as a result of the thruster L1A oxidizer leak, and due to the oxidizer leaks found at 11 OV-104 thrusters, all surrounding TPS tiles were examined. A tile adjacent to R4D was removed to assess the degree of contamination. The Strain Isolation Pad (SIP) was found discolored (yellow); however, this contamination was considered minor, and a new tile was installed over the SIP. Proof tests of 23 tiles adjacent to Right-Hand (RH) and LH leaking thrusters were performed at 1.25 times the design limit stress. All tested tiles passed. No other problems with the TPS resulting from thruster oxidizer leaks were found. Rationale for STS-38 flight was: • Extensive examination and test of TPS tiles near leaking thrusters found only 1 contaminated tile. This tile was removed and replaced. ***This risk factor was resolved for STS-38.***
16	OV-104 Engine Interface Unit (EIU) Power-On Reset (POR) anomaly. HR No. INTG-165 {C} ORBI-066 {AR} ***There were no further PORs reported after the replacement of EIU S/N 12 on STS-38/OV-104.***	During STS-38/OV-104 EIU testing on October 13, 1990, S/N 12 experienced a single POR. Several single PORs were experienced with other EIUs on other Orbiters. The determination was made to replace S/N 12. This replacement was completed on October 21, 1990. The concern was that a simultaneous POR-A and POR-B in the last 30 sec prior to MECO will result in the General Purpose Computer (GPC) closing prevalves on running engines, resulting in a catastrophic shutdown. (This is the worst-case failure scenario.) Failure of GPC command of the engine requires manual shutdown. Flight rules/crew procedures exist for this condition. System management alert and Main Engine (ME) status (amber) light on the panel (F-7) will alert the crew. Crew reaction is required to manually shut down MEs prior to prevalve closure. The crew would require a ground call to confirm POR indication. The indication would reset after EIU recovery. This condition was briefed to the STS-38 crew prior to flight.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
16 (Continued)		The subject "Power-On Reset" was characteristic of previous occurrences (8 units, 10 vehicle flows since January 1983). POR was transient and self clearing, and troubleshooting did not reproduce the problem. Rationale for STS-38 flight was: • Each occurrence was a single POR (POR-A or POR-B). • No single POR will fail more than one data command/data channel. • POR has not occurred in flight. • A single POR prior to T–0 will result in a launch hold or launch abort. • Simultaneous reset of both channels has never been experienced and requires 2 failures. • The last 30 sec prior to MECO is a very short time window in which the crew cannot react, and flight rules and procedures exist for this condition. Probability for simultaneous failure of both POR-A and POR-B during this time is estimated to be 2.5×10^{-11}. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
17	Damage resulting from horizontal access beam free fall in aft compartment. HR No. SAR-A70-0702 P1 {C} SAR-A70-0702 P4 {C} ***There were no inflight anomalies reported that were attributed to the access beam incident on STS-38/OV-104.***	While raising STS-38/OV-104 to the vertical position for mate with the ET, 3 distinct sounds were heard emanating from the vehicle. Mating operations continued through hard mate with the ET. Upon inspection of the aft compartment, a 9-foot beam, G070-502677-021, part of the aft compartment horizontal access kit, A70-0702, was found. While it was believed that this beam had been removed during aft compartment closeout, it was inadvertently left in place. An investigation team was formed to assess the extent of damage to aft compartment components. The following is a preliminary list of team findings and resulting action taken: • Purge, vent, and drain system duct, Part Number (P/N) V070-385221-004, was damaged and was replaced. • MPS LO_2 manifold relief line, P/N V070-415470-003, was found with a deep ding and was replaced. • Engine #3 MPS LO_2 feedline was found with a 6" long ding in the fiberglass cover. Evaluation determined that the feedline was undamaged. The fiberglass cover was repaired. • No major structural damage was found; however, there were 14 scrapes and dents evident. Minor repairs were completed. No damage to the boron fiber was seen.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
17 (Continued)		• Electrical Power Distribution and Control (EPD&C) subsystem inspection found 2 minor problems that were repaired, including the replacement of several damaged clamps. • No problems were reported with the APUs and hydraulic subsystems or with the SSMEs. Shuttle interface testing, S0008, was completed with no additional problems relating to this incident identified. Rationale for STS-38 flight was: • An investigation of structural/mechanical damage was performed, and all identified damage was repaired. • An investigation of the EPD&C was performed. The damaged cable was repaired and functionally tested. No additional damage was identified. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
18	STS-38/OV-104 FES water supply valve retest criteria. HR No. ORBI-117 {C} ORBI-276B {C} ***There were no FES water leaks reported after the replacement of the water supply valve on STS-38/OV-104.***	Tests of FES "A" system after changeout revealed 3 internal dynatube-fitting leaks: at the topping "A" and "B" isolation valves and at the hi-load "A" spray valve. Retests of FES "A" after several days of wet time indicated that these leaks had stopped; another leak was identified in the "A" system water supply topping spray valve dynatube fittings. The potential consequence of a leak is the possibility of water freezing in the FES core during ascent, leading to FES functional failure. It was not believed that a valve dynatube leak of this nature would result in the failure of the fitting resulting in a leak of the magnitude required to freeze the FES core. The decision was made, however, to change out this valve. The issue with this valve was with the retest criteria. OMRSD requirements dictate that the FES water supply valve and associated lines be overpressure tested with helium to 300-310 pounds per square inch gage (psig) and leak checked at 110-120 psig. These overpressure tests and leak checks would require intrusion into the payload bay. Water pressure and leak checks that could be performed without payload bay intrusion would only reach 25-27 psig and 15.5-17 psig, respectively. Operating pressure for the water system is 16 psig; however, pressures up to 20 psig can be experienced on ascent. An OMRSD exception was approved to use substitute helium overpressure and leak checks. The substituted tests comprised a helium leak test at 15.5-17.0 psig and a water leak test at 15.5-17.0 psig. The rationale for this substitution was based on the fact that, in the vertical position, the FES water system has a resulting pressure head of approximately 45 psig. This is twice the operating pressure of approximately 20 psig experienced on ascent. These tests are considered adequate to identify gross leakage. Additionally, experience demonstrates that water leaks are common at the valve dynatube fittings upon wetting after an extended dry time. Once wetted, the seals normally swell to perform the required sealing function. This was demonstrated with leaking of 3 valves after initial FES wetting.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
18 (Continued)		Rationale for STS-38 flight was: • Dynatube fittings are highly reliable. • A small leak experienced on ascent would not affect FES performance. • The tests prescribed by the OMRSD exception are considered adequate to identify gross leaks. ***This risk factor was acceptable for STS-38.***
19	Inertial Measurement Unit (IMU) #2 "power conditioner fail" Built-In Test Equipment (BITE) indication. HR No. ORBI-051 {C} ***There were no IMU anomalies reported on STS-38.***	Postlanding on STS-41, BITE indicated an IMU #2 "power conditioner fail" 1 sec prior to powerdown. The IMU data appeared valid for several seconds after the BITE indication until loss of data due to the powerdown. No other BITE indications were issued. The cause was determined to be related to the power-off procedure utilized. The normal power-off sequence is to take the IMUs from operate to standby and then to off. The sequence used on STS-41 was to go from operate to off without stopping/pausing in the standby mode. The IMU BITE would then be flagged by caution and warning. The GPC periodically polls the IMUs for 14 words of serial data. The periodicity of this poll is determined by the IMU subsystem operating program submode. As the IMU is "losing power", the BITE circuits within the IMU witness the power loss and set appropriate BITE indications, such as "power conditioner fail". At the higher data rates between the IMU and GPC, there is a greater probability that the BITE fail indications will be seen as the IMU transmits its data words to the GPC while power is being removed. Variation in data rates and timing of poll cycles are considered to be the reasons why different IMUs and power off sequences show different results and BITE indications. However, when the

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
19 (Continued)		IMU is first taken to standby, the GPC reads this indication from the IMU and removes the "operate" command. During the next polling cycle, the GPC will find the IMU in standby and bypass it. This failure mode is Criticality 1R/3. There is no workaround; reentry would be performed with the 2 remaining IMUs. The failure is detectable by caution and warning. If an IMU is failed by Redundancy Management (RM) when the 3 IMUs are operating, the erroneous IMU is deselected. This was the first failure indication of this type. The same powerdown sequence was repeated with IMU #2 at KSC, and the same problem was seen. The other 2 STS-41 IMUs were put through the same sequence with the same "power conditioner fail" indication. Similar testing was performed on STS-38/OV-104. This testing did not result in any BITE indication. Rationale for STS-38 flight was: • This was the first failure occurrence of this type and was not considered generic. • There is adequate redundancy. This failure mode is Crit 1R3. • A similar failure is detectable by caution and warning. ***This risk factor was acceptable for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
20	GN_2 SOV weld failure on OV-105. HR No. ORBI-071 {C} ORBI-111 {C} ***There were no GN_2 SOV weld anomalies reported on STS-38/OV-104.***	A GN_2 SOV inlet fitting weld failed during installation on OV-105. Investigation into the failure determined that an improper installation technique was used that resulted in application of excessive torque to the fitting. The technician making the installation should have used the "double-wrenching" technique, but instead used only 1 wrench when torquing down the fitting. Further investigation found that there was a lack of proper weld penetration; actual of 0.005" versus the 0.020" minimum required. Due to the lack of weld penetration, and due to the fact that all GN_2 SOVs in the fleet used the same subassembly buildup process, there was the potential for a generic problem. Review of build records determined that all GN_2 SOVs in the fleet, other than those installed on OV-105, were fabricated 15 years ago; the same time as the qualification unit. The qualification unit, the first of the original lot, successfully passed all tests. A recent review of the qualification unit x-rays found adequate weld penetration in all welded valve joints. It was discovered during this review that the welding equipment, and thus the welding procedure, was changed after fabrication of the OV-104 GN_2 SOVs; the last of the original lot. Review of the qualification test results, and the change in welding procedures prior to the fabrication of the OV-105 valves, cleared the GN_2 SOVs installed in the fleet for continued use. Rationale for STS-38 flight was: • This was not a generic problem. • The original GN_2 SOV lot, including those installed on STS-38/OV-104, were cleared for flight. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
21	STS-41/OV-103 RH elevon cove components found charred during postflight inspection. HR No. ORBI-003 {C} ***There was no report of similar charring on STS-38/OV-104 elevon cove components.***	During postflight borescope inspection of the STS-41/OV-103 elevons, charring was found on RH elevon components. No charring was found on the LH elevons. The following is a summary of the inspection findings: • RH inboard elevon actuator hinge seals were charred. • RH outboard elevon actuator hinge seals were charred. • RH inboard elevon block seal at YW212 was badly charred. These findings were reported to be the second worse case experienced in the program, although some charring was seen on most flights. The worst case was the charring witnessed on the STS-28/OV-102 elevon cove seals which resulted from an early asymmetric boundary layer transition during reentry. Experience indicated that elevon cove charring is a function of elevon positioning. Review of flight data had not indicated abnormal ascent or reentry elevon positioning. There was no indication of hot-gas flow through the seals and no indication of internal or structural damage. Current plans are to rework the OV-103 elevon cove seals in preparation for STS-39. ***This risk factor was acceptable for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	Engine #0213 spark igniter potential failure. HR No. ME-B2S {AR} ME-C3S {AR} ***There were no spark igniter anomalies reported on STS-38 SSMEs.***	Engine #0213 hybrid igniter failed during mainstage on the B1 test stand at Stennis Space Center (SSC). This failure mode did not preclude satisfactory performance of the redundant igniter circuit. A sneak circuit, caused by the failure, excited the other igniter on the same channel. There are 2 igniters on the same circuit, one for the Oxidizer Preburner (OPB) and one for the Fuel Preburner (FPB). This engine had 15 starts and 6035 sec of operating time. Teardown for trouble analysis revealed a broken hybrid lead. Microscopic examination of the lead revealed a high-angle fatigue fracture. Stress analysis showed susceptibility to High-Cycle Fatigue (HCF) due to igniter base excitation. The fleet leader had 48 starts and 22,849 sec of operating time without a fracture. On this flow, there were 11 hybrids out of 18 igniters [6 per engine: 2 OPB, 2 FPB, and 2 Main Combustion Chamber (MCC)]. A related igniter failure occurred during a recent STS-35 Flight Readiness Test. This failure was attributed to moisture on the igniter tip. Rationale for STS-38 flight was: • There is a benign environment during the initial start phase. During this phase, there is low vibration, and engine ignition occurs by 0.6 sec. • There are redundant igniters in all chambers. • The probability of a single igniter failure is very low. • Dual igniter failure will result in a pad abort. ***This risk factor was acceptable for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
2	Low-Pressure Fuel (LPF) duct tripod cracks. HR No. ME-D3 (All Phases) {AR} ***There were no SSME anomalies reported on STS-38.***	Inspection of LPF duct tripod S/N 4881383 found cracks in the upstream tripod leg radii. This inspection was performed primarily for observation of cracks in the radii of downstream legs where similar cracks were experienced. An improved borescope was used for the first time to perform the downstream leg inspection. Previous cracks were experienced in tripods; however, this was the first time cracks were witnessed in the upstream legs. No cracks were found on the downstream leg. The tripod is a structural element internally supporting the LPF duct flex line at joint C. Tripod S/N 4881383 was only used in development testing and will not see future flight engine exposure. This unit experienced 69 starts with a total run time of 18,527 sec. S/N 4881383 experienced 12,414 sec of operation on engines run at 104% or greater. In comparison to other test units, no cracks were found on 2 tripod units, S/N 4887572 and S/N 4881753, with 25,286-sec and 36,114-sec total run time, respectively, encompassing operations at or above 104% of 5,296 sec and 8,811 sec, respectively. A third unit, S/N 4917969, experienced 89 starts and 31,705-sec run time with 12,517 sec of operation at or above 104%. One leg of S/N 4917969 broke free during a test run on engine #2206 in June 1989. This failure and subsequent investigation led to the requirement for borescope inspection of downstream legs. Recent hot-fire tests on engine #0213 indicated that the tripod leg failures were due to resonance with High-Pressure Fuel Turbopump (HPFTP) synchronous frequencies. LPF ducts are instrumented with strain gages, accelerometers, and frequency pressure transducers for this hot-fire test. The test procedure requires ramping the engine power level from 65% to 104%, increasing 1% every 5 sec. For power functions of 0.5 to 1.0, there is a random frequency response. For power functions greater than 2.0, there is a sine response. Two frequencies are dominant: Low-Pressure Fuel Turbopump (LPFTP) at 250 Hertz (Hz) at the LPF duct resonance, resulting in low-level response at all flex joints;

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
2 (Continued)		HPFTP at 600 Hz that approached the duct resonance of 615 Hz. Here, there is a high response potential at flex joint "C" at higher speeds The failure mechanism is axial pressure oscillation with high amount of time at greater than 104% power level. Given that the experience base for cracked tripod legs at high operating time was limited, LPF duct tripods on OV-104 SSMEs were considered safe for flight based on low exposure to 104% or greater operation. An effort is underway to redefine the life limit and inspection criteria for future LPF duct use. Rationale for flight of LPF ducts on STS-38/OV-104 was: • Duct failures to date had greater then 12,500 sec of hot-fire operation at 109% power level; maximum OV-104 duct operation at greater than 104% was 313 sec. • All OV-104 HPFTPs had synchronous frequencies less than 590 Hz at 104%; this was below the LPF resonance frequency. ***This risk factor was acceptable for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
3	POGO precharge pressure software anomaly. HR No. ME-C1C Rev. A {AR} *There was a post-MECO Fault Identification (FID) on engine #2019 presented on STS-38. This FID indicated that the POGO precharge pressure transducer failed qualification limits on the low side. POGO precharge pressure qualification limits were implemented in AR01; however, because this FID was presented after nominal engine performance and shutdown, this FID was not considered a problem. For STS-39 and subsequent flights, POGO precharge pressure qualification limits will be inhibited after MECO +16 sec to preclude recurrence of this problem.*	During verification testing of the new SSME controller software, AR01, it was determined that the logic did not detect a helium POGO precharge system failure after previously disqualifying 1 sensor. This error was the result of an incorrect requirement definition in the original Logic Change Notice (LCN) for AR01. The intent was to have the AR01 logic issue a Major Component Failure (MCF) and shut down the SSMEs prior to SRB ignition after disqualifying 1 of 2 POGO precharge pressure sensors and a second sensor either fails or there is a failure of the helium precharge system. The logic will not issue an MCF but will post a FID code for helium precharge system failure. Indication of POGO precharge pressure prior to launch gives a positive indication that the helium precharge valve had operated. While it is not critical to precharge the POGO system prior to launch, operation of the POGO system is critical at MECO to pressurize the oxidizer system for a zero-g shutdown, thus preventing possible cavitation of the oxidizer pumps. Oxidizer pump cavitation is a Crit 1 failure mode. Rationale for STS-38 flight was: • This problem was fixed with the use of a patch in AR01. *This risk factor was resolved for STS-38.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
4	Potential for defects in critical zone of LO_2 main injector posts on engines #2019 and #2022. HR No. ME-B4 (All Phases) {AR} *SSME performance on STS-38 was good. Specific impulse loss due to Liquid Oxygen (LOX) post plugging on engine #2022 was demonstrated to be real on STS-38.*	The potential for defects existed in the critical zone of engines #2019 and #2022 LO_2 main injectors in rows 12 and 13. Projected LO_2 post damage fraction exceeded 0.25; calculated damage fraction for post 8, row 13 of engine #2022 was 0.46 after a worst-case Return-to-Launch Site (RTLS) abort prior to STS-32. Post 8 was subsequently plugged. The injectors on engines #2019 and #2022 were fabricated prior to the implementation of RL00860, injector post inspection requirements for surface finish and defects. During a pre-STS-32 inspection of engine #2022 post, using RL00860 requirements and a flex borescope, an offset of 0.0045" was found at the inertia weld. Light recast was also found in the 0.183" and 0.156" diameter areas of the post. At that time, recasts were determined to be acceptable. An earlier inspection of engine #2019 with a flex borescope also identified recasts, gouges, and offsets in this critical area. Reinspection of engine #2022 to RL00860 requirements with a rigid borescope identified significant gouges and scratches. An additional 9 LO_2 posts were plugged. Because of the findings on engine #2022, a decision was made to reinspect engine #2019. This inspection resulted in the determination that 2 posts had to be plugged because the accumulated damage fraction exceeded 0.25. Three additional engines in the fleet that had the same fabrication sequence as engines #2019 and #2022 are engines #2011, #2030, and #2107. Critical posts in these engine main injectors were borescoped in accordance with RL00860 and found to have steps, offsets, recasts, overburn, gouges, and scratches. Up to 4 posts in these engines were plugged because of accumulated damage greater than 0.25. Up to 25 critical posts may be plugged before main injector retirement. Rationale for STS-38 flight was: • All critical posts on engines #2019 and #2022 were inspected to an acceptable level of detail. *This risk factor was resolved for STS-38.*

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
5	HPFTP turbine bearing support crack. HR No. ME-D1 (All Phases) {AR} ***There were no SSME anomalies reported on STS-38.***	Circumferential cracks were found in a reworked area of the turbine bearing support from HPFTP S/N 4404R2, a development pump. Investigation determined that HPFTP S/N 4109R1, installed on STS-38/OV-104 engine #2027, had similar rework performed. Cracks found on S/N 4404R2 run adjacent to the electron beam weld area for the full circumference. These cracks were only at the bottom of a sharp step in a machined radius. The majority of cracking was located in a single 80° arc. This support accumulated 49 starts and 13,301 sec of hot-fire time. The turbine bearing support design was modified in 1982 to incorporate an outer lip to protect the fishmouth seal static seal. Existing units were reworked by machining the lip from a ring that was electronic beam welded to the support. New units were machined from existing forging stock. Twenty-nine supports were reworked and tested. There were 6 reworked supports with more hot-fire time and 4 reworked supports with more starts than the S/N 4404R2 cracked support. Nine reworked supports were visually inspected with 10-power magnification; no cracks or sharp radii similar to the cracked unit were found. Analysis of the S/N 4404R2 support cracks under 70-power magnification determined that the crack mode was most likely due to stress rupture or Low-Cycle Fatigue (LCF). Cracks were found to be predominant in the areas with the sharpest radius. The radius was measured, using molds, to be approximately 0.005". Review of the S/N 4404R2 rework fabrication record discovered that the sharp radius was the result of a 0.080" deep x 0.040" wide, 360° tool gouge. This condition was documented on Material Report #218843. The Material Review Board (MRB) dispositioned that only the sharp edges be removed. Nothing was done about the gouge. Review of the S/N 4109R1 rework record found no report of a similar condition.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
5 (Continued)		Rationale for STS-38 flight with S/N 4109R1 was: • There is a successful test history of reworked supports. • The sharp-step condition in the radius was not present. • Deflection of the cracked area would be identified by increased blade platform and shank rubbing, evident during testing. ***This risk factor was acceptable for STS-38.***
6	SSC test stand Frantz screen failure. HR No. ME-C1 (All Phases) {AR} ***There were no SSME anomalies reported on STS-38.***	Contamination was discovered in a development test engine during post-test inspection at SSC. Investigation determined that the contamination was material from the Frantz screen used in the B-1 engine test stand. A large portion of the screen was found to be missing upon removal. Only 34% of the missing screen was recovered. No determination was made as to when the actual screen failure occurred. The concern was that 2 High-Pressure Oxidizer Turbopumps (HPOTPs) on STS-38/OV-104, S/N 2323 and S/N 4107, were tested on the B-1 test stand prior to flight-engine installation. Prior to identification of the contamination, the Frantz screen was last inspected in January 1990: that was prior to the S/N 2323 and S/N 4107 test runs. Because the potential existed that the Frantz screen failed prior to these test runs, contamination could be present in the STS-38/OV-104 HPOTPs. The last test run of an STS-38/OV-104 HPOTP, test run 904-074, occurred 6 test runs prior to the contamination discovery, after test run 904-080. Review of test run data indicated that the major failure occurred 2 runs after the last STS-38/OV-104 HPOTP test; however, loss of a few small screen pieces during earlier tests was possible. This was indicated by a noticeable change in the Frantz screen delta pressure during test run 904-076. No contamination was found during post-test inspection of S/N 2323 or S/N 4107. No

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
6 (Continued)		contamination was reported during receiving inspection of these pumps at KSC. Inspection of S/N 6402, the pump installed for test run 904-075, found no contamination. The Frantz screen is comprised of 316 CRES, 0.004"-0.006" diameter wire. The wire ended where he screen came loose which indicated both overload and HCF fractures. Wear was evident on one side of the wire only, and the wear pattern interval was found to be twice the mesh screen weave. All loose ends pointed in the same rotational direction: counterclockwise, looking down. The failure investigation also determined that the Frantz screen assembly was installed with the wrong gasket at the downstream flange. This error was due to an incorrect part number that was called out in the installation procedure. The screen fit loosely between the fine screen holder and the downstream flange. LO_2 flow rotational forces caused the screen to rotate. Rotation resulted in overload and/or HCF failure as the wire was worn. Rationale for STS-38 flight was: • STS-38/OV-104 HPOTPs were tested prior to the discovery of the screen contamination and prior to any indication of a problem (delta screen pressure). • No contamination was reported during post-test and receiving inspection of the STS-38/OV-104 pumps. • The potential for contamination was small, and the mass of potential screen contamination was low. ***This risk factor was acceptable for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
7	LPFTP second stage rotor crack. HR No. ME-D2 (All Phases) {C} ***There were no SSME anomalies reported on STS-38.***	A crack was found in the outer shroud of the LPFTP second stage rotor on development unit #82106 (UCR A026237). The LPFTP rotor is fabricated from 2 parts, and the outer shroud is brazed to the vane tips. The crack was located in the outer shroud between the vanes. The crack was tight with no displacement of material. There was no noticeable effect on LPFTP performance. The rotor last installed in LPFTP #82106 was the fleet leader for single build at 63 starts for 31,046 sec. The rotor was the fleet leader for time at 35,256 sec and was the fifth fleet leader for starts at 77. Analysis determined that the crack had a single initiation point on the shroud interior diameter near the vane trailing edge. Crack initiation and propagation were determined to be caused by HCF. Rotor materials were verified to be correct, and material hardness was found to be nominal. A finite element model was used to analyze the rotor shroud segment and vane. Potential interference with the third harmonic of the 107 stratovanes was identified at approximately 13700 revolutions per minute (rpm). Only slight shroud participation was projected. Laboratory modal tests of the complete rotor assembly found no excitable diametral modes in the turbine operating range. A speed histogram for the cracked unit and 13 high-time rotors determined that the cracked unit time in speed ranges was comparable to other rotors. However, the cracked unit was the significant fleet leader only in the speed range of 16,000 to 16,200 rpm. The conclusion of this failure analysis was that the rotor crack was due to HCF from extended time exposure over the entire pump operating range. A proposal to limit rotor operating time to 15,000 sec is in work.

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

SSME

7 (Continued)

Rationale for STS-38 flight was:

- There was significant hot-fire experience on the cracked unit relative to OV-104 units:

	LPFTP	Starts	Time, sec
Failure	#82106	77	35,256
Engine #2019	#2022R1	13	4,675
Engine #2022	#2024	8	3,165
Engine #2027	#4005	7	2,325
Configuration End Item Spec. Life		55	27,500

- There were no other occurrences of cracks in the 6 units inspected with greater than 15,000 sec.
- There were no performance effects from the crack as it existed.

This risk factor was acceptable for STS-38.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
8	Engine #2031 HPFTP seal fragments. HR No. ME-D3 (All Phases) {AR} ***There were no SSME anomalies reported on STS-38.***	During postflight disassembly of engine #2031, contaminants were found at the HPOTP to hot-gas manifold interface, joint G3. Contaminant dimensions were approximately 0.4" x 0.4" x 0.3" and was identified as INCO X-750 with gold plate. The most likely source of this material was the HPFTP mount ring static seal. Disassembly of HPFTP #6102R1, the last HPFTP installed on engine #2031, revealed 2 missing sections of the pump-end outboard static seal. These missing sections were 3.3"- and 0.5"-long segments. A total mass of 0.61 grams was calculated to be missing; 0.35 grams were not retrieved. During operation, fragments from the static seal could enter the hot-gas flow. Fragments of the size missing from the HPFTP #6102R1 seal would not affect HPFTP performance and would be of insufficient mass to cause downstream damage. It was postulated, however, that fragments had the potential to migrate to the LOX hot-gas manifold in a zero-g environment. This migration could result in damage to the heat exchanger coil when the engine is next started. The scenario needed to result in this damage requires a "smart", 0.6-gram particle to strike a turbine blade in such a way as to gain sufficient velocity to impact the heat exchanger. A review of the static seal test history found 53 cases of outboard seal cracks, with 5 cases where seal segments were missing. These cases were determined from a data base of 83 dual pilot housing builds with a total of 434 starts and 168,638 sec of operation. Missing pieces ranged from 0.5" on HPFTP #2105 with 7 starts and 3,432 sec of operation, to 6.0" on HPFTP #4204 with 6 starts and 2,323 sec of operation. HPFTP #6102R1 had 6 starts and 3,135 sec of operation. Of the HPFTPs on STS-38/OV-104 engines, only HPFTP #6008 on engine #2019 was disassembled and inspected since its last mission. HPFTP #6008 had 2 starts and 815 sec of operation prior to STS-38. The last disassembly inspection of HPFTP #6008 found no indication of static seal cracking. HPOTP #4406R3 was removed after STS-36. There was no contamination found at that time.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
8 (Continued)		Rationale for STS-38 flight was: • HPFTP #6008 had only 2 starts and 815 sec; lowest HPFTP starts with missing seal segments was 6, lowest HPFTP operating time with missing seal segments was 2,323 sec. • HPFTP #6008 was inspected after the last green run, with no indication of seal cracks. • No contamination was found upon removal of HPOTP #4406R3 after the last flight of engine #2019. ***This risk factor was acceptable for STS-38.***
9	SSME controller software HPOTP intermediate seal purge pressure anomaly. ***There were no SSME anomalies reported on STS-38.***	LCN 5023 was designed to accommodate a problem with the ground GN_2 supply for engine purge. Hot-fire certification testing of LCN 5023 determined that the problem was not totally corrected. During normal countdown, Purge Sequence Number (PSN)-4 is commanded at T−4 min. At this time, the ground-supplied GN_2 used during PSN-3 is switched off, and vehicle helium is used for purge during the remainder of the countdown. If a hold occurs after T−4 min and a return to PSN-3 is required, purge pressure at the HPOTP intermediate seal may be below the 175-psia LCC redline. This violation can result because the ground GN_2 supply is slow to come up to pressure. If the LCC redline is violated in PSN-3, a non-resumable major component failure is issued. This condition is cleared by a commanded "controller reset" that will close the bleed valves. After loss of bleed flow, the SSMEs must be conditioned for 60 min after bleed flow is reestablished. This delay could result in exceeding the available launch window and result in a scrub.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
9 (Continued)		LCN 5023 was intended to sufficiently delay deenergizing the helium purge for 10 sec following rollback from PSN-4 to PSN-3 to allow GN_2 ground pressure to come up. During the recent hot-fire test, a fault identification was issued immediately upon rollback. Normal helium demand generated by rollback caused a drop in supply pressure. This was a different manifestation of the original problem first seen on STS-2. Because this problem was an accepted risk since STS-2, and because of the uncertainty of LCN 5023, it was recommended to fly STS-38 "as is" without LCN 5023 active. Safety concurred with this recommendation. ***This risk factor was acceptable for STS-38.***
10	HPOTP bearing etching/corrosion issue. HR No. ME-C1 (All Phases) {AR} ME-C2 (All Phases) {AR} ***There were no SSME anomalies reported on STS-38.***	HPOTP bearing radial faces are electrochemically etched with a lot serial number during manufacturing. The etched serial number should be removed prior to final assembly in accordance with applicable drawings. It was determined, however, that an inspector misinterpreted the drawing requirements and allowed etched bearings to be installed in HPOTPs. A total of 107 bearings were identified as processed through this inspector. Forty of the 107 bearings were designated as turbine-end bearings; 30 were inspected with 2 found etched and 28 unetched. The remaining 67 were designated as pump-end bearings; 39 were inspected with 21 found etched and 18 unetched. Only 5 suspect bearings were installed on pumps. There was 1 on HPOTP #930921, on engine #2027, ME #3 on STS-38/OV-104. There were 2 each in HPOTPs on engines #2031 and #2026 at the KSC engine shop. There were 2 suspect bearing faces on HPOTP #2521R1 on engine #2031 and 2 other suspect bearings on the HPOTP on engine #2026.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u> 10 (Continued)		Material analysis of an etched bearing was performed to determine the worst-case effects. Dissection found that the average etch depth was 0.0002" with the deepest at 0.0005". Stress analysis indicated that an etch depth of 0.001", less than the critical flaw size, could be tolerated with an adequate margin of safety. Bearing hardness in the etched area was measured to be within specification. No intergranular attack was present; however, traces of chloride contamination were found in the etched area. Chloride contamination is of concern, because it can lead to stress corrosion problems. Stress corrosion induced by chloride contamination led to the elimination of a chilling process previously used in the bearing assembly process. Bearings are now installed using a new drying process. All HPOTP bearings on STS-38/OV-104 engines were installed using this new process. The investigation performed by the SSME Project and Rocketdyne determined that HPOTP #9309R1 on engine #2027, STS-38/OV-104, was acceptable for flight with the suspect bearings installed. This determination was based on an inspection of HPOTP #2522R1, which had a similar etched bearing installation and operating condition as HPOTP #9309R1. No corrosion, pitting, or cracking was found on HPOTP #2522R1. HPOTP #9309R1 will be torn down and inspected following STS-38. Rationale for STS-38 flight was: • Electrochemical etch is much less than critical flaw size. The FOS is greater than 2 for the critical flaw size. • The new drying procedure was used to ensure absence of moisture, which can lead to stress corrosion. • A similar HPOTP was determined not to have corrosion, pitting, or cracking. *This risk factor was resolved for STS-38*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
11	HPOTP first stage turbine disc cracking. HR No. ME-C1 (All Phases) {AR} ***There were no SSME anomalies reported on STS-38.***	Dye penetrant inspection of a high-time, developmental HPOTP first-stage turbine disc S/N 3188616 identified radial cracks in the interstage pilot rib. Sixty-five of 72 pilot rib fillet radii were found with cracks measuring 0.010" to 0.120" long. These cracks were not detected or obvious prior to removal of gold plating. The high-time HPOTP, where these cracks were found, was the fleet leader with 21,908 sec and 52 starts and was removed from the flight program for a long time. Similar cracks were found on a second disc. Seventeen of 72 pilot rib fillets have cracks 0.020" to 0.120" long. This second disc had 15,198 sec of operation and 38 starts. Seven discs were examined, with only 2 found with cracks. 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 crystallographic appearance. The fracture mode showed the effects of hydrogen influence which indicated probable LCF or sustained load crack propagation. Structural analysis indicated a cyclic strain range overwhelmingly dominated by thermal shock at shutdown caused by hydrogen 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 was insufficient to result in cracking without hydrogen embrittlement.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
11 (Continued)		Rationale for STS-38 flight was: • The 2 cracked first-stage turbine discs were the first seen in the program. • HPOTPs on OV-104 had less than 25% of the total cracked disc time, mainstage starts, or starts greater than 80 sec (3589 sec and 11 starts maximum) prior to launch. • There was an extensive, safe HPOTP operational history compared to OV-104 HPOTPs; >42 discs with more starts and operating time, >21 discs with twice the starts and operating time. ***This risk factor was acceptable for STS-38.***
12	Preburner fuel duct failed during proof test. HR No. ME-D3 (All Phases) {AR} ***There were no SSME anomalies on STS-38.***	A preburner fuel duct P/N RS007030 failed during recent proof-pressure testing. The failure occurred at approximately 7400 psi, slightly below the 7860 ± 160 psi proof-pressure requirement. The duct was made from seam-welded tubing, constructed of INCO 903. All welding was done at Rocketdyne. Post-failure inspection of the duct found a significant longitudinal tear parallel to the seam weld, approximately 1" from the weld. A similar failure was experienced on RS007022 and was traced to improper heat treatment. A bench-top hardness tester was used on the failed duct, and it was determined that the duct was soft. Further metallurgical analysis indicated that the duct was not properly heat treated. Visual inspection of the failed duct found no discoloration of the parent metal usually associated with heat treatment. Other results of the failure investigation revealed

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
12 (Continued)		a problem with the portable hardness tester used on the P/N RS007030 duct that failed. A determination was made that the portable hardness tester requires a special calibration technique to be employed when the tester is used on high-strength alloys and nickel-based alloys. Failure to use this special calibration technique results in high hardness readings of up to 20 Rockwell hardness points. An effort is now underway to identify all SSME components where the portable hardness tester was used. There are 115 to 120 INCO 903 constructed ducts in the inventory. To date, 2 other preburner fuel ducts were determined to have come from the same material lot and the same heat treatment lot; 1 of the 2 was on engine #2030 on STS-38/OV-104. A visual inspection of this duct found indications of discoloration associated with heat treatment. All INCO 903 parts on STS-38/OV-104 engines were inspected and tested for proper heat treatment and hardness. This effort was completed with no problems found. Rationale for STS-38 flight was: • All suspect INCO 903 parts were verified to have proper heat treatment and hardness. *This risk factor was resolved for STS-38.*

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Hydraulic system Quick Disconnect (QD) spring anomalies. HR No. A-20-04 Rev. C-DCN3 {C} B-20-09 Rev. C-DCN3 {C} ***The STS-38 SRB hydraulic system performed nominally, and there were no anomalies attributed to the hydraulic system QDs.***	During visual inspection of a 3/4" QD spring sealing surface prior to installation, a crack was observed. QD S/N 1000120 had flown once on STS-34, was disassembled, inspected, reassembled, and passed ATP prior to the discovery of the cracked spring. The spring is an electropolished PH 17-7 CH900, compression-type spring with ends closed and ground. This was the first observed crack in PH 17-7 springs in the history of the SRB program. The crack emanated from a machined notch on the first coil at the trailing edge of the ground end. There was also a notch on the last coil. These notches resulted from the finish grinding operation. Inspection revealed notches on other springs. Clamping marks were also found at other positions on various springs. Approximately 32 of the 54 3/4" springs inspected had indications of notches and clamp marks. Details of the metallurgical analysis are found in STS-41 MSE, L–2 Edition, October 4, 1990, Section 4, SRB 1. Based on finding unacceptable 3/4" QD springs, all eight 3/4" QDs on STS-38 SRBs were removed and replaced. No cracks were found on any removed QD spring; therefore, no contamination resulting from associated flaking was considered present in the STS-38 SRB hydraulic systems. Rationale for STS-38 flight was: • All suspect QD springs were removed from STS-38 SRBs and replaced. There were no notches or cracks in the replacement springs. • No contamination was evident based on not finding cracked springs. • SRB hydraulic system performance would not be degraded if this type of contamination was present. ***This risk factor was resolved for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
2	SRB hydraulic pump failure during acceptance testing. HR No. A-20-04 Rev. C-DCN3 {C} B-20-09 Rev. C-DCN3 {C} B-20-21 Rev. B-DCN4 {C} ***There were no SRB hydraulic pump failures reported on STS-38.***	Hydraulic pump S/N 192984 failed proof-pressure testing during acceptance testing at Abex Corporation, the vendor. S/N 192984 was at Abex for routine reacceptance testing following post-STS-34 refurbishment. The investigation into this failure determined that the proof-pressure test was incorrectly set up due to a problem during the previous test operations with the test equipment. This was believed to be an isolated case. The previously tested equipment was not part of the SRB flight program. Further research by Abex management, including discussions with test operations personnel, indicated a lack of confidence that reacceptance proof-pressure tests were actually performed on all refurbished hydraulic pumps. Fatigue and fracture mechanics analysis were performed on critical areas. Steady-state operational pressure, peak impulse pressure, vibratory loading, and water impact effects were considered. The critical flaw size was found to be a through crack. This would result in a leak-before-burst failure mode. Calculated life of the worst-case flaw size exceeds 5 flights. Rationale for STS-38 flight was: • The failure, due to test operator error, was believed to be an isolated case. • STS-38 SRB hydraulic pumps passed initial proof-pressure acceptance tests without incident. However, there was no confidence that reacceptance proof-pressure tests were performed after the last refurbishment. • Fatigue and fracture mechanics analyses of critical case areas indicated a leak-before-burst design and a worst-case life of 5 flights.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
2 (Continued)		• Leakage was not detected during any of the refurbishment or acceptance checkout operations. The pumps on STS-38 had flown previously only one time. Therefore, the pumps on STS-38 had more than adequate fatigue life to safely support the STS-38 mission. • SRB hydraulic systems are redundant. Failure of 1 hydraulic pump will not result in loss of hydraulic function. ***This risk factor was resolved for STS-38.***
3	STS-38 Reservoir Linear Variable Differential Transformer (LVDT) problem. HR No. B-20-21 Rev. B-DCN4 {C} ***There were no further problems with the LVDT on STS-38.***	RH Thrust Vector Control (TVC) system rock hydraulic reservoir LVDT remained at 74% while reservoirs were draining during system integration test hydraulic operations. The other 3 LVDTs tracked fluid removal. Proper reservoir level was verified by looking at the sight gage. The LVDT was tapped with a screwdriver, and the fluid level reading immediately went to 0%. The reservoir was cycled 3 times, and proper LVDT movement was verified. It was cycled 2 additional times during hydraulic closeout with no further problem noted. The LVDT is used prior to the launch countdown as a means of identifying hydraulic leaks. The LVDTs are checked periodically, and reservoir levels are recorded prior to entering the S0007 launch countdown. During S0007 countdown, however, hydraulic supply pressure is used to determine adequate levels of hydraulic fluid. Violation of hydraulic supply pressure redlines would result in a launch scrub and subsequent investigation of the problem.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
3 (Continued)		Rationale for STS-38 flight was: • System leak tests and inspections indicated a leak-tight system. • The LVDT tracked properly up and down; snapshots were taken up to launch with no anomaly noted. • Violations of redlines (hydraulic supply pressure and nozzle gimbal profile) between T−19 to T−10 sec would result in detection of gross leaks. ***This risk factor was resolved for STS-38.***
4	Actuator brackets found cracked during refurbishment of aft skirt. HR No. BN-05 Rev. B {AR} ***There were no SRB anomalies attributed to actuator bracket cracks on STS-38.***	During refurbishment of aft skirt S/N 015, cracks were found in the splice fittings of the upper ring to the actuator bracket. Corrosion was noted in the area of the cracks, indicating that the cracks had existed for some time. Aft skirt S/N 015 had not flown since 1985 on STS-20 and had flown only one other time on STS-14. The actuator bracket, splice plate, upper ring, and splice fittings were all 2219-T87 aluminum. There are 4 splice plate fittings per actuator bracket installation. Materials and processing analysis of S/N 015 confirmed that all 2219-T87 aluminum chemical, hardness, and strength requirements were met. SEM inspection confirmed that the cracks initiated at the splice fitting radius. Analysis confirmed that fractures in the splice fittings which led to the cracks were caused by ductile overload (one time event).

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u> 4 (Continued)		A review of the design found that the chamfer dimension on the splice fitting caused interference with the radius of the upper ring web. This splice fitting was determined to be the only SRB installation with this kind of interference. Tolerance buildup allowed gapping to occur at the fitting interface and torque loads applied during installation caused the fitting to fracture. A gap of only 0.001" is needed between the splice fitting and the upper ring-to-actuator bracket joint to fracture the fitting during installation. A stress analysis performed as part of this investigation found flight environment loads to be less than 10% of ultimate at the splice fitting radius. This analysis also determined that the splice fitting can be fractured along the radius without affecting the structural integrity of the actuator joint. The resulting FOS was still in excess of 1.4. An inspection of 12 aft skirts at United Space Boosters, Inc. (USBI) found no other cracked splice fittings. Because of the location of these fittings in the aft skirt, and due to aft skirt foaming, splice fittings on STS-38 SRBs could not be inspected prior to flight. Rationale for STS-38 flight was: • A splice fitting can be fractured without degrading the FOS below 1.4. • Flight loads are less than 10% of ultimate at the splice fitting radius. ***This risk factor was resolved for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
5	Improper spotface on SRB TVC Check Valve Filter Assembly (CVFA) and piston accumulators. HR No. A-20-04 Rev. C-DCN3 {C} B-20-09 Rev. C-DCN3 {C} ***There were no SRB anomalies attributed to improper spotfacing of the TVC piston accumulators.***	An improper spotface was found on a TVC CVFA during buildup inspection. An inventory search revealed improper spotfaces on 7 of 36 CVFAs and 3 of 18 piston accumulators checked to date. Because of these findings, all CVFAs and piston accumulators in the fleet, including those on STS-38 SRBs, potentially have this spotface problem. No other SRB TVC components examined as part of this investigation were found with similar problems. The concern was that presence of an improper spotface could affect the dynatube fitting sealing capability, leading to a high-pressure hydraulic fluid leak, and potentially resulting in an aft skirt fire. The investigation into the origin of this problem determined that it was the direct result of use of an improperly sized spotface tool during CVFA refurbishment operation. The origin of the improper piston accumulator spotface was not identified. The spotface tool cuts an O-ring bevel and shoulder for dynatube fittings. Use of the wrong tool leaves a 0.020" gap, because the shoulder cut is too large. Because the shoulder diameter is too large, the dynatube O-ring is prevented from being in complete compression. However, the O-ring seal is pressure assisted. If the improper spotface condition existed on STS-38 SRBs, O-ring extrusion into the 0.020" gap was insufficient to violate the seal. The investigation also determined that the CVFAs on STS-38 were not spotfaced during the last vendor refurbishment and, therefore, were not considered suspect. CVFAs and other hydraulic system components on STS-38 SRBs were subjected to a minimum of 8 leak checks, including vendor acceptance testing, acceptance checkout, and system integration tests. None of these tests resulted in identification of leak paths. Among leak checks performed was a relief valve cracking pressure test greater than 1.2 times the system operating pressures. Dynamic analysis of the improper spotface on CVFAs and piston accumulators was performed to determine acceptability for flight. The dynamic analysis results determined that preload precludes the fitting from backing off due to flight-induced vibrational loads.

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB 5 (Continued)		Marshall Space Flight Center (MSFC) Dynamics Laboratory assessment of the potential for fitting displacement in the seal area due to vibrational loads was found to be negligible. It was also learned during this analysis that the fittings are lockwired in place after preload is applied. Rationale for STS-38 flight was: • CVFAs on STS-38 SRBs were not spotfaced during vendor refurbishment and were therefore not exposed to the use of the improper spotface tool. • All STS-38 SRB hydraulic system components successfully passed a minimum of 8 leak tests. • Dynamic analysis of flight-induced vibrational loads demonstrated that the fitting would not back off. • Fittings are lockwired. ***This risk factor was resolved for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1	Potential Solid Rocket Motor (SRM) thrust imbalance. HR No. BC-06 Rev. B {AR} ***SRM performance on STS-38 was nominal. There was no indication of thrust imbalance during SRM operation.***	Due to an inadequate leak check of the nozzle joint #3 seals, Flight Set #10B aft segment was replaced with Flight Set #11B aft segment. As a result of the swapout, reassignment of aft segments for Flight Sets #10 through #13 took place. The reassignment created predicted worst-case thrust imbalance for Flight Sets #10 through #13 which exceeded the Contract End Item (CEI) specification limits. Waivers were written for Flight Sets #10, #11, and #12 to allow for the out-of-specification condition. Although the predicted potential thrust imbalance for these flights exceeded the CEI limits, it was within the thrust imbalance differential allowed by NSTS 07700, Volume X. The worst-case predictions of potential thrust imbalance on Flight Sets #10, #11, and #12 were within the performance capability of the Space Shuttle without additional risk to flight safety. The first reassigned aft segments were flown on STS-31, and the remaining sets were scheduled for STS-35 and STS-38. Data relative to measured thrust imbalance during STS-31 indicated that thrust actual imbalance was in the range of 20,000 pounds force (lbf). For STS-38, Discrepancy Report #400251 identified the worst-case thrust imbalance during the 98- to 103-sec time interval as 103,000 lbf. ***This risk factor was acceptable for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2	Putty on igniter inner gasket of test motors. HR No. BC-02 Rev. B {AR} BC-03 Rev. B {C} BI-02 Rev. B {C} ***There was no putty found on the igniter inner gasket on either STS-38 SRM upon disassembly.***	Putty was found on the inner gasket of Test Evaluation Motor (TEM)-5 and TEM-6 at the postfire inspection. The putty could impair gasket resiliency and allow blowby. The putty could mask a leak during a leak test, thereby preventing the detection of a potentially defective gasket assembly. The LCC was raised to 100°F at T–9 min to guarantee a seal temperature of 95°F at T–0. The igniter heater setpoint was raised from 95 ± 1°F to 110 ± 1°F. (For more information, see STS-31 MSE, L–1 Update, April 23, 1990, Section 4, SRM 4.) Rationale for STS-38 flight was: • The recommended LCC of 100°F and igniter heater setpoint of 110°F provided the required tracking factor with a potential TEM-5/TEM-6 type condition for the STS-38 as-built condition. The required tracking factor of 1.4 was met, and the tracking factor was greater than 1.0 for the highly unlikely condition of putty in 3 of 4 grooves in the seal. • Tests with putty in the gasket showed that the bolt preload is not affected, an overfill condition is not created, and the seal crown footprint is unaffected. • The design is safe because redundant seals exist and function as-designed (10 tests; SRM, DM-6, TPTA, JES, and NJES experience). STS-38 leak tests were normal and within the data base, and the gap closes after the igniter operation (0.55 sec). ***This risk factor was acceptable for STS-38.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
3	STS-31 right SRM igniter adapter-to-forward dome joint putty blowhole. HR No. BC-02 Rev. B {AR} BC-03 Rev. B {C} ***Putty blowholes were experienced on both STS-38 SRMs. Cadmium plating damage and sooting witnessed on STS-38 were within the SRM experience base. Because this was an expected occurrence, no formal Inflight Anomaly was recorded.***	A blowhole was found in the STS-31 right SRM adapter-to-forward dome (outer) joint putty at 180°, with no soot past the seals. Soot was noted on the outer gasket retainer Inside Diameter (ID) edge from 117° through 0° to 18°. Soot was also found on the inner igniter gasket retainer Outside Diameter (OD) edge and aft face of the full circumference. The cadmium plating was corroded from 155° to 220° with the majority of the corrosion between 175° and 185° on the igniter inner gasket retainer aft face and OD edge. Minor pitting to a maximum depth of 2 mils was also observed at the above location. A blowhole was also found on the STS-31 left SRM. A blowhole and pitting were observed on the STS-36 right SRM igniter/forward dome boss interface (IFA STS-36-M-01). (See STS-36 MSE, Postflight Edition, June 15, 1990, Section 7, SRM 1 for further details.) Rationale for STS-38 flight was: • Blowholes through the igniter joint putty were witnessed on the majority of flight and test SRMs, with no damage to the sealing capability of the joint (no evidence of blowby or damage of the elastomer and no damage to structural components). • Worst-case blowhole of 0.1" would result in no damage to the elastomer seal. • Worst-case analysis predicted a positive structural margin of safety. • There is no known mechanism that would lead to hot-gas circulation in the igniter joint. ***This risk factor was acceptable for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
4	Debris in the SRM nozzle flex bearing cavity. HR No. BN-06 Rev. B DCN64 {AR} ***There were no SRM anomalies attributed to debris found in the nozzle flex bearing cavity.***	Recent inspection of 4 flight SRM sets found an extensive amount of foreign material in the nozzle flex bearing cavity. The concern with this foreign material was the possibility of interference with the nozzle TVC functions and the potential fire hazard. The following is a summary of foreign material found: • A dry-fit bolt from nozzle internal joint #4 was found on STS-35. The bolt size was 1.5" long x 0.5" diameter with a 0.75" diameter head (removed). • "Bubble" wrap packing material and a 3" x 5" squeegee were found between the snubber and nozzle aft end ring on STS-40. • Small pieces of masking tape, a small square of "bubble" wrap, and a cotton swab were found on STS-39. • Two pieces of tape (8" x 1.5" and 2" x 1.5"), EA934 adhesive (0.5" diameter) at 185°, and polysulfide from 260° to 270° in the STS-38 LH SRM nozzle cavity. • EA934 adhesive at 210°, 270°, and 283° in the STS-38 RH SRM nozzle cavity. The discovery of the dry-fit bolt in the STS-35/SRM nozzle joint caused the most concern of all foreign material found. Analysis determined that the worst-case scenario occurs when the bolt becomes lodged between the nozzle snubber and the aft end ring. In this case, the bearing rubber would compensate to some unknown extent for the restriction; however, a possible new nozzle pivot point could be established. Structural assessment of this scenario indicated that the worst-case loading incurred by the lodged bolt could stall the nozzle actuator. If this occurs, it is anticipated that the failure mode would be local yielding of the snubber support ring with additional damage probable to the snubber segment. In the case

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u> 4 (Continued)		where the bolt remains lodged through splashdown, the snubbers would be inhibited from performing their splashdown function, resulting in significant damage to nozzle components. Borescope inspections of nozzle flex bearing cavities were performed on all completed nozzles at KSC and Thiokol and will be performed as part of a mandatory preshipment inspection for all future flight nozzles. Rationale for STS-38 flight was: • All known foreign material was removed from STS-38 SRM nozzles. • The STS-38 nozzle was returned to print and was safe to fly. ***This risk factor was resolved for STS-38.***
5	Aft dome factory joint internal insulation voids. HR No. BC-10 Rev. B DCN71 {C} ***There were no internal insulation anomalies reported on disassembly on STS-38 SRMs.***	Aft dome factory joint internal insulation verification was performed on all aft SRM segments using ultrasonic inspection techniques. On the STS-40 RH aft segment, ultrasonic inspection identified insulation voids. The aft dome factory joint was x-rayed, and 14 voids were discovered. This was the first time that the x-ray inspection technique was employed to verify insulation integrity. Nine additional SRM aft segments were x-rayed (all motors subsequent to STS-40), and similar internal insulation voids were found. The concern was the effect the voids would have on maintaining the required 2.0 erosion safety factor in the aft dome factory joint insulation. The most probable cause of these voids was the insulation layup process. This process was used on all SRMs to date. Minimum insulation layup was increased from 2.11" to 3.23" for reflight to comply with the increase to a 2.0 erosion FOS. Other factory joints using the

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
5 (Continued)		same insulation technique have less than half the required thickness of the aft dome joint. Because the aft dome joint insulation is so thick, it is difficult to avoid entrapping small amounts of air. Rational for STS-38 flight was: • Postfire evaluations of all SRMs were completed for the aft dome factory joint region with no erosion safety factor violations found (minimum experienced = 3.46). • Based on the process used, STS-38 insulation was concluded to have a safety factor greater than 2.0 over all factory joints. ***This risk factor was acceptable for STS-38.***
6	SRM Ignition Initiator (SII) leak test. HR No. BI-01 Rev. B {C} ***Disassembly and inspection of STS-38 SIIs did not identify grease in the leak check ports. An indentation was found on the LH SII O-ring; however, there was no resulting blowby. On the RH SII, raised metal (bubble) was discovered in the leak check port and is believed to be the result of the original machining process.***	A postflight anomaly report on STS-31 indicated that grease was found in the leak check hole (10-25 mils in diameter) on the SRM SII. Since that time, a record review determined that grease was visually noted at the leak test through a hole in 9 other SIIs at disassembly. Grease was also noted in the leak check port of the Flight Support Motor (FSM)-1 SII. This resulted in general concern regarding the validity of the leak test on the SII during seal installation. The SII is a modified NASA Standard Initiator (NSI) designed with redundant and verifiable seals. The primary O-ring is a shoulder seal (packing). The secondary O-ring is a face seal which seals against the weld washer of the SII. The SII assembly and inspection criteria include a visual inspection of all seal surfaces in the Barrier-Booster (BB) housing prior to assembly. The O-rings are greased with a thin film before installation. The torque is verified, the SII is lockwired, and leak test is performed. The seal leak test criteria includes a 50-psig pressure decay check. The allowable is no more than a 1-psi loss in a 10-min

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
6 (Continued)		period. Seal use history revealed that no leak test failures occurred in Redesigned Solid Rocket Motors (RSRMs) to date. The current SII and seal configuration has been used since STS-7 (282 samples demonstrated a reliability of 99.75 %). All showed no seal violations or seal distress for this configuration. (RSRM postfire inspection showed no soot past the third thread of the SII or up to the primary seal). Grease was visually noted at the leak test through a hole in several SII assemblies. Joint seal design provides a high confidence that the seals will function properly. Testing showed that shoulder seals provide reliable sealing at low torques. Tests on initiator pressure port plugs showed that a similar configuration will seal even with anomalous assembly. Tests without the primary seal indicated the secondary seal functioned as designed. Rationale for STS-38 flight was: • STS-38 SIIs, BB seal surfaces, and O-rings met all inspection criteria. • There was a demonstrated SII reliability of 99.75%. ***This risk factor was acceptable for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
7	STS-41 Stat-O-Seal rubber damage. HR No. BN-03 Rev. B {AR} ***There were no damaged Stat-O-Seals found on disassembly of STS-38 SRMs.***	Packing with retainers (Stat-O-Seals) on STS-41 were found to be damaged at disassembly. Although grease is used in the assembly process, the damage was attributed to abrasive adhesion of the rubber to the adapter metal surfaces after assembly. Disassembly damage is a common occurrence. However, potential failure of igniter inner secondary seals is a Criticality 1R function. Rationale for STS-38 flight was: • Stat-O-Seal integrity was verified 3 times by leak testing during the assembly process (including verification prior to final torquing). • Inner igniter primary seals functioned properly on the RSRM. • There was no evidence of heat effects on the STS-41 Stat-O-Seals. ***This risk factor was acceptable for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
KSC		
1	LOX T–0 carrier plate anomaly. HR No. INTG-089 {C} INTG-166 {C} ***There were no further problems with the LOX T–0 carrier plate after reinstallation.***	Damaged links were found on the LOX T–0 carrier plate. The carrier plate was removed and sent to the Rockwell Service Center for repair. During rework, it was found that the detent socket assemblies were loose. Links were removed, detents were adjusted, and a pull test was performed on the hockey stick mechanism to verify that detent load settings were correct. The carrier plate was reinstalled and was successfully mated to the Orbiter. Rationale for use with STS-38 was: • Retest prior to demate was successful. • Test of carrier plate was performed after mate. ***This risk factor was resolved for STS-38.***
2	T–0 umbilical foot support. HR No. INTG-089 {C} INTG-166 {C} ***There were no further problems experienced with the T–0 umbilical foot support after completion of repair activities.***	A Problem Report (PR) was initiated as a result of a sheared 3/8" bolt found on OV-102 RH T–0 umbilical. A summary of OV-104 attach hardware includes: • LH umbilical: – Lower 3/8" bolt was deformed. – Longeron fitting had minor thread scoring. – Upper 3/8" bolt was deformed. – Clevis fitting had minor thread scoring.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
KSC 2 (Continued)		• RH umbilical: – Lower 3/8" bolt was deformed. – Longeron fitting had significant thread scoring and elongation of bolt holes. – Upper 3/8" bolt was not damaged. – Clevis fitting had elongation of bolt holes. Efforts are underway to define T–0 umbilical foot support engagement verification techniques. Techniques under consideration include borescope inspection prior to carrier plate rotation or incorporating a view port in the longeron fitting. Rationale for STS-38 flight was: • All of the bolts were removed and replaced with new fasteners. • The bolt hole thread scoring was cleaned up, and elongated conditions were accepted on a MR for one flight only. • The procedures were modified to ensure proper engagement prior to mate. ***This risk factor was resolved for STS-38.***

RESOLVED STS-38 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>GFE</u>		
1	Unrestrained camcorder during STS-38 ascent/descent. HR No. CCH/T-001 {C} ***There were no problems associated with the use of the tethered camcorder on STS-38.***	The STS-38 crew requested authorization to use a Cannon Camcorder during the STS-38 mission. The camcorder was not manifested by the Configuration Control Board, and a concern developed that an unrestrained camcorder might impact the flight deck panels or a crew member, causing damage to the vehicle or launch/entry suit. The camcorder weight is 4.61 lb and includes a tethering system which consists of the camcorder harness assembly and the general-purpose tether assembly. The system allows a crew member to tether the camcorder for use during launch and reentry. The harness fits around the camcorder and attaches through a series of loops on the bottom. It is constructed of 1/2" Nomex webbing and Nomex size "E" thread. The tether is fabricated from 1" Nomex webbing with a 2" snap hook on one end and a 3" snap hook on the other end. The tether hooks have a functional limit of 1320 lb. The tensile strength is 900 lb for 1/2" harness webbing and 1200 lb for 1" tether webbing. The camcorder harness is attached to a 11.5" tether that is attached around the crew member's seat belt harness. The tether is sized to prevent impact with the forward crew members and aft panel/window [sized/designed for use at the Mission Specialist (MS) #1 seat only]. The camcorder is used by the MS #1 who is located behind the commander. Crew members are taught proper and safe use of the camcorder tethering system in training exercises and briefings. The camcorder is used during reentry and landing only. The camcorder, including the harness assembly and tether assembly, will be stowed during launch. ***This risk factor was resolved for STS-38.***

SECTION 5

STS-41 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-41/OV-103 mission, the previous Space Shuttle flight. 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.

The following risk factors, contained 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.

SRM 1	Solid Rocket Motor igniter outer joint putty blowholes with cadmium plating damage and sooting.

SECTION 5 INDEX

STS-41 INFLIGHT ANOMALIES

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
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} ***There was no similar anomaly reported on STS-38***	During performance of STS-41 post-insertion procedures, the crew discovered that GPC #2 had been assigned to string 3 according to SM2 NBAT; GPC #2 should have been unassigned. Investigation determined that this condition existed prior to launch and 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 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 exist. 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 Lockup, all LPS Ground Operations Aerospace Language (GOAL) programs that issue DEU equivalent commands verify proper SPEC prior to releasing the command. The IUS Telemetry Format Load (TFL) Lockup is now included 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 is limited to S0007 operations.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)		Rationale for STS-38 flight was: • IUS TFL Lockup procedure is only used for non-classified IUS flights; corrective actions were implemented to eliminate this exposure for future IUS missions. • Other software change procedures already incorporate verification techniques to protect against this type of error. • Nominal crew procedures verify correct NBAT configuration. ***This risk factor was resolved for STS-38.***
2	Aft compartment Hydrogen (H_2) concentration high during ascent. IFA No. STS-41-I-03 HR No. ORBI-306 {AR} ***Aft compartment H_2 concentrations during STS-38 prelaunch and ascent were within the OV-104 experience base.***	Postflight analysis of STS-41 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 the STS-41 catch bottles ranged from 25,000 standard cubic inches per minute (scim) to 37,000 scim. Average H_2 leakage during ascent for the fleet was 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 was a trend of increasing H_2 concentrations in the catch bottles. 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

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)		gas chromatograph. There are 6 catch bottles in each aft compartment, and it is not unusual to have only 1 to 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. At present, the cause of the leakage is unknown, and there were no indications of any leakage prior to launch. A possible leak source is the Space Shuttle Main Engines (SSMEs). Prior to engine start, only approximately 5% of the SSME H_2 joints are wetted. The 3 SSMEs on STS-41 were on OV-103 for 3 flights, and all were removed. Because of the potential for leakage, special tests, including bagging each SSME individually, were performed. Additionally, all Main Propulsion System (MPS) and SSME interface joints were leak checked prior to SSME removal with no anomalies reported. Rationale for STS-38 flight was: • Previous OV-104 flights have not demonstrated a similar trend of increasing aft compartment H_2 concentrations as has OV-103. • All MPS decay and helium signature tests were performed on STS-38/OV-104 with no anomalies identified. • Aft compartment H_2 concentrations during the recent STS-38 tanking test were well within the OV-104 experience base. ***This risk factor was resolved for STS-38.***

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
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-04 HR No. INTG-135 {C} ***There were no similar NSI anomalies reported on STS-38.***	Postflight inspection of STS-41 Solid Rocket Motors (SRMs) found that NSI lot number MPX, Serial Number (S/N) 1193, had been 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 testing or flight. NSI applications in SRM Igniter Initiators (SII) have had no similar ejection problems either in qualification tests or 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 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.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
3 (Continued)		Each NSI is subjected to proof-pressure testing at 15,000 pounds per square inch gage (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) has 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. Rationale for STS-38 flight was: • NSI blowout was expected based on the aft strut separation device qualification test experience. • NSI blowout is a post-functional aft strut separation device failure mode. • There is no history of similar NSI ejections in applications on the Orbiter. ***This risk factor was resolved for STS-38.***

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Auxiliary Power Unit (APU) Gas Generator (GG)/Fuel Pump (FP) heater system "B" failed on during STS-41. IFA No. STS-41-03 HR No. ORBI-104A {C} ***There were no APU heater system failures reported on STS-38.***	During Flight Day (FD) 4 checkout of the Flight Control System (FCS), normal switchover of the APU 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 temperatures peaked at 258°F approximately 3 min following switchover. The crew immediately switched back to the system "A" heaters, and normal GG/FP heater cycling resumed. This failure mode was 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 detonates at 350°F and this could 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 located where a clamp was used to secure the wiring to the fuel line. 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
<u>ORBITER</u>		
1 (Continued)		Retest of system "A" was performed after the thermostat changeout with no anomalies noted; however, no tests were performed on system "B". No APU thermostats were changed during the STS-38 flow. For STS-38, all APU system "A" and "B" heater circuits were operated during the Liquid Hydrogen (LH_2) tanking test on October 24, 1990. Console operators switched between system "A" and "B" heater circuits with no anomalies encountered. Additionally, because of the low ambient temperature in the aft compartment, thermostatic cycling of all APU heaters was successfully demonstrated. Action was assigned to the Orbiter Project at the STS-38 Flight Readiness Review to determine the acceptability for flight with a potential "smart" APU heater circuit failure on orbit, as experienced on STS-41. The response to this action is summarized as follows: • Two APUs, S/N 208 and S/N 305, were delivered from Sundstrand, where heater resistance checks were performed during acceptance testing. The third APU, S/N 311, came from OV-102 after STS-32. There were no APU heater or thermostat anomalies detected on STS-32. • Prelaunch tests were conducted during tanking for flight on all APU heaters. This test included switching between system "A" and "B" heaters.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)		• A new APU high-temperature FDA limit was set at 150°F. This provides 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 were performed in Acquisition of Signal (AOS) conditions only to allow ground monitoring of any failure. • On FD 1, heater reconfiguration was performed early, at 6 hour (hr) Mission Elapsed Time (MET), to allow verification of system "B" heaters. • Any APU heater anomalies detected during Loss of Signal conditions would result in the crew powering down all APU GG/FP heaters. Heater reconfiguration would, if needed, follow for the failed heater, and the remaining heater strings would reactivate. • To enhance response time, "booties" were 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.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)		Rationale for STS-38 flight was: • The above listed procedural changes were made in the event of an APU heater/thermostat failure. • All APU heater circuits were successfully checked out during the STS-38 tanking test, and there were no anomalies. • All 3 APUs are new to OV-104 since STS-36, and no work was performed on the APU heater circuits during the STS-38 flow processing. • OV-103 troubleshooting and flight history indicated a unique failure on STS-41. • APU heaters are redundant and are monitored by onboard FDA and the Mission Control Center (MCC) during operation. High-temperature limits were lowered to enhance response time. ***This risk factor was resolved for STS-38.***

C-2

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2	Inertial Measurement Unit (IMU) #1 experienced Z-axis accelerometer transients. IFA No. STS-41-04 HR No. ORBI-051 {C} ***There were no IMU anomalies reported on STS-38.***	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 to 15 min. 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 is a generic IMU problem; however, there have 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 3 or downmode to off in all OPS if IMUs are clearly degraded beyond use. This could result in possible crew/vehicle loss due to multiple axis second failure. If the second failure was multiple axis, it could 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 ground 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 would result in loss of crew/vehicle. Rationale for STS-38 flight was: • The IMU system has triple redundancy. Flight rules are in place to deal with single or double loss of redundancy. • This is not considered a generic IMU problem. ***This risk factor was resolved for STS-38.***

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3	Backup Flight Software (BFS) backup cabin delta pressure/delta temperature alarm was triggered at Main Engine Cutoff (MECO). IFA No. STS-41-05 ***There were no anomalies reported that were attributed to BFS delta pressure/delta temperature calculations on STS-38.***	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 determined that there was 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 reviewing this anomaly for a potential software change. The potential existed for a repeat of this anomaly on STS-38 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-38 crew was briefed on the potential for this alarm prior to launch.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)		Rationale for STS-38 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-38 crew was briefed on the potential for a repeat of this anomaly. ***This risk factor was resolved for STS-38.***
4	STS-41 Commander's LH Attitude Direction Indicator (ADI) rate scale switch failure. IFA No. STS-41-06 ***There were no switch failures reported on STS-38.***	During FD 4 deorbit preparations, a failure message was presented that identified that the Commanders LH ADI rate scale switch showed both "HI" and "MED" simultaneously. Postflight data analysis confirmed that the both signals were active simultaneously for 26 sec. There were 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. The switch P/N ME452-0102-7101 is a single-contact, triple-pole, hermetically-sealed toggle switch. This switch has a Criticality (Crit) 3/3 assessment in the ADI circuitry. Rockwell International (RI) identified 196 similar type switches (7106) at other locations in the Orbiter. Two applications are Crit 1R2, and 2 applications are Crit 1R3. There are no Crit 1/1 applications in the Orbiter.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4 (Continued)		There have been approximately 11,400,000 actuations of this switch (pre- and post-51-L), and one failed closed. The probability of this failure is very unlikely (probability = 8.77×10^{-8}). Rationale for STS-38 flight was: • This failure was not considered generic and was a first time failure occurrence (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 risk factor was acceptable for STS-38.***
5	Orbiter/ET LH_2 aft attach/separation hole plugger failed. IFA No. STS-41-07 HR No. ORBI-302A {AR} ***Aft attach/separation hole pluggers worked nominally on STS-38.***	The Orbiter/ET LH_2 aft attach/separation hole plugger failed to fully close. 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 and STS-34. There was concern that loose debris could block the ET umbilical door from fully closing, resulting in the potential loss of the crew and vehicle during reentry. The probability for 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

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)		latching is impeded. The Orbiter performs a maneuver at ET separation, moving away from the ET and possible escaping debris prior to ET umbilical door closure. Rationale for STS-38 flight was: • The probability for 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 potentially escaping debris. ***This risk factor was acceptable for STS-38.***
6	STS-41 LH Rotational Hand Controller (RHC) trim inhibit switch indicated a contact miscompare. IFA No. STS-41-08 ***There were no switch failures reported on STS-38.***	During FD 4 operations, a failure flag indicated that the LH RHC trim inhibit switch "ENABLE" and "INHIBIT" contacts were simultaneously made. Data review confirmed that both signals were present for a 15-sec period. The problem disappeared after this 15-sec period and was not repeated. 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). The are a total of 274 P/N ME452-0102-7201 switches per orbiter. Two applications (CRT SEL Switch #7 and #8) are located in the Data Processing Software System (DPS) and have been identified as Crit 1/1. Switch #7, the left-side CRT SEL switch, provides the means for switching the left keyboard from the

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)		left Cathode Ray Tube (CRT) to the center CRT or vice versa, and 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 were to occur 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. 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 previous 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 indicate that there were no failures related to age or number of cycles within the qualified lifetime. Ground turnaround tests, including keyboard testing, verified that all Crit 1/1 and 1R2 switches were functional.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)		Rationale for STS-38 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 risk factor was acceptable for STS-38.***

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	SRM igniter outer joint putty blowholes with cadmium plating damage and sooting. IFA No. STS-41-M-01 HR No. BC-02 Rev. B {AR} BC-03 Rev. B {C} BI-02 Rev. B {C} ***Putty blowholes were experienced on both STS-38 SRM igniter outer joints. Cadmium plating damage and sooting witnessed on STS-38 were within the SRM experience base. Because this was an expected occurrence, no formal IFA was recorded.***	Blowholes were found during disassembly of both STS-41 SRM assemblies. The blowholes, and resulting sooting and damage to the cadmium plating of the seal retainer, were similar to that seen on previous flight SRMs and test motors. (See Section 4, SRM 2 and SRM 3 for details of the previous experience.) The following is a summary of the STS-41 SRM findings. A single blowhole in the outer joint putty was located at the 165° position on the LH SRM . The blowhole measured 0.25" at the start and widened to 0.5" before contracting to 0.2" at the gasket face. This condition was similar to blowholes experienced on STS-36 (IFA No. STS-36-M-01). The significance of this occurrence was the cadmium plating damage on the outer Gask-O-Seal in an arc from 171° to 162° in line with the blowhole. Most of the cadmium plating was missing up to the cushion material, and small amounts of cadmium were folded up over the edge of the cushion. On the aft seal face, cadmium was missing from the retainer in the 92° to 108° area; the damage was not as severe as on the forward face. Pitting was observed at various locations on the adapter, with a maximum depth of 0.0015" at 165° which corresponded to the location of the blowhole. Soot was observed to the primary seal in the 126° to 171° area on the forward face. Light-to-intermittent corrosion was found 2.5" inboard of the adapter outer diameter for the full circumference. The igniter boss insulation was in good condition with normal charring and erosion. Igniter boss insulation char and erosion thickness measurements were taken at Zone A and the adapter 6.0" radial station. The 0.010" virgin insulation thickness requirement for Zone A was met, with the minimum thickness being 0.148". The 1.5 thermal safety factor requirement for the igniter was met; the lowest calculated safety factor was 3.19. No edge unbonds were found.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)		A blowhole was found in the outer joint putty at the 268° position on the Right-Hand (RH) SRM. The blowhole measured 1.3" at the start and was 0.25" at the through point. Soot was observed on the forward face to the primary cushion through 262° to 270°. There was soot on the full circumference of the gasket and on the aft face of the metal retainer through the arc 171° - 0° - 45°. There was no evidence of soot past the primary seal. Light heat effects to the cadmium were found from 262° to 270°. Light corrosion was present on the aft face and the inside diameter at 270°. The condition of the inner joint putty/insulation and chamber insulation was normal. Chamber insulation char and erosion measurements were taken at Zone A. The 0.010" virgin insulation thickness requirement was met; the minimum thickness was 0.070". The 1.5 thermal safety factor requirement for the igniter was met; the lowest calculated thermal safety factor was 3.69. The increase in putty blowhole occurrences is believed to be related to the reduction in putty layup in the case-to-adapter joint. This reduction in putty was directed to reduce the probability for putty to extrude into the joint sealing surfaces. Putty was seen in the joint and on the outer Gask-O-Seal of the STS-33 and other SRMs. It is expected that blowholes will occur on STS-38 SRMs because the putty layup process used was the same that was used on STS-31 and STS-36, which also had blowholes. Both outer Gask-O-Seals from STS-41 were sent to the vendor for evaluation. The evaluation determined that there were no heat effects on either Gask-O-Seal elastomer. Worst-case thermal analysis predicted no adverse elastomer damage due to blowhole heat effects.

STS-41 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1 (Continued)		Rationale for STS-38 flight was: • SRM flight history indicated no evidence of blowby or damage to the Gask-O-Seal elastomer. Examination of the STS-41 Gask-O-Seal sealing surfaces found no heat effects. • Worst-case thermal analysis predicted no elastomer damage. • Worst-case structural analysis predicted a positive margin of safety. ***This risk factor was acceptable for STS-38.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
2	Abnormal erosion on SRM aft segment factory joint internal insulation. IFA No. STS-41-M-02 HR No. BC-10 Rev. B-DCN71 {C} ***There were no SRM internal joint insulation anomalies reported on STS-38.***	Abnormal erosion at the forward edge of the internal insulation was observed on both SRM aft dome-to-stiffener and stiffener-to-stiffener factory joints. The erosion was most evident at the areas of ply overlaps. Erosion was random over the full circumference of the aft dome-to-stiffener factory joints and was found at only 4 to 6 locations on the stiffener-to-stiffener factory joints. The remainder of the internal case insulation was in very good condition, with no indication of unusual erosion or indication of hot-gas passage through the insulation. Measurements were made to determine the thickness of the remaining insulation. These measurements were used to calculate the resulting safety factor. Within 5" of the inner clevis tip, the required safety factor is 2.0; minimum calculated safety factor on STS-41 was 2.25. Beyond 5", the requirement is 1.5; minimum calculated safety factor on STS-41 was 1.57. Rationale for STS-35 flight was: • The resulting safety factor for STS-41 SRMs after erosion exceeded requirements. • Safety factors were predicted to be met on STS-38 SRMs if similar erosion occurred. ***This risk factor was acceptable for STS-38.***

SECTION 6

STS-36 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the OV-104/STS-36 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 risk factor in this section are listed beneath the anomaly 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.

SECTION 6 INDEX

STS-36 INFLIGHT ANOMALIES

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1	Engine #2027, nozzle #2027 bluing. IFA No. STS-36-I-01 HR No. ME-B7 (All Phases) {C} ***There was no nozzle bluing reported on STS-38 Space Shuttle Main Engines (SSMEs).***	During STS-36 postflight inspection, approximately 3" of bluing was noted on nozzle #2027 aft manifold adjacent to the High-Pressure Oxidizer Turbopump (HPOTP) primary drain exit line. This bluing was similar to that seen on STS-33 (IFA No. STS-33-I-01) and is believed to be a new nozzle phenomenon. On STS-33, the nozzles on both engines #2031 and #2107 were new; however, only the nozzle on engine #2107 showed bluing. Rocketdyne analysis found, through Rockwell hardness tests, that there was no annealing in the area of the discoloration. They approved the nozzle for further flight use. Causes resulting from contamination, ascent heating, improper material properties, and flight profile were all ruled out. The most probable cause of the bluing was descent heating during a steep reentry profile. Both STS-33 and STS-36 were Department of Defense (DoD) missions with high inclinations (actual inclinations are classified). All nozzles on STS-38 were flown previously. ***This anomaly was not a safety concern for STS-38.***

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Fuel Cell (FC) #2 Alternating Current (AC) phase "A" inverter failure. IFA No. STS-36-01 HR No. ORBI-127A {C} ***There were no FC system failures reported on STS-38.***	During the first launch attempt, FC #2 AC phase "A" inverter experienced numerous voltage and current fluctuations in a 2-minute (min) period. Fluctuations were from 112 volts AC (VAC) to 122.8 VAC; Operational Maintenance Requirements and Specifications Document (OMRSD) limit is 110-120 VAC. Inverter #4, Serial Number (S/N) 51, was removed from avionics bay #2 and replaced with S/N 42. FC #2 was satisfactorily retested prior to the next launch attempt. S/N 51 was returned to the vendor for failure analysis. The vendor was able to repeat the failure mode. The problem was isolated to loose connections within the inverter; four screws were found improperly torqued and loose. The remaining suspect units were forwarded to the vendor for proper screw torquing. Suspect units, S/N 38 and S/N 49, were removed and replaced. ***This anomaly was resolved for STS-38.***
2	Liquid Hydrogen (LH_2) 17" disconnect "B" open indication intermittent. IFA No. STS-36-02 HR No. ORBI-306 {AR} ***There were no LH_2 17" disconnect anomaly indications, real or erroneous, reported on STS-38.***	The LH_2 17" disconnect open indication dropped out for approximately 12 seconds (sec) during fast fill. The ground launch sequencer software issued an LH_2 stop fill command. No explanation was found for the dropout. Because the Launch Commit Criteria (LCC) requires only 1 of 2 indications, and the "A" indication was good, the LH_2 fast fill was resumed. The indication was normal for the remainder of the launch preparations. Troubleshooting by the vendor found no problems; switch operation was nominal. The anomaly was closed as unexplained. ***This anomaly was not a safety concern for STS-38.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Reaction Control System (RCS) thruster R3D failed "off" during External Tank (ET) separation. IFA No. STS-36-04 HR No. INTG-172 {AR} ***Thruster R1U indicated low Chamber Pressure (P_c) but operated nominally. The crew manually transitioned R1U to last in the RCS thruster priority list for the remainder of the flight. Postlanding R1U decay tests determined a pressure drop of 6 pounds per square inch (psi) to 8 psi in a 2-hour (hr) period. Sniff checks indicted no leak. See Section 7, Orbiter 7 for further details.***	P_c in thruster R3D S/N 228 did not reach the required pressure within the specified time period; therefore, the Redundancy Management (RM) system deselected the thruster. This failure occurred during ET separation and is a Crit 1R/3 failure [there are 3 redundant, down-firing thrusters on each Orbital Maneuvering System (OMS) pod]. Previous experience indicated that this failure was due to the oxidizer valve poppet not opening. Contamination, in the form of a "varnish" type deposit of nitrates, was found on oxidizer valve poppets on other failed thrusters. This contamination is created when the oxidizer [nitrogen tetroxide (N_2O_4)] comes in contact with moisture. It was previously noted that the rain cover was found missing from R3D prior to launch. A puddle of water was found in the R3D thruster throat and was successfully educed. The presence of water could be a contributor to R3D failure. The thruster was removed and sent to the vendor for failure evaluation. A replacement thruster was installed and successfully retested. Visual inspection at Dryden found no contamination in the thruster throat. Failure analysis at Marquardt determined that nitrates were formed in the oxidizer valve poppet, preventing it from opening in the allotted time. Rationale for STS-38 flight was: • RCS thruster redundancy. ***This risk factor was acceptable for STS-38.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Right RCS manifold #1 oxidizer isolation valve position indication intermittent. IFA No. STS-36-06A HR No. ORBI-244 {AR} ***There were no RCS oxidizer isolation valve position indication anomalies reported on STS-38.***	At L–4 sec, the right RCS aft oxidizer manifold #1 isolation valve open indication changed to not open. This mission indication caused RCS RM software to annunciate an "RM DLMA MANF" message and to override the right RCS manifold #1 closed indication at Solid Rocket Booster (SRB) separation. The right RCS manifold #1 was overridden to open via the override display after ET separation. The open indication returned when the crew moved the right RCS manifold #1 switch from "GPC" to "OPEN" after the OMS-2 burn. Troubleshooting at Kennedy Space Center (KSC) found no wiring anomalies. Changeout of the isolation valve LU318 was performed, and it was satisfactorily tested. Rationale for STS-38 flight was: • Redundant valves. • Preflight checkout. ***This anomaly was not a safety concern for STS-38.***

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Left RCS 3/4/5 "B" oxidizer tank isolation valve open position indication intermittent. IFA No. STS-36-06B HR No. ORBI-244 {AR} ***There were no RCS oxidizer isolation valve position indication anomalies reported on STS-38.***	At L–7 sec, the left RCS 3/4/5 "B" oxidizer tank isolation valve open indication changed to not open; the indication returned to open during a 2-sec period. The valve position indication worked well for the remainder of the mission. This failure was not representative of previous contaminated switch problems that occurred on-orbit and did not clear. This could be an erroneous data problem or loss of telemetry. Rationale for STS-38 flight was: • Redundant valves. • Preflight checkout. ***This anomaly was not a safety concern for STS-38.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	Left RCS 1/2 oxidizer crossfeed valve closed position indication intermittent. IFA No. STS-36-06C HR No. ORBI-244 {AR} ***There were no RCS oxidizer isolation valve position indication anomalies reported on STS-38.***	At L–45 sec, the left RCS 1/2 oxidizer crossfeed valve closed indication changed to not closed, identifying the loss of closed position. This loss of position indication occurred during a high-vibration period. The closed indication returned when the crew performed the post-OMS-2 burn reconfiguration, moving the left RCS 1/2 crossfeed switch from "GPS" to "CLOSED". This failure was not representative of previous contaminated switch problems that only occurred on-orbit and did not clear. A potential failure mode is that the Launch Process Sequencer (LPS) command could equal the maximum valve travel time (1.3 sec); therefore, the valve was potentially not driven into the stops/detente. Corrective action included increasing the LPS command to 2.0 sec. In addition, oxidizer crossfeed valve LV272 was replaced. Rationale for STS-38 flight was: • Redundant valves. • Preflight checkout. ***This anomaly was not a safety concern for STS-38.***

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	Water Spray Boiler (WSB) #2 vent system "A" heater failed. IFA No. STS-36-07 HR No. ORBI-170 {C} ***There were no WSB heater problems on STS-38. There was a problem with WSB #2 controller "A" cooling the Auxiliary Power Unit (APU) lube oil during ascent. See Section 7, Orbiter 1 for further details.***	WSB #2 vent system heater "A" began to degrade about 75 min after initial activation. Heater "B" was selected and operated nominally. Heater "A" was reselected for entry, was slow to come up, and operated erratically during reentry. This was a repeat of IFA No. STS-34-18, which was closed as unexplained because it could not be repeated. Troubleshooting for the STS-34 anomaly included operating "A" and "B" heaters for a number of cycles and shaking the wiring and connectors. Ice was considered to be the most probable cause for this anomaly. A similar anomaly occurred on OV-103 during STS-33 and STS-31. Additional testing after the STS-36 anomaly resulted in the removal and replacement of controller "2A". Testing at the supplier failed to duplicate the anomaly. A resistance test was performed on vent heater "2A" with nominal results. Troubleshooting at the vendor continues. An alternative heater activation sequence (activate system "B" following APU shutdown) is under evaluation. The alternate sequence was employed to obtain additional on-orbit data. Rationale for STS-38 flight was: • Inflight workarounds are adequate. • The heaters are redundant. WSB controllers are Crit 1R3. The Orbiter can safely return with nominal performance from 2 WSB systems. • The mission can be completed on a single heater string. • Heater loss can be detected in flight through temperature measurement. ***This risk factor was acceptable for STS-38.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8	STS-36 APU #1 hydraulic flex hose anomaly. IFA No. STS-36-08 HR No. ORBI-036 {AR} ORBI-047A {AR} ORBI-184 {AR} ORBI-188 {C} ***There were no APU flex hose anomalies reported on STS-38.***	Hydraulic system #1 exhibited anomalous indications during STS-36 ascent. The reservoir pressure dropped as expected after SRB ignition, but unexpectedly continued to drop. The reservoir temperature increased as expected, but the 6% volume increase due to temperature rise did not materialize. Early during reentry, flight controllers concluded that hydraulic system #1 was leaking at a rate that would likely deplete the fluid and cause system #1 shutdown prior to landing. Controllers directed the crew to temporarily select "low pressure" on system #1 in order to reduce the leakage rate in an attempt to assure availability of system #1 during the approach and landing at Dryden Flight Research Center (DFRC). The action was successful, but the reservoir volume had decreased to 27% by the early postlanding APU shutdown. An extensive discussion of this anomaly and the associated failure analysis activities are found in STS-36 MSE, Postflight Edition, June 15, 1990, Section 7, Orbiter 8. Postflight inspection of the aft compartment found hydraulic fluid sprayed over most of the aft compartment components. Hydraulic leakage was tracked to a high-pressure flex hose S/N 153, which was removed and sent to Rockwell International (RI) for failure analysis. The flex hose Part Number (P/N) ME271-0079-1129 is certified to MIL-H-38360A. It has a 0.045"-thick teflon liner, surrounded by 2 braids of stainless steel and covered by a 1/8" rubber chaffing strip. It is 42" long with an inner diameter of 1". Installation is supported by a saddle with a slight bend. This type of flex hose is proof-pressure tested at 6000 psi. A 1/2" split was initially found in the stainless steel braid at the leak site. This cut indicated that the leak was not caused by external forces (i.e., twisting). Failure analysis at RI found a pin hole in the teflon liner, 19" from the swage fitting. Additionally, 2 kinks were found on the teflon liner about 1-1/2" from the pin hole and were considered contributors to the leak. X-ray inspection found no cracks in the metal end fitting.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		RI performed a pneumatic test of the flex hose, submerged in water, to verify the leak. Subsequent to leak verification, an extensive teardown was performed. Titeflex, the vendor, assisted RI with the failure analysis. Similar flex hoses are used in NASA, DoD, and commercial aircraft applications. A review of flex hose applications found no other similar failures of high-pressure flex hoses were experienced in any application to date. All leaks were found to have originated at the hose fitting end(s). Three flex hose leaks occurred in Orbiter Program applications. All leaks were in fittings, not in the liner. The first was in August 1975, where a flex hose failed proof test due to a thin fitting wall. The second was experienced in November 1979, and a fitting leak indication was found; however, further examination and subsequent use found no repeated problem. The third was found during postflight inspection of STS-1 hydraulic systems. A crack was found in a fitting that had previously passed proof tests. Subsequent failure analysis found that this surface crack did not leak or degrade the performance of the fitting. The failed hose was initially installed on OV-104 during original production at Palmdale. High-pressure flex hoses on OV-104 experienced the least amount of operating time, approximately 9 hr, of all Orbiter vehicles. The actual leak originated at the center of the flex hose, approximately 16" from the crimp sleeve on the elbow end. External leakage was through a longitudinal split in the chafe guard, occurring approximately 12" from the leak in the teflon liner. The area around the split appeared as a blister in the extrusion skin. An area of permanent deformation, or kink, was also observed approximately 1" from the leak site. Microscopic examination of the leak site confirmed that the leak was at a

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (continued)		longitudinal surface crack, originating in the inner diameter of the liner. A semi-elliptical shaped flaw, 0.180" x 0.036", had grown and broken through the teflon liner to a leak site of 0.020". X-ray examination found no break or disturbance in the braid layers. Laboratory tests using fractography were performed on three 1/2" sections of the failed teflon liner. The sections were used as test coupons and subjected separately to monotonic loads with and without a 0.180" x 0.036" notch, and to fatigue cycling. Results of fractographic analyses of these coupons, and comparison to the failed area of the flex hose, determined that the initial surface crack or flaw formation mechanism was unknown; it could not be conclusively tied to overload, fatigue, or sustained stress. In addition, the internal surface crack was extended by Low-Cycle Fatigue (LCF) from pressure rippling effects associated with pump operational characteristics; this led to final flaw breakthrough and a stable leak. In another test, a remnant portion of the failed hose was bent to a radius of approximately 5" to evaluate kinking and buckling characteristics. This radius produced 2 transverse buckles on the compression side and 1 longitudinal buckle on the tension side. Compression-side buckles were found to have a similar appearance to buckles found approximately 1" from the STS-36 flex hose leak site. Because longitudinal craze damage, or "whitening", was observed near all buckles of the failed flex hose liner, effects of flex hose pulsing on the craze area were also examined. This was initiated because of the theory that mishandling or unacceptable bending of the flex hoses had occurred and, with the pulsing effects induced by normal operation, led to the failure on STS-36. Fatigue tests were performed. After 1.3 million cycles, no cracks were found in the craze area. This represents only half the number of cycles in a single flight for one of these hoses;

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		however, these tests were performed using a much higher load profile. Similar tests were performed with a teflon hose liner notched to simulate the surface flaw on the failed hose. Through the cycling of the hose, growth of this flaw or crack was achieved. The results of these tests led to the conclusion that a crack cannot occur as a result of bending or mishandling alone; a crack initiated by improper sintering or other flaws can grow as a result of mishandling and operational cycling. Chemical properties of the failed teflon liner were analyzed and compared to reference properties for unsintered teflon, minimally sintered teflon, and teflon after maximum sintering. These inputs were provided by duPont, the teflon manufacturer. Analyses of the failed teflon through infrared spectrophotometry and differential scanning calorimetry techniques – comparison to reference properties of the melting point, specific gravity, heat of transition, and tensile strength – determined that the failed teflon liner was minimally sintered. "Bump extrudate" occurs during manufacture after the liner is extruded but not cured. The liner may bump against the side of the extrusion machine. This results in a bump that, when cured, will not achieve proper temperature. The bump extrudate phenomenon causes minimum sintering, believed to be the originating factor leading to the STS-36 flex hose failure. This type of flaw should be caught during quality inspection testing after manufacture through a crush and roll test. The crush and roll test flattens the teflon liner under glass for observation and detection of similar flaws. The lot from which the failed STS-36 flex hose was made did not have a quality stamp, or buy-off, next to the crush and roll test on the buy-

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		off sheet. All other required tests had the appropriate quality stamp of approval next to the test step. It is believed, therefore, that either the crush and roll test was not performed on this lot of teflon liner or an anomaly was discovered which the quality inspector could not accept. The results of the failure analysis and investigation previously described are as follows: • The leak resulted from a single crack in the teflon liner, which grew by fatigue from a surface flaw on the inner diameter. • The surface flaw formation and growth was facilitated by the combination of: – Local incomplete sintering in the flaw area, probably because of a phenomenon called "bump extrudate". – Minimally complete sintering of the entire liner. – Buckling of the hose, possibly caused by mishandling. – Stressing (LCF) induced by the operational environment. • It is probable that a test was omitted which would have detected the lack of sintering.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		• There is a low probability that the combination of the factors which led to the STS-36 flex hose failure will be repeated. • There was no indication that this is a generic teflon liner problem. The Johnson Space Center (JSC) Safety Division researched the concern for fire resulting from a hydraulic fluid leak into the aft compartment. Two locations on each APU exceeded the autoignition point of MIL-H-83282A hydraulic fluid used in the Orbiter. These were the injector well (1200°F) and the interface area of exhaust duct to APU housing (1100°F). These surfaces are covered with insulation and stainless steel foil, except the injector well which has Kao-wool insulation which is also a liquid barrier. Except for "smart leaks", it is not credible for the hydraulic fluid to come in contact with the high-temperature areas. The number and location of hydraulic fluid autoignition temperature exceedences on the main engines were not completely catalogued. There are several which may approach or exceed the autoignition temperature and are not insulated or isolated. Tests performed at White Sands showed that spraying a hydraulic mist at 3000 psi at an oxyacetylene torch, or at an arcing energy source at 150 amps, would cause the hydraulic fluid to ignite. Varying the distance to the ignition source varied the size of the flame cone, but the flame did not propagate upstream. No spray test onto a hot surface was performed, so the potential for lowering the autoignition

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		temperature due to a fine spray was unknown. Therefore, it was necessary to consider other mitigations to the potential for igniting a hydraulic fluid leak in the aft compartment. APU/hydraulics operation is limited to ascent and reentry phases. On a straight-line basis assuming no purge effects, hydraulic fluid combustion is oxygen dependent and could be sustained in a practical sense only to an altitude of 80,000 feet (ft). Because the autoignition temperature increases with decreasing pressure, autoignition temperatures rise such that the threat would no longer exist. The threat, both on ascent and descent, is mitigated by purge effects. The Gaseous Nitrogen (GN_2) prelaunch purge dilutes possible air intrusion and remains positive during ascent due to decreasing pressure as the Space Shuttle climbs. During reentry, a Main Propulsion System (MPS) helium purge is initiated at approximately 80,000 ft and continues until wheel stop plus 100 sec. While there is a finite probability of ignition of the hydraulic fluid, the mitigation measures make the probability low and a secondary risk to the potential of losing the use of a hydraulic system due to hydraulic fluid depletion. The flex hose on OV-103 hydraulic system #2, S/N 003 from lot #20W598C, did not have each individual test step, such as crush and roll, Quality Control (QC) stamped. Instead, the acceptance test sheet for lot #20W598C had the Quality Assurance Director's signature and stamp. Titeflex provided assurance to JSC Safety, Reliability, and Quality Assurance (SR&QA) that this indicated that all tests were performed and passed. This procedure is sometimes used if the "traveler" sheet that accompanies the flex hose lot through each test has the appropriate test steps stamped.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8 (Continued)		Relative to the high-pressure flex hoses on STS-38/OV-104, acceptability for flight was based on the following: • There was an excellent history of performance of this type of flex hose throughout DoD and NASA; the STS-36 failure of this type was the first. • Buckling of a non-flawed area of a flex hose would not initiate or propagate a crack in a flight environment. • There was no indication of a generic problem with high-pressure flex hoses. • High-pressure flex hoses on STS-38 were not from the same lot as that which failed. ***This risk factor was resolved for STS-38.***

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	Cathode Ray Tube (CRT) #4 screen went blank. IFA No. STS-36-09 ***There were no CRT anomalies reported on STS-38.***	During STS-36, CRT #4, the aft Display Unit (DU), went blank, and a power supply problem was indicated (IFA No. STS-36-12). Power cycling regained temporary use of the DU; however, after the third failure, it was turned off for the remainder of the mission. Loss of this DU is a Crit 1R3 failure. Failure analysis isolated a cracked solder joint on a capacitor in the horizontal deflection amplifier page at location C37. A review of failure history found 4 previous solder joint failures at location C37. Two failures were attributed to poor solder joint wetting, 1 was found to have a cracked solder joint and was attributed to stress, and the fourth was attributed to a cracked plated-through-hole in the multilayer interconnection board. All failed solder joints were from the same lot, with serial numbers above S/N 030. The failed STS-36 DU had over 17,000 hr of operation. Other DUs with failed C37 solder joints also had high operating times. Because of these previous failures, there was a concern that this problem was generic. There are 2 failure mode theories: • The capacitor is physically too close to a screw hole used to mount a board stiffener. This hole may act as a heat sink during soldering which leads to poor solder wetting. • Because the capacitor is glued to the board prior to soldering, it is conjectured that the adhesive may expand and contract during thermal cycling causing stress at the solder joint.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
9 (Continued)		Rationale for STS-38 flight was: • Preliminary assessment had not revealed a generic problem. • There are 4 CRTs available; 2 or more CRTs must fail before there is a mission impact. ***This anomaly was not a safety concern for STS-38.***
10	Oxygen (O_2) bleed orifice leak. IFA No. STS-36-10 HR No. ORBI-299 {C} ***There was no O_2 bleed orifice anomaly reported on STS-38.***	During cabin operations at 10.2 psi, the crew manually controls the O_2 content of the atmosphere. Presleep O_2 partial pressure was 2.85 psi. Per procedures, the crew connected the bleed orifice, resulting in a rise of O_2 partial pressure to 2.9 psi during the sleep period. The crew tightened the elbow B-nut in an attempt to slow any possible leak, and the bleed orifice functioned nominally. It is thought that the rise in O_2 partial pressure may have been due to a manual control lag effect. Rationale for STS-38 flight was: • The OV-104 O_2 bleed orifice was checked prior to flight with no problems found. ***This anomaly was not a safety concern for STS-38.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11	Free water found near humidity separator "A". IFA No. STS-36-11 HR No. ORBI-051 {C} ORBI-254 {C} ORBI-321A {C} ***There was no loss of water from the humidity separators reported on STS-38.***	The crew found 1 to 2 cups of water outside humidity separator "A" during required inspection procedures. The Waste Water Management System (WWMS) wand was used to recover the free water. The crew reconfigured to use humidity separator "B" for the remainder of the mission. Because of recent free water problems experienced on STS-32/OV-102, a decision was made to remove the humidity separator package and send it to the vendor for failure analysis. Vendor inspection at Dryden found contamination in the condensing heat exchanger, an area of the separator where it was not previously seen or experienced on other problem humidity separators. Vacuum cleaning of the STS-38/OV-104 condensing heat exchanger was performed. All contaminants were removed. Humidity separator water flow tests were successfully completed. Rationale for STS-38 flight was: • Crew procedures are in place which require periodic inspection of the humidity separators and surrounding areas for free water. • Contingency deorbit can be implemented should both separators and workaround procedures fail. ***This risk factor was acceptable for STS-38.***

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
12	Thruster R4R failed off during pre-entry hot-fire test. IFA No. STS-36-12 HR No. ORBI-119 {AR} ***Thruster R1U indicated low P_c but operated nominally. The crew manually transitioned R1U to last in the RCS thruster priority list for the remainder of the flight. Postlanding R1U decay tests determined a pressure drop of 6 psi to 8 psi in a 2-hr period. Sniff checks indicted no leak. See Section 7, Orbiter 7 for further details.***	During thruster R4R S/N 235 firing, the P_c did not reach the required pressure within the specified time, and the thruster was deselected by the RCS RM system. This was a failure mode similar to thruster R3D failure. Contamination of the oxidizer valve poppet was suspected. Prior to launch, the rain cover of R4R was found soaked with water. Thruster R4R was removed and sent to the vendor for analysis. Visual inspection at Dryden found no contamination in the thruster throat. Failure analysis at Marquardt determined that nitrates were formed in the oxidizer valve poppet, preventing it from opening in the allotted time. Rationale for STS-38 flight was: • There are 4 redundant yaw firing thrusters on each OMS pod. • The aft RCS can handle any 2 thruster failures. ***This risk factor was acceptable for STS-38.***

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
13	Flash Evaporator System (FES) controller "A" shutdown. IFA No. STS-36-14 HR No. ORBI-276B {C} ***There were no FES controller problems reported on STS-38.***	FES controller "A" shut down during on-orbit operations. The shutdown occurred when the water dump mode was initiated. The crew selected the high radiator set point in an attempt to correct the problem. A transducer sensed inadequate FES cooling, which resulted in FES controller "A" shutdown. The crew then cycled the radiator switch twice; the FES came on line and performed nominally. Troubleshooting at KSC found no problems with the controller. Rationale for STS-38 flight was: • OMRSD requirements verify the integrity of the FES prior to launch. • Adequate redundancy is available. • Crew workaround procedures are available. ***This risk factor was acceptable for STS-38.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14	STS-36 hydraulic system depressurization anomaly. IFA No. STS-36-17 HR No. ORBI-036 {AR} ORBI-047A {AR} ORBI-184 {AR} ORBI-188 {C} ***There were no hydraulic system anomalies reported on STS-38.***	During reentry, hydraulic system #1 pump pressure did not respond properly when commanded to the low-pressure mode to reduce the rate of a suspected hydraulic leak. Initially, pressure dipped to 2100 psi and then leveled off at 2600 psi for 5 min before dropping to 650 psi; nominal low pressure is 800 psi (500- to 1000-psi range). The low-pressure mode is normally required only for APU start. This was the first time to command a hydraulic system to "low" during reentry; however, periodic load tests verify this capability. Pressure was returned to normal shortly before landing and shutdown after wheel stop. A complete discussion of this anomaly and failure analysis are found in STS-36 MSE, Postflight Edition, June 15, 1990, Section 7, Orbiter 15. Pump teardown revealed severe scoring in the piston cap through the entire bore, 360° in circumference. Additionally, 6 score marks were found through the housing bore hardcoat anodize. Burnishing was also indicated on the piston. This pump previously flew on OV-099 and was removed after 5 flights to investigate an anomalous APU vibration. APU pump testing at JSC determined that the vibration was caused by the APU. After being subjected to this anomalous vibration environment on OV-099 and testing at JSC, the pump went through acceptance testing prior to installation on OV-104. STS-36 was the fourth flight for this pump on OV-104. In addition to operation on OV-099 and OV-104 and testing at JSC, the pump was used with 3 other hydraulic systems at Abex (the vendor), the KSC Orbiter Processing Facility (OPF), and Sundstrand. This was significant because analysis of cap and piston fluid samples found 73 particles larger than 100 microns; the largest were found to be 300 series Corrosion Resistant Steel (CRES), MP35N, and iron oxide, ranging from 125 to 360 microns. None of these materials were used in the

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

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

14 (Continued)		A key factor in determining whether the scoring and wear were operating-time related was the examination of the 3 pumps operating in the Flight Control Hydraulics Laboratory (FCHL). The total operating times of FCHL pumps ranged from 682 to 916 hr. These pumps experienced thousands of depressurization/ pressurization cycles and hundreds of operating hours in the depressurization mode. Examination of these pumps found only indications of incipient wear and no scoring. The pumps were considered to be in good condition. The vibration environment experienced by the FCHL pumps is extremely low compared to flight environments on the Orbiter and SRB. Examination of other pumps at Sundstrand and in commercial industry is underway; however, there is no indication that major problems will be found.
		
		The results of the STS-36 pump anomaly investigation formed the rationale for STS-38 flight:
		
		• The STS-36 depress anomaly was not a hard failure.
		  – The anomaly cleared itself.
		  – The anomaly occurred during off-nominal operations.
		  – There was no prior history of depressurization problems with this or other pumps.
		
		• Examination of other pumps determined that the piston cap and housing bore scoring found in the anomalous pump was by far the worst case.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
14 (Continued)		• The anomalous pump was subjected to an unique set of operating environments: – Exposure to excessive vibration on OV-099. – Exposure to 6 different hydraulic systems, increasing the opportunity for introduction of contamination. • Piston cap scoring did not appear to be operating time related. – FCHL pumps indicated slight scoring of the piston caps. – JSC test stand pumps indicated very little evidence of scoring. – SRB pumps indicated the beginnings of localized wear after a relatively short operating time. • Anomaly characteristics and failure analysis evidence were consistent with transient contamination. • Similar conditions existing on the STS-38 pump would result in inability to start the APU at T–5 min and cause a launch scrub. For the on-orbit start, redundant capability exists among hydraulic systems for reentry. ***This risk factor was acceptable for STS-38.***

STS-36 INFLIGHT ANOMALIES

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

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Right Solid Rocket Motor (SRM) igniter/forward dome boss interface surface metal pitting and Gask-O-Seal damage. IFA No. STS-36-M-01 HR No. BC-02 Rev. B {AR} BC-03 Rev. B {C} ***Putty blowholes were experienced on both STS-38 SRM igniter outer joints. Cadmium plating damage and sooting witnessed on STS-38 was within the SRM experience base. Because this was an expected occurrence, no formal Inflight Anomaly (IFA) was recorded.***	During disassembly of the STS-36 booster assemblies, a blowhole was found at the 175° position in the igniter vacuum putty. This putty was laid-up between the igniter and the forward dome as a thermal barrier to stop hot-gas excursion to the igniter-to-case sealing surfaces. The blowhole measured 0.3" circumferentially at the igniter adapter and widened to 2.5" circumferentially at a position 4" below the adapter. Blowholes through the putty were experienced on approximately 65% of all flight and test SRMs/Redesigned Solid Rocket Motors (RSRMs); however, the results were not as severe as that witnessed on the STS-36 Left-Hand (LH) SRM. Significant to this occurrence was the discovery of a depression, or pitting, in both the inner diameter of the forward dome and the outer diameter of the igniter chamber body, as well as a missing portion of the cadmium plating on the inner igniter gasket seal. Sooting was also seen around the outside of the inner igniter gasket seal, extending approximately 100° in either direction from the blowhole. Blowholes were observed on the STS-27 Right-Hand (RH) SRM and on Transient Pressure Test Article (TPTA) 1.2; corroded metal surfaces and cadmium plate were also similar to the STS-36 LH SRM. In these cases, the minimum blowhole circumferential measurement was 0.16". This supported the belief that corrosion is not worse with smaller blowholes. A similar blowhole was found in the STS-31 adapter-to-forward dome (outer) joint putty at 180°. Details regarding this anomaly and associated failure analysis activities are found in STS-36 MSE, Postflight Edition, June 15, 1990, Section 7, SRM 1. The blowhole through the putty was large enough to allow sufficient hot gas to pass to clean the putty off the surfaces of the forward dome and igniter case. Pitting of both of these surfaces is believed by Marshall Space Flight Center (MSFC) and Thiokol Corporation metallurgists to be due to "corrosion" as opposed to hot gas "erosion". The hot propellent gases contain a large amount of chlorine, hydrogen

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)		chloride, and other corrosive materials. The chlorides are believed to be the primary cause of the corrosion; however, the corrosion process was continued by sea water after splashdown. The pitting is refurbishable; igniter chamber pitting was measured at 1 to 2 mils in depth. In addition to the pitting of the forward dome and igniter case, examination found missing cadmium plating from the Gask-O-Seal over an area of 1.5" circumferentially by 0.15" radially in the area of the blowhole. Metallurgical analysis found powdery cadmium chloride. The melting point of cadmium is 610°F, and it oxidizes when heated. From hardness testing, observations of metal corrosion, and removal of cadmium plating from the Gask-O-Seal, it was estimated that the igniter joint experienced a temperature in the range of 450°F to 550°F. This finding indicated that the cadmium was removed through corrosion as opposed to melting. Thiokol engineers stated that cadmium stripping is acceptable as long as there is no damage or degradation of the elastomer seal. In this case, no degradation of the elastomer was found. Analysis by Thiokol showed that exposure to temperatures up to 800°F are acceptable for seal performance. The volume on the seal side of the blowhole was very small (3.8 in^3 versus 15 in^3 for a field joint and nozzle-to-case joint). It had a 0.61-sec fill time, and there was no circulation producing additional flow in this area. Therefore, the temperature rise was limited to less than 800°F. Flow/thermal analysis of a worst-case blowhole, measuring 0.1" circumferentially, was performed. A blowhole of 0.1" is considered worst-case because no blowhole was observed to be less than 0.16", and blowholes less than 0.1" would tend to self-plug. For this size blowhole, the void fill time is determined through this

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)		analysis to be 2.4 sec. No damage to the seal would result because the seal surface temperature would be below 450°F, well within the 800°F limit. The analysis showed, however, that the cadmium on the retainer would be exposed to temperatures greater than 610°F, the melting point for cadmium, for a period of 1.2 sec until flow stagnation would occur. It was determined that even if the cadmium melts, no embrittlement or damage to the elastomer was expected. The fact that damage seen on STS-27 and TPTA, with a blowhole of 0.16", was similar to that seen on STS-36 was an indicator that the analysis results were conservative. Worst-case thermal analysis of the igniter chamber steel indicated that the surface temperature rose to 2750°F. This prediction was based on the pitting seen, less than 2 mils in depth. At 2750°F, analysis showed that there was no loss in structural margins of safety. Stresses in the heat-affected zone ranged from 40-to-140 ksi. The overall joint capability was determined not to be compromised by the localized heat-affected zone. The joint Factor of Safety (FOS) was demonstrated by burst tests to be greater than 1.8. Based on the localized heat-affected zone experienced on STS-36, the remaining margin of safety was greater than 0.3. The only resulting concern with a localized heat-affected zone is reuse because of the loss of corrosion protection on the metal surface. A thorough analysis of the likelihood of circulation flow within the igniter joint found no mechanism to generate circulation in the joint. The igniter joint is unlike the nozzle-to-case joint, where nozzle gimballing occurs, or the field joints. In both cases, the dynamic environment provides the potential for creation of a delta pressure in the joint, leading to circulation. In addition, of all SRM igniter joints experiencing putty blowholes, none were seen with more than 1 blowhole.

STS-36 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1 (Continued)		Rationale for STS-38 flight was: • Blowholes through the igniter joint putty were witnessed on the majority of flight and test SRMs, with no damage to the sealing capability of the joint (no evidence of blowby or damage of the elastomer and no damage to the structural components). • Worst-case blowhole of 0.1" would result in no damage to the elastomer seal. • Worst-case analysis predicted a positive structural margin of safety. • There is no known mechanism which would lead to hot gas circulation in the igniter joint. ***This risk factor was acceptable for STS-38.***

SECTION 7

STS-38 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-38/OV-104 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-38 INFLIGHT ANOMALIES

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1	Water Spray Boiler (WSB) #2 did not cool Auxiliary Power Unit (APU) lube oil while under operation of controller "A". IFA No. STS-38-01 HR No. ORBI-036 {AR}	WSB #2 controller "A" failed to cool APU lube oil after the end of the pool boiling period during ascent. The crew switched to controller "B" when the temperature reached 275°F, and APU #2 was left "on" after APUs #1 and #3 were shut down. After switching to controller "B", lube oil temperature peaked at 300°F before cooling was observed after 1 minute (min) 6 seconds (sec). Controller "A" was selected for reentry to determine if temperature control operated properly, and controller "A" operated normally. It is believed that an apparent spray bar freeze-up on controller "A" during ascent caused the problem. Freezing of the spray bar could be due to low heat load on APU #2, or controller "A" was not functioning properly. A similar cooling problem was experienced on STS-1 through STS-4. Troubleshooting on controller "A" is continuing.
2	Flash Evaporator System (FES) water supply accumulator heater system biased low. IFA No. STS-38-02 HR No. ORBI-276B {C}	FES heater #1 did not cycle "on" within its prescribed temperature range of 55-75°F. When the temperature reached 49°F, heater string #2 was activated and heater string #2 cycled in the 48-54°F range with apparent normal duty cycles. Heater string #1 was reactivated and cycled like heater string #2. A temperature sensor debond problem is the suspected cause of this anomaly. FES heater #1 will be removed and replaced at Kennedy Space Center (KSC).

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	APU #3 X-axis acceleration trace erratic. IFA No. STS-38-03B	During reentry, APU #3 X-axis acceleration trace was erratic. The problem was believed to be a failed accelerometer. There was no previous accelerometer failure history. Troubleshooting at KSC will include connector and accelerometer checkouts. APU #3 will be removed and replaced due to life-time cycle limits.
4	Vacuum cleaner short circuit. IFA No. STS-38-04A HR No. ORBI-301 {C}	When the crew turned on the vacuum cleaner, Circuit Breaker (CB) #29 on panel L4 was activated by a current surge. Utility outlet M013Q was not used during the remainder of the flight. Utility outlet M013Q provides electrical interface for the food heater and vacuum cleaner. Outlet testing was completed, and the outlet tested good. The vacuum cleaner was removed and sent to the Johnson Space Center (JSC). Postflight troubleshooting verified a phase "B" short-to-case on the vacuum cleaner. Prelaunch vacuum cleaner checkout will be modified to include a check for shorts on all 3 phases. The vacuum cleaner was replaced with a stock vacuum cleaner that passed the new checkout procedures for shorting condition on all 3 phases.

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	APU Exhaust Gas Temperature (EGT) instrumentation interaction with injector tube temperature instrumentation. IFA No. STS-38-05 HR No. ORBI-106A {AR}	APU #2 EGTs #1 and #2, and APUs #2 and #3 injector tube temperatures, became erratic during launch. APU injector tube temperatures became erratic concurrent with EGT sensor failures. The EGT failure affected the common Designated Signal Conditioner (DSC) power supply isolation card. The injector temperatures and EGTs utilize a common DSC. Analysis indicated that a momentary EGT short-to-ground may have provided a ground loop between the injector temperature and common power supply resulting in erratic injector temperatures. There was no problem with the injector tube temperature measurements; only the EGT measurements were affected. No action was taken by the crew. Injector temperature measurements remained functional. Troubleshooting will be performed.
6	Right vent door 1, 2 purge position failure. IFA No. STS-38-06 HR No. ORBI-178A {AR}	During postlanding vent door purge positioning operation, the right vent doors #1 and #2 drove to the "closed" position instead of the "purge" position. The right vents #1 and #2 are used to purge the forward Reaction Control System (RCS). STS-38 purge could not be performed via the right vent #1 and #2. This failure may be a limit switch/contact problem. There are 2 limit switches for door "open" position, 2 limit switches for door "close" position, and 2 limit switches for "purge" position. Limit switch failure is Crit 1R4. The worst-case failure for the vent doors is Crit 1R2. Failure analysis is continuing.

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	Thruster R1U showed low Chamber Pressure (P_c). IFA No. STS-38-07 HR No. ORBI-056 {C}	Thruster R1U showed degraded P_c by approximately 20 pounds per square inch absolute (psia) during reentry. The normal operating pressure is 150 psia. Thruster R1U worked properly. R1U was fired normally but was lowered to last priority. Thruster R1U was not deselected. For previous low P_c failure, the thruster failed "off" when the thruster indicated approximately 10 psia degradation. In addition, 3 other thrusters, R3D, RF3L, and R4U, showed low P_c indications. It is believed that a partially clogged P_c sensor tube or pressure transducer caused the degradation. Visual inspection of R1U revealed no anomalous condition. Chamber decay tests performed on November 27, 1990 found a 6 psi to 8 psi pressure decay in a 2-hr period. Post-decay test mass spec identified no leakage. The P_c transducer will be calibrated following hypergolic deservicing. Troubleshooting of thrusters R1U, R3D, RF3L, and R4U degradation is continuing.
8	Continuous "tire press" Fault Detection and Annunciator (FDA) message following landing gear safing. IFA No. STS-38-08 HR No. ORBI-018 {AR}	Continuous "tire press" messages were observed following landing gear safing procedures. During this procedure, one set of messages is expected after removal of the landing gear "arm" flag; however, continuous messages were noted. Initial visual inspection of the Tire Pressure Monitoring System (TPMS) showed no abnormalities. Troubleshooting will be performed.

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	Transient smoke detector event indication anomaly. IFA No. STS-38-09 HR No. ORBI-300 {C}	Smoke detector "3A" in avionics bay 3A did not have enough voltage to ring the alarm, but the event indicators (lights) lit. There were several previous instances of smoke detector failures where no apparent cause could be found. This occurrence was similar to the problem experienced on STS-41. On STS-32, avionics bay "3A" smoke detector "3A" experienced repeated transient alarms and associated lights. A decision was made to pull the sensor circuit breaker to avoid nuisance alarms during sleep, reentry, and landing periods. The sensor was removed and replaced. The defective unit was sent to the vendor for failure analysis but the unit was damaged before the vendor could examine it. STS-38 smoke detector "3A" will be removed, and failure analysis will be performed in conjunction with failure analysis of the STS-32 smoke detector anomaly.

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Unidentified debris observed between Mission Elapsed Time (MET) 26 sec to 40 sec from base region of both Solid Rocket Boosters (SRBs). IFA No. STS-38-B-01 HR No. B-60-05 Rev. C-DCN4 {C}	A review of launch films revealed debris exiting from both the Left-Hand (LH) and Right-Hand (RH) SRB base regions during the MET 26- to 40-sec period. Four pieces, estimated at 38" x 18", were observed from the LH SRB, and 1 piece, estimated at 44" x 18", was observed from the RH SRB. There were 3 potential debris sources: the viton-coated nylon, the thermal curtain layers, and the aluminum glass laminate tape. A review of build and installation papers found no Problem Reports (PRs) or hardware discrepancies. A potential SRB debris source was the viton-coated nylon on the thermal curtain (the outermost layer of the thermal curtain). This layer is designed to burn away during ascent; however, no definitive data on burn patterns were analyzed. The loss of large pieces of this nylon is not considered an anomalous condition because the nylon had adequately performed its function. Another potential debris source was the loss of thermal curtain layers. Postflight assessment of the aft skirt performance revealed normal hardware condition with no indications of thermal curtain loss. Thermal curtain loss would produce severe localized ablation; there were no indications of thermal curtain decomposition on STS-38 aft skirts. Also, thermal curtain loss in the projected Thrust Vector Control (TVC) area would have produced hydrazine detonation. The most probable SRB debris source was the 6"-wide aluminum glass laminate tape that is used to keep the purge inside the aft skirt. When the tape is applied, a new layer of the tape is lapped over the previous layer. During ascent, the tape loses adhesiveness and, by design, the tape is expended during ascent. This condition was observed on previous flights.

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
2	Right SRB External Tank Attachment (ETA) ring missing Instafoam at forward face. IFA No. STS-38-B-02 HR No. INTG-037B {AR}	Instafoam was missing from 2 locations on the forward side of the ETA ring at voids created during Instafoam application. There was a 10" x 12" piece missing from the forward face of the right SRB ETA ring near the diagonal strut. There was also a 4" x 6" piece of foam missing above the Integrated Electronic Assembly (IEA). Instafoam is not required for thermal protection but is used to eliminate water collection. The voids in the Instafoam were introduced during the 2-step installation processing and were considered the cause of this anomaly. Foam was layed up too thick causing higher exothermic heating, thus creating voids with the applied layers. Confusion in interpretation of design and application documentation led to the thick Instafoam layer. Design and installation documentation will be changed to clarify the procedures. Postflight assessment found no heat effects on the exposed surfaces of the missing Instafoam attributed to ascent thermal environment. Instafoam loss is not predicted during ascent due to aerodynamic loads. The lack of sooting and charring at the bottom of the voids indicated descent damage. In addition, debris damage was considered unlikely due to the location and density of the foam (3 - 4 lb/ft^3).

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
MOD		
1	Unexpected General Purpose Computer (GPC) #3 talkback indication. IFA No. STS-38-MOD-01 HR No. ORBI-066 {AR}	During the normal post-insertion freeze-dry procedure, the crew moded GPC #3 from "run" to "stby" to "halt". However, an unexpected run talkback indication was encountered during the deorbit preparation procedure. A GPC #3 memory dump was performed on GNC OPS3 software. The data indicated that GPC #3 was not allowed to complete standby processing before it was moded. The data also indicated that there were no software or hardware problems. The crew believed that they had performed the function correctly in accordance with training.

SECTION 8

BACKGROUND INFORMATION

This section contains pertinent background information on the safety risk factors and anomalies addressed in Sections 3 through 6. It is intended as a supplement to provide more detailed data if required. This section is available upon request.

APPENDIX A

LIST OF ACRONYMS

AC	Alternating Current
ADI	Attitude Direction Indicator
AFB	Air Force Base
AOS	Acquisition of Signal
APU	Auxiliary Power Unit
AR	Accepted Risk
ATP	Acceptance Test Procedure
BB	Barrier-Booster
BFS	Backup Flight Software
BITE	Built-In Test Equipment
C	Controlled
CA	California
CB	Circuit Breaker
CEI	Contract End Item
CRES	Corrosion Resistant Steel
Crit	Criticality
CRT	Cathode Ray Tube
CVFA	Check Valve Filter Assembly
DEU	Data Entry Unit Display Electronic Units
DFRC	Dryden Flight Research Center
DM	Development Motor
DoD	Department of Defense
DPS	Data Processing Software System
DSC	Designated Signal Conditioner
DU	Display Unit
EAFB	Edwards Air Force Base
ECP	Engineering Change Proposal
EGT	Exhaust Gas Temperature
EIU	Engine Interface Unit
EPD&C	Electrical Power Distribution and Control
EST	Eastern Standard Time

APPENDIX A

LIST OF ACRONYMS - CONTINUED

ET	External Tank
ETA	External Tank Attach External Tank Attachment
F	Fahrenheit
FASCOS	Flight Acceleration Safety Cutoff System
FC	Fuel Cell
FCHL	Flight Control Hydraulics Laboratory
FCL	Freon Coolant Loop
FCS	Flight Control System
FCV	Flow Control Valve
FD	Flight Day
FDA	Fault Detection and Annunciator
FES	Flash Evaporator System Flash Evaporator Subsystem
FID	Fault Identification
FMEA/CIL	Failure Modes and Effects Analysis/Critical Items List
FOS	Factor of Safety
FP	Fuel Pump
FPB	Fuel Preburner
FRR	Flight Readiness Review
FSM	Flight Support Motor
ft	Feet
g	Gravitational Acceleration
GG	Gas Generator
GGVM	Gas Generator Valve Module
GH_2	Gaseous Hydrogen
GN_2	Gaseous Nitrogen
GO_2	Gaseous Oxygen
GOAL	Ground Operations Aerospace Language
GOX	Gaseous Oxygen
GPC	General Purpose Computer
H_2	Hydrogen
HCF	High-Cycle Fatigue
HDP	Holddown Post
HGDS	Hazardous Gas Detection System
HPFTP	High-Pressure Fuel Turbopump
HPOTP	High-Pressure Oxidizer Turbopump
HPU	Hydraulic Power Units
HR	Hazard Report

APPENDIX A

LIST OF ACRONYMS - CONTINUED

hr	Hour
Hz	Hertz
ICHR	Integrated Cargo Hazard Report
ID	Inside Diameter
IEA	Integrated Electronic Assembly
IFA	Inflight Anomaly
IMU	Inertial Measurement Unit
in-lb	Inch-Pound
IUS	Inertial Upper Stage
JSC	Johnson Space Center
KSC	Kennedy Space Center
L-2	Launch Minus 2 Day
lb	Pound
lbf	Pounds Force
LCC	Launch Commit Criteria
LCF	Low-Cycle Fatigue
LCN	Logic Change Notice
LD	Leak Detector
LH	Left-Hand
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LOX	Liquid Oxygen
LPF	Low-Pressure Fuel
LPFTP	Low-Pressure Fuel Turbopump
LPOTP	Low-Pressure Oxidizer Turbopump
LPS	Launch Process Sequencer
	Launch Processing Set
LSFR	Launch Site Flow Review
LVDT	Linear Variable Differential Transducer
MCC	Main Combustion Chamber
	Mission Control Center
MCF	Major Component Failure
ME	Main Engine
MECO	Main Engine Cutoff
MET	Mission Elapsed Time
min	Minute
MLP	Mobile Launch Platform

APPENDIX A

LIST OF ACRONYMS - CONTINUED

MMT	Mission Management Team
MPS	Main Propulsion System
MRB	Material Review Board
MS	Mission Specialist
MSE	Mission Safety Evaluation
MSFC	Marshall Space Flight Center
N_2	Nitrogen
N_2O_4	Nitrogen Tetroxide
NASA	National Aeronautics and Space Administration
NBAT	Nominal Bus Assignment Table
NLG	Nose Landing Gear
NSI	NASA Standard Initiator
NSRS	NASA Safety Reporting System
O_2	Oxygen
OD	Outside Diameter
OMRSD	Operational Maintenance Requirements and Specifications Document
OMS	Orbital Maneuvering System
OPB	Oxidizer Preburner
OPF	Orbiter Processing Facility
OPO	Orbiter Project Office
OSMQ	Office of Safety and Mission Quality
OV	Orbiter
P/N	Part Number
PAR	Prelaunch Assessment Review
P_c	Chamber Pressure
PCV	Pulse Control Valve
PLI	Preload Indicating
POR	Power-On Reset
ppm	Parts Per Million
PR	Problem Report
PRCB	Program Requirements Control Board
PRCBD	Program Requirements Control Board Document
psi	Pounds Per Square Inch
psia	Pounds Per Square Inch Absolute
psig	Pounds Per Square Inch Gage
PSN	Purge Sequence Number
QC	Quality Control
QD	Quick Disconnect

APPENDIX A

LIST OF ACRONYMS - CONTINUED

QM	Qualifiction Motor
RCS	Reaction Control System
RH	Right-Hand
RHC	Rotational Hand Controller
RI	Rockwell International
RM	Redundancy Management
RMS	Redundancy Management System
rpm	Revolutions Per Minute
RSRM	Redesigned Solid Rocket Motor
RTLS	Return-To-Launch Site
S/N	Serial Number
SAIL	Shuttle Avionics Integration Laboratory
sccs	Standard Cubic Centimeters Per Second
scim	Standard Cubic Inch Per Minute
sec	Second
SEM	Scanning Electron Microscope
SII	Solid Rocket Motor Igniter Initiator
SIP	Strain Isolation Pad
SLF	Shuttle Landing Facility
SM	System Management
SOV	Shutoff Valve
SR&QA	Safety, Reliability, and Quality Assurance
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSC	Stennis Space Center
SSME	Space Shuttle Main Engine
SSRP	System Safety Review Panel
TEM	Test Evaluation Motor
TFL	Telemetry Format Load
TPMS	Tire Pressure Monitoring System
TPS	Thermal Protection System
TPTA	Transient Pressure Test Article
TVC	Thrust Vector Control
USBI	United Space Boosters, Inc.

APPENDIX A

LIST OF ACRONYMS - CONTINUED

VAC	Volts Alternating Current
WSB	Water Spray Boiler
WWMS	Waste Water Management System