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MISSION SAFETY EVALUATION REPORT FOR STS-35

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-35

Postflight Edition: March 15, 1991

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EXECUTIVE SUMMARY

After many months delay due to Liquid Hydrogen (LH_2) leaks in the aft compartment and 17" LH_2 umbilical disconnect system, *Columbia* was launched from Kennedy Space Center (KSC) launch complex 39B at 1:49 a.m. Eastern Standard Time (EST) on December 2, 1990. Four launch scrubs, and a number of mini-tanking tests performed to troubleshoot the Hydrogen (H_2) leaks, took place between the initial launch attempt in May of 1990 and the successful tanking test on October 30, 1990 that cleared *Columbia* for the December 2, 1990 launch. Both the Orbiter and External Tank (ET) 17" disconnects were replaced with new units prior to the final tanking test and the December launch. It is believed that crushed prevalve detent cover seals were the primary leak path that resulted in the high aft compartment H_2 concentrations observed during the scrubbed launch attempts and the tanking tests.

Planned mission duration was shortened one day due to forecasted inclement weather at the primary landing site, Edwards Air Force Base (EAFB), California. Deorbit took place on orbit 141, with a landing on EAFB concrete runway 22 at 12:54 a.m. EST on December 10, 1990.

Based on the limited number and type of Inflight Anomalies (IFAs) encountered, the Space Shuttle operated satisfactorily throughout the STS-35 mission. However, only about 75% of the mission's planned scientific objectives were accomplished due to spacelab computer equipment failures.

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-35 flight were also documented prior to the STS-35 Flight Readiness Review (FRR) (FRR Edition) and the STS-35 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.

TABLE OF CONTENTS

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-35 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-35 mission, including launch data, crew size, mission duration, launch and landing sites, and other mission- and payload-related information.

Section 3 - Contains a list of safety risk factors/issues, considered resolved or not a safety concern prior to STS-35 launch, that were impacted or repeated by anomalies reported for the STS-35 flight.

Section 4 - Contains a list of safety risk factors that were considered resolved for STS-35.

Section 5 - Contains a list of Inflight Anomalies (IFAs) that developed during the STS-38 mission, the previous Space Shuttle flight.

Section 6 - Contains a list of IFAs that developed during the STS-32 mission, the previous flight of the Orbiter vehicle (OV-102).

Section 7 - Contains a list of IFAs that developed during the STS-35 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-35 MISSION SUMMARY

2.1 Summary Description of the STS-35 Mission

After many months delay due to Liquid Hydrogen (LH_2) leakage in the aft compartment and 17" LH_2 umbilical disconnect system, *Columbia* was launched from Kennedy Space Center (KSC) launch complex 39B at 1:49 a.m. Eastern Standard Time (EST) on December 2, 1990. Four launch scrubs, and a number of mini-tanking tests performed to troubleshoot the Hydrogen (H_2) leaks, took place between the initial launch attempt in May of 1990 and the successful tanking test on October 30, 1990 that cleared *Columbia* for the December 2, 1990 launch. Both the Orbiter and External Tank (ET) 17" disconnects were replaced with new units prior to the final tanking test and the December launch. Section 4, Integration 8 contains a detailed description of the H_2 leakage encountered during the launch scrubs and tanking tests, and the investigations conducted to isolation the cause and to successfully resolve the H_2 leaks. It is believed that crushed prevalve detent cover seals were the primary leak path that resulted in the high aft compartment H_2 concentrations observed during the launch attempts and tanking tests.

Postlaunch inspection of the Mobile Launch Platform (MLP) and the pad found no flight hardware or Thermal Protection System (TPS) materials, with the exception of one Orbiter base shield Q-felt plug. These plugs had also been found on previous flights. South Solid Rocket Booster (SRB) Holddown Post (HDP) erosion was normal. Shim material on HDP #1, #2, and #6 was intact, but debonded to various degrees. Shim material on HDP #5 was also debonded and partially missing from 2 sidewalls. A total of 5 pieces of Instafoam were recovered; the largest piece measured 4" x 3" x 2". In addition, an 8" x 10" x 4" curved section of scorched foam was found on the pad apron.

Based on the limited number and type of Inflight Anomalies (IFAs) encountered, the Space Shuttle operated satisfactorily throughout the STS-35 mission. The more significant anomalies observed are summarized below.

During egress and prelaunch operations, the pilot attempted to make seat adjustments. The pilot seat failed to drive down. (This is a repeat of an STS-32 anomaly and has been a unique OV-102 problem.) However, the seat worked properly on orbit. The down-limit switch was replaced during STS-35 turnaround, and troubleshooting will be performed at KSC.

During ascent, Water Spray Boiler (WSB) #3A did not initiate spray cooling until Auxiliary Power Unit (APU) #3 lube oil return temperature reached 277°F. WSB cooling operations should begin at 250°F. Preliminary analysis indicated the presence of wax in the APU #3 lube oil that may have caused the spray bar to freeze. A hot oil flush will be performed during the STS-40/OV-102 turnaround process, and WSB #3A will be tested. During both ascent and entry, indication of a large amount of wax was also noted in the APU #2 lube oil system. APU #2 is scheduled to be replaced during STS-40/OV-102 turnaround due to the Gas Generator Valve Module (GGVM) life-limit criteria.

About 9 hours into the flight, the astronauts smelled a burning odor coming from 1 of the 2 Spacelab control Dedicated Display Units (DDUs) on the flight deck. The unit was promptly shutdown, and the backup unit was used. However, on the morning of December 6, the second Spacelab DDU shut itself off, and a burning odor was again noticed. Later attempts to reactivate the primary DDU were unsuccessful. The crew's only means of communication with the Astro-1 payload was by the DDUs. A workaround was devised whereby the astronauts used their joystick to perform final targeting while the Huntsville scientists coached the astronauts via radio to refine the Astro-1 instruments alignment on the target. The procedure was cumbersome, but it enabled the acquisition of an appreciable amount of important scientific data.

On orbit, gradual degradation of the waste water dump rate was noted during the first 3 dump cycles, and the line was completely blocked on the fourth dump. Troubleshooting found that the waste water dump line polyurethane filter material had deteriorated due to age and caused the blockage. The filters were manufactured in 1980 and were found still in stock. The polyurethane material deteriorates badly after approximately 8 years.

The planned mission duration was shortened one day due to forecasted inclement weather at the primary landing site, Edwards Air Force Base (EAFB), California. Deorbit took place on orbit 141, with a landing on EAFB concrete runway 22 at 12:54 a.m. EST on December 10, 1990. Only about 75% of the mission's planned scientific objectives were accomplished due to Spacelab computer equipment failures.

Postflight inspection of OV-102 found a 24" piece of the environmental material hanging loose at the top of the right Payload Bay Door (PLBD). No damage internal to the payload bay was apparent. An investigation into the cause of this STS-35/OV-102 anomaly is continuing. On STS-41/OV-103, a 6" splice segment of the PLBD-to-aft bulkhead environmental seal was found debonded; the cause was determined to be insufficient application of the seal (bad etching and bonding).

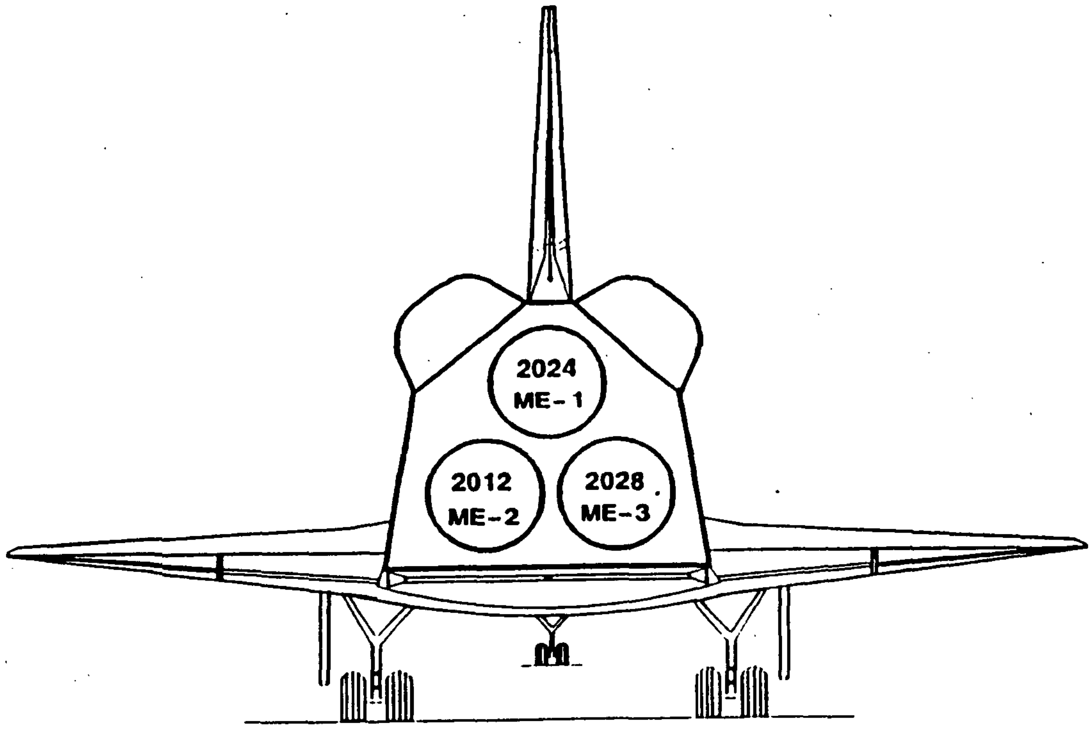
The Orbiter/ET Liquid Oxygen (LO_2) aft attach/separation hole plugger did not complete its stroke. One of the 2 pyros was jammed between the plugger and the rim of the hole. The other pyro device was not found and may have escaped. The ET door latched normally, and no debris was found on the runway after the ET doors were open. The ET doors may be recycled in flight in the event that closing or latching is obstructed. Similar hole plugger failures occurred on STS-29 and STS-34.

During postflight inspection of the left Solid Rocket Motor (SRM) nozzle joint #3, a 1.5" gas path was observed through a Room Temperature Vulcanizing (RTV) void at 195° of the Carbon Cloth Phenolic (CCP). Surface heating effects and sooting resulted. However, there was no blowby erosion or heat effects to the primary O-ring, and no metal nozzle components were affected. The RTV contributes only as a thermal barrier and is not considered a seal in the nozzle joints. This was the first occurrence of heat-affected CCP in nozzle joint #3. However, similar effects were seen in joint #2 (nose inlet bearing/cowl) of STS-36, QM-7, and PVM-1; no O-ring heat effects resulted. Thiokol plans to conduct an aero/thermal analysis of nozzle joint #3 to determine the gas volume and flow characteristics associated with this STS-35 anomaly.

During review of STS-35 ET photographs taken after separation, 11 TPS divots from 7" to 10" in diameter were observed. All were located in the intertank-to-hydrogen flange. The divots seen on STS-35 were the first experienced since STS-28. Photograph review by the Martin Marietta Corporation (MMC) determined that the divots did not expose ET metal. Worst-case analysis performed by MMC indicated that bare tank metal as seen on STS-35 would not result in structural or thermal problems.

2.2 Flight/Vehicle Data

- Launch Date: December 2, 1990
- Launch Time: 1:49 a.m. EST
- Launch Site: KSC Pad 39B
- RTLS: Kennedy Space Center, Shuttle Landing Facility
- TAL Site: Banjul, The Gambia
- Alternate TAL Site: Ben Guerir, Morocco
- Inclination: 28.45 Degrees
- Altitude: 190 Nautical Miles, Circular Orbit
- Landing Site: Edwards AFB, CA, Concrete Runway 22
- Landing Date: December 10, 1990
- Landing Time: 12:54 a.m. EST
- Mission Duration: 8 Days, 23 Hours, 5 Minutes
- Crew Size: 7
- Orbiter: OV-102 (11) *Columbia*
- PRSD Tank Sets: 5 (Extended Duration Orbiter)
- SSMEs: (1) #2024, (2) #2012, (3) #2028
- ET: ET-35
- SRBs: BI-038
- SRMs: RSRM Flight Set #11
- MLP: MLP #3



ENGINE	#2024	#2012	#2028
POWERHEAD	#2026	#2025	#4005
MCC*	#2013	#2020	#2018
NOZZLE	#4006	#4002	#4012
CONTROLLER	F5	F16	F21
FASCOS*	#01	#09	#21
HPFTP*	#4008	#4007R3	#6009
LPFTP*	#2131	#4006	#2026R2
HPOTP*	#4205R3	#6003R3	#4502R1
LPOTP*	#2106R1	#4306	#2027R1

* Acronyms can be found in Appendix A.

2.3 Partially Fixed Orifice Flow Control Valves

STS-35/OV-102 was the third Space Shuttle to fly 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 was 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-35 by replacing the spherical spring sensing assembly. This reduced the minimum ullage pressure from 16.0 psig to 14.2 psig, reducing 4 Crit 1R failure modes to Crit 3, and eliminating the effect of 15 other failure modes. The 31-psig GO_2 vent/relief valve was test qualified. The 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%	78%

The first use of shimmed FCVs flew on STS-41/OV-103. Postflight data analysis indicated that the partially fixed orifices worked well, with no anomalies.

STS-35 was the third of the series of flights with partially fixed orifices that will lead to implementation of a fixed orifice FCV. OV-105 hardware shimmed to the Step 1 configuration was installed on OV-102 for STS-35. 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 Landing Site Priorities

The Astro-1 and Broad Band X-Ray Telescope (BBXRT) were not deployed from the Payload Bay; they remained with and returned to Earth with the Orbiter vehicle. Nominal End-of-Mission (NEOM) landing weight for STS-35 was 226,461 pounds (lb); the Center of Gravity (CG) was 1081.1". The landing weight and CG for STS-35 Abort Once Around (AOA) were 229,901 lb and 1079.3", respectively. Landing on the Edwards Air Force Base (EAFB) lakebed under heavy weight/forward CG conditions can result in nose gear design loads being exceeded. For both STS-32 and STS-31 (heavy weight, forward CG missions), the runway priorities were changed to place EAFB concrete runway ahead of the lakebed.

Due to the high derotation rates for landing on the EAFB lakebed, calculated STS-35 Nose Landing Gear (NLG) slapdown loads for a lakebed landing exceeded the nose gear certified design load limit of 90,000 lb. For a nose gear load above 90,000 lb, there is a possibility of structural damage and a requirement for a detailed postflight landing gear inspection. Major structural damage to the Orbiter can be expected if the NLG load exceeds 111,000 lb. Therefore, the following landing site priorities were established for the STS-35 mission.

1. EAFB concrete
2. KSC
3. Northrup
4. EAFB lakebed.

2.5 17" Disconnect Hydrogen Purge.

A high flow rate Gaseous Nitrogen (GN_2) system was under consideration for use on STS-35 to purge Hydrogen (H_2) leakage from the 17" disconnect area. An initial design was presented at a Final Design Review meeting at Kennedy Space Center (KSC) on August 20, 1990. There were many unanswered safety-related questions (e.g. acoustic levels, vibration characteristics, foreign object damage, oxygen concentration effects, welding at the pad, etc.). Headquarters Safety, in looking at the proposed purge system, recognized the potential utility but believed that implementation should not be rushed due to the many safety-related questions and the need for satisfactory review by the responsible Space Shuttle elements. Subsequent to the STS-35 Delta Flight Readiness Review on August 20, 1990, the Space Shuttle Program decided not to implement this purge system for the STS-35 launch due to the extremely tight schedule conditions. The purge system will be available for use on subsequent Shuttle launches, starting with the next flight - STS-39. Details of the design and application of this GN_2 purge system will be addressed in the STS-39 Mission Safety Evaluation (MSE) Report.

2.6 External Tank Tanking Lightning Constraint

A change to the electrical storm activity constraint for ET tanking was in effect for STS-35. For the previous flight, STS-38, the requirement was verification that no more than a 20% probability existed for potential electrical activity within 5 miles of the launch pad during the first 4 hours (hr) of ET tanking. For STS-35, the first 4 hr of ET tanking condition was reduced to the first hour of ET tanking. This change was promulgated by Requirements Change Notice (RCN) S59684 approved by the August 2, 1990 Program Requirements Control Board (PRCB).

The rationale behind this change was:

- Present pad lightning protection is adequate:
  - Lightning masts (pad and ET)
  - Catenary system
  - Ground systems incorporate protection.
- Prediction accuracy has been improved for the first hour of tanking.

This change will provide increased assurance that tanking can be accomplished effectively during summer months.

2.7 Payload Data

Payload Bay:

- Astro-1 Observatory
  - Hopkins Ultraviolet Telescope (HUT)
  - Ultraviolet Imaging Telescope (UIT)
  - Wisconsin Ultraviolet Photo-Polarimeter Experiment (WUPPE)
- Broad Band X-Ray Telescope (BBXRT)

Middeck:

- Shuttle Amateur Radio Experiment (SAREX)
- Air Force Maui Optical Site (AMOS) Calibration Test

2.8 Astro-1 Description.

The Astro-1 mission objective was to conduct a coordinated study of a large variety of astrophysically important celestial sources in the ultraviolet and x-ray wavelength regions. Astro-1 was the first of a series of missions that will make precise measurements of objects such as planets, stars, and galaxies in relatively small fields of view. Observations emphasized hot and energetic sources which are some of the most dynamic objects in the universe. A prime target was supernova SN1987A.

The Astro-1 mission operated a complement of 3 experiments and their associated equipment (Astro-Spacelab) in addition to a related but unique experiment, the Broad Band X-Ray Telescope (BBXRT), to make precise measurements of astronomical objects for obtaining scientific data. The 3 Astro-1 Spacelab experiments were the Wisconsin Ultraviolet Photo-Polarimeter Experiment (WUPPE), the Ultraviolet Imaging Telescope (UIT), and the Hopkins Ultraviolet Telescope (HUT). The Astro-1 Spacelab payload utilized the Spacelab Instrument Pointing System (IPS) that was installed on a Spacelab 2-pallet train configuration in the payload bay. The BBXRT was mounted on a special carrier that was integrated directly into the Orbiter payload bay, and utilized the Two-Axis Pointing System (TAPS). The integrated payload consisted of Mission-Peculiar Equipment (MPE), Spacelab equipment, and the 4 experiments.

The Spacelab experiments and supporting equipment were attached to the IPS through a cruciform to enable pointing and alignment for stellar observations. The star tracker, which supported the IPS, was also mounted on the cruciform. MPE required to integrate, support, and align these 3 experiments included the cruciform, a Payload Support Strut Assembly (PSSA), an Integrated Radiator System (IRS), Multi-layer Insulation (MLI), an Image Motion Compensation System (IMCS), instrument mounting provisions on the cruciform, electrical cables, and other equipment. Spacelab-provided equipment consisted of the IPS with Optical Sensor Package (OSP), pallet structure and pallet-mounted equipment, the Payload Retention Latch Assembly (PRLA), actuators and controls, Igloo structure and Igloo-mounted equipment, Aft Flight Deck (AFD)-mounted equipment, Subsystem Mounting Structure, an Environmental Control Subsystem, an Electrical Power Distribution System (EPDS), and a Command and Data Management System (CDMS).

The purpose of the 4 Astro-1 mission experiments is briefly described below.

- **HUT** studied faint astronomical objects such as quasars, active galactic nuclei, and supernova remnants in the little-explored ultraviolet range below 1200 Angstroms.

- **WUPPE** measured the strength and polarization of ultraviolet light from celestial objects in our galaxy and beyond, such as hot stars, galactic nuclei, and quasars. Polarized light provides information about the physical conditions at the radiation source and the interstellar dust through which it travels enroute to Earth.

- **UIT** acquired images of faint objects in broad ultraviolet bands in the wavelength range from 1200 to 3200 Angstroms. It also investigated the present stellar content and history of star formation in the galaxies, the nature of spiral structures, and non-thermal sources in galaxies.

- **BBXRT** studied various targets, including active galaxies, clusters of galaxies, supernova remnants, and stars. It directly measured the amount of energy in electron-volts of each X-ray detected. BBXRT was a latecomer to the Astro-1 mission. BBXRT was added to the STS-35 payload after a supernova erupted within the large Magellanic Cloud in February 1987 to identify the chemical elements created by this explosion and to provide a better understanding of the processes that create so-called "soft" x-rays.

SECTION 3

SAFETY RISK FACTORS/ISSUES IMPACTED BY STS-35 ANOMALIES

This section lists safety risk factors/issues, considered resolved (or not a safety concern) for STS-35 prior to launch (see Sections 4, 5, and 6), that were repeated or related to anomalies that occurred during the STS-35 flight (see Section 7). The list indicates the section of this Mission Safety Evaluation (MSE) Report in which the item is addressed, the item designation (Element/Number) within that section, a description of the item, and brief comments concerning the anomalous condition that was reported.

	ITEM	COMMENT

<u>Section 4: Resolved STS-35 Safety Risk Factors</u>

SRM 3	STS-31 right Solid Rocket Motor (SRM) igniter adapter-to-forward dome joint putty blowhole.	A single blowhole through the igniter joint putty was experienced on both STS-35 SRMs, and there was damage to the cadmium plating and sooting. There was no damage to the Gask-O-Seal elastomer. Because this was an expected event, no Inflight Anomaly was assigned. A blowhole was found in the STS-31 right SRM adapter-to-forward dome (outer) joint putty, with no soot past the seals. Soot was noted on the outer gasket retainer Inside Diameter (ID) edge. 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 on the igniter inner gasket retainer aft face and OD edge. Minor pitting with a maximum depth of 2 mils was also observed. A blowhole was also found on the STS-31 left SRM. Putty blowholes were also experienced on STS-41 and STS-38 SRM igniter joints. These occurrences were within the SRM experience base. It is believed that the change in the putty layup procedure, and reduction of putty used to alleviate the problem, led to the increased incidence of blowholes. A redesign effort on the igniter-to-dome joint is in work to delete the joint putty. An investigation team is working on changing the gasket retainer material from cadmium-plated steel to stainless steel. The SRM Project Office is investigating a design change to remove cadmium plating from the Gask-O-Seal.

ITEM		COMMENT
Section 4: Resolved STS-35 Safety Risk Factors		
SRM 6	Aft dome factory joint internal insulation voids.	Aft dome factory joint internal insulation verification was performed on all aft SRM segments using ultrasonic inspection techniques. On the STS-40 Right-Hand (RH) aft segment, ultrasonic inspection identified insulation voids. The aft dome factory joint was x-rayed; 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. Because of these findings, aft dome factory joints from STS-36 and STS-31 SRMs were dissected. Similar insulation voids were identified. It was determined, however, that these SRMs, and others based on visual postflight inspections, met the erosion safety factor. Postfire insulation samples were taken from all SRMs. These samples had small, entrapped air voids. STS-35 SRM aft dome factory joints have not yet been dissected to inspect for insulation voids. A terminated blowhole was found in the polysulfide insulation on the RH nozzle-to-case joint. This was the first occurrence in a flight motor but was not unexpected, because of ground test experience.

ITEM		COMMENT
Section 5: STS-38 Inflight Anomalies		
Orbiter 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	WSB #3 did not initiate cooling of APU lube oil at the required temperature on STS-35. This anomaly was attributed to the presence of wax in the lube oil (instead of a problem with the APU controller as experienced on STS-38). (See Section 7, Orbiter 5 for further details.) (IFA No. STS-35-17)
Orbiter 7	Thruster R1U showed low Chamber Pressure (P_c). IFA No. STS-38-07	Thruster R5D failed "off" indicating low P_c on STS-35, and was deselected by Redundancy Management (RM). Investigation by the crew determined that helium was present in the crossfeed line of R5D. This condition is indicative of bubbles in the feedline. Manual firing of the thruster restored R5D to operational status. (See Section 7, Orbiter 8 for further details.) (IFA No. STS-35-20)

	ITEM	COMMENT
Section 6: STS-32 Inflight Anomalies		
Orbiter 1	APU #3 lubrication oil outlet pressure high. IFA No. STS-32-02	APUs #2 and #3 had indications of wax in the lubrication oil on STS-35. APU #2 will be removed and replaced due to life-limit constraints. APU #3 as well as WSB #2 and #3 will require a hot oil flush to remove the wax. (See Section 7, Orbiter 5 and 7 for further details.) (IFA No. STS-35-17 and STS-35-19)
Orbiter 4 and Orbiter 5	Humidity separators "B" and "A" water bypass. IFA No. STS-32-07A STS-32-07B	A small amount of water was found near the air outlet of humidity separator "B" on STS-35 Flight Day (FD) 6. It was decided to allow the water to air dry. Followup inspections found no additional water for the remainder of the mission.
Orbiter 11	Waste water dump line/nozzle blockage. IFA No. STS-32-21	The waste water dump function on STS-35 degraded to the point of failure. Repeated attempts to clear the line were unsuccessful. Alternate methods for storing waste water were used in order to continue the mission. Waste tank offload into the Contingency Water Container (CWC) and urine collection devices was required for the remainder of the mission. Troubleshooting found that the waste water dump line filter had deteriorated and was the root cause of the line blockage. A decision was made to manifest additional CWCs on all flights. (See Section 7, Orbiter 2 for further details.) (IFA No. STS-35-05)

	ITEM	COMMENT

<u>Section 6: STS-32 Inflight Anomalies</u>

	ITEM	COMMENT
Orbiter 13	WSB #3 controller "A" overcooling. IFA No. STS-32-23	WSB #3 operations on controller "A" on STS-35 did not initiate lube oil cooling as required at 250°F. This problem was attributed to the presence of wax in the lube oil. During reentry operations, WSB #3, while under the control of controller "A", was reported to be overcooling the lube oil. (See Section 7, Orbiter 5 for further details.) (IFA No. STS-35-17)
Orbiter 15	RH stop bolt was found slightly deformed on the STS-32 centering ring of the forward External Tank (ET) attach/separation assembly. IFA No. STS-32-26	Postflight inspection of STS-35/OV-102 found the RH stop bolt bent approximately 5° from center. Damage was worse than that seen on STS-32 and STS-38, but not as bad as experienced on STS-34. The bolt was removed and sent to the vendor for analysis. (See Section 7, Orbiter 10 for further details.) (IFA No. STS-35-22)
Orbiter 16	Pilot seat down drive motor did not operate. IFA No. STS-32-27	The pilot seat down limit switch failed to allow the seat to be driven down on STS-35. The seat motor and limit switch will be investigated by the vendor. (See Section 7, Orbiter 11 for further details.) (IFA No. STS-35-23)

	ITEM	COMMENT

<u>Section 6: STS-32 Inflight Anomalies</u>

	ITEM	COMMENT
SSME 1	Main Combustion Chamber (MCC) aft end debond found on engine #2022. IFA No. STS-32-ME-01	Postflight ultrasonic inspection of STS-35 engine #2024, MCC #2013, found a debond less than 3/64" in diameter. This debond was considered to be within acceptable limits, and no action was taken.
ET 1	Review of the ET separation photos from STS-32 showed 4 Spray-On Foam Insulation (SOFI) divots in the bipod area. IFA No. STS-32-T-01	STS-35 ET separation photos illustrated 11 circular Thermal Protection System (TPS) divots in the intertank-to-hydrogen flange area. Divots ranged from 7" to 10" in diameter (estimates). (See Section 7, ET 1 for further details.) (IFA No. STS-35-T-01)
MCC 1	State vector uplink incident. IFA No. STS-32-MOD-01	No uplink errors were reported on STS-35. There was, however, an error in a patch to the Backup Flight System (BFS) software prior to launch. Because of the relocation of STS-35 to pad B for launch, launch pad locational data was required to be updated in both the Primary Avionics Software System (PASS) and BFS. During ascent, a difference of 143 feet was witnessed in the positional data sent from the BFS and PASS. Post-ascent evaluation of telemetry data identified an error in the sixth digit of the longitude string of the BFS patch. (See Section 7, Integration 1 for further details.) (IFA No. STS-35-I-01)

SECTION 4

RESOLVED STS-35 SAFETY RISK FACTORS

This section contains a summary of the safety risk factors that were considered resolved for STS-35. 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 safety risk factor.

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 Baseline risk. The NASA safety community assessed the relative risk increase of each and determined that the associated increase was acceptable.

Integration 1	Difference between the Primary Avionics Software System and Backup Flight System could lead to recontact between the Orbiter and External Tank after separation.
Integration 7	Liquid Oxygen fill and drain valve closure in the event of power loss in terminal sequence.
Orbiter 9	OV-102 Engine Interface Unit Power-On Reset anomaly.
Orbiter 15	Auxiliary Power Unit Gas Generator Valve Module issue.
Orbiter 17	OV-102 Freon Coolant Loop flow rate degradation.
SSME 13	High-Pressure Fuel Turbopump Serial Number #4405R1 liftoff seal anomaly during a 10,000-second certification run.
SRM 3	STS-31 right Solid Rocket Motor igniter adapter-to-forward dome joint putty blowhole.

SECTION 4 INDEX

RESOLVED STS-35 SAFETY RISK FACTORS

SECTION 4 INDEX - CONTINUED

RESOLVED STS-35 SAFETY RISK FACTORS

RESOLVED STS-35 SAFETY RISK FACTORS

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1	Difference between the Primary Avionics Software System (PASS) and Backup Flight System (BFS) could lead to recontact between the Orbiter and External Tank (ET) after separation. HR No. INTG-138 {C} ***No Orbiter/ET separation problems were experienced on STS-35.***	Prior to the first STS-35 launch attempt, it was discovered that a difference existed between the PASS and BFS requirements which, under certain conditions, could lead to recontact of the Orbiter and ET at the forward attach point after separation. The PASS inhibits all rotational commands during the first 3 seconds (sec) of the ET separation maneuver; the BFS does not. In the case where the BFS is in control prior to the ET separation maneuver and a negative pitch rate command is issued as the -Z translation burn begins (ET separation maneuver), recontact is possible. The required pitch rate to overcome the -Z translation can be achieved by either 2 Reaction Control System (RCS) jets failing "on" after the negative pitch rate command is issued or by a transient induced by a Space Shuttle Main Engine (SSME) failure close to Main Engine Cutoff (MECO). The probability of either of these occurrences is extremely low. If a negative pitch rotational command is received as the -Z maneuver starts, the Orbiter nose will move toward the ET. The forward RCS would have no -Z jets firing, and recontact would occur at the forward strut. Damage on recontact would be minimal, if any, due to the slow closing rate. Positive pitch rate commands are not a problem, because the result would be to move the Orbiter away from the ET. Plans are to implement a fix in the BFS source code in the OI-21 build. Discussions among the technical community concluded that implementation of a crew workaround procedure or software fix prior to flight would result in a higher risk than flying "as is". A crew user note was prepared to caution against negative pitch commands during ET separation while in the BFS mode. Safety concurred with the assessment that there was a low probability of the required sequence of events occurring and agreed that nothing more should be done until OI-21 is available. A waiver was approved accepting this condition through STS-50, the first planned flight with OI-21.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1 (Continued)		Rationale for STS-35 flight was: • Low probability of occurrence. • If recontact should occur, damage would be minimal and would not result in loss of crew or vehicle. ***This risk factor was acceptable for STS-35.***
2	OV-102 Main Propulsion System (MPS) contamination. HR No. INTG-023 {AR} ***No engine problems attributed to this contamination issue were reported on STS-35.***	A blowing leak was discovered at MPS relief valve PV11 during STS-32/OV-102 postlanding inspection. The PV11 relief valve [Liquid Hydrogen (LH_2) outboard fill and drain valve] was found leaking, and metal particles (Corundum and Calcite) were found on the valve seat. PV11 was removed and replaced; however, during retest of the newly installed PV11 valve, the PV5 relief valve (SSME #2, engine #2101, LH_2 prevalve) was found leaking. Inspection revealed Corundum and Calcite contamination on the valve seat and housing. PV5 was removed and replaced. Concern over these contamination incidents prompted an inspection of the MPS including the LH_2 fill and drain system that was borescoped through PV11 (LH_2 8" disconnect). One particle was found and removed that was green in color and not similar to the general contamination. The PV11 interior flange seals, feedline bellows, and approximately 5 feet (ft) into the fill and drain line were inspected and cleaned. The topping valve PV13 outlet at the fill and drain line was inspected with the ball valve in the closed position. The recirculation pump package was removed and borescoped through the manifold and down through the inboard fill and drain line. The system was cleaned. It was not possible to access 3 ft of the line.

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)		The LH_2 prevalve system was inspected. Engines #2024, #2012, and #2028 prevalve screens were found to contain minor to heavy contamination levels. Predominant contaminants were Calcite and Corundum; there was also some 300-series stainless steel particulate. The system was vacuumed clean. All prevalve relief valves were removed and cleaned. The Liquid Oxygen (LO_2) prevalve screens were also inspected. The detect cover and rollers were removed, and the LO_2 prevalve screens were borescoped. Engines #2024 and #2012 were clean. One teflon particle was removed from engine #2028; 2 particles could not be removed. Approximately 90% of the Orbiter LH_2 system was inspected on a square inch basis, and most contamination particles were vacuumed/wiped clean. The few remaining particles that were inaccessible were estimated at 400 to 500 microns. The SSME can tolerate particles of less than 1000 microns; particles greater than 1000 microns are filtered by prevalve screening. The 17" manifold was borescoped and vacuumed. There were no anomalies in the Operational Maintenance Requirements and Specifications Document (OMRSD) checkout, either LO_2 or LH_2, other than PV5 and PV11 relief valves that were replaced. Mobile Launch Platform (MLP) #3 was recleaned and inspected. KSC reported no significant contamination (>1000 microns) remaining after cleaning. (MLPs #1 and #2 were also inspected and found acceptable for use.) The suspected source of the Orbiter contamination was in the MLP #3 downstream of the 70-micron filter. The SSME showed no contamination. Rationale for STS-35 flight was: • All known contamination was removed. • The Hydrogen (H_2) leak investigation uncovered no additional contamination. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
3	Solid Rocket Booster (SRB) Rate Gyro Assembly (RGA) workmanship and inspection discrepancies. HR No. INTG-144A {C} INTG-165 {C} B-50-18 Rev. C DCN-1 {C} ***No SRB or Orbiter RGA anomalies were reported on STS-35.***	NASA Shuttle Logistic Depot (NSLD) assumed the task of refurbishing SRB RGAs after every flight in early 1990. Previously, SRB RGAs were refurbished after each flight by the vendor. During receipt inspection at NSLD, 13 of 19 RGAs were found to exhibit various workmanship and quality inspection discrepancies. NSLD analysis of the discrepancies led to the conclusion that the condition occurred during the previous refurbishment and was not related to flight. This placed doubt on the integrity of the RGAs on the STS-31 and STS-35 SRBs. To date, there has been only 1 inflight failure of an SRB RGA; this occurred on STS-26. There is no record of an Orbiter RGA inflight failure. Discrepancies found by NSLD included bent terminals, missing screws, damaged and exposed wires, cracked or broken connectors, loose wire clippings, missing or incorrect part numbers, and miscellaneous contamination. All noted discrepancies were found by visual inspection; no failures were recorded that led to a discrepancy discovery. SRB RGAs are supplied by the Orbiter Project as Government Furnished Equipment (GFE) and are very similar to those used on the Orbiter. (The Orbiter RGAs also have a roll axis.) The Orbiter Project coordinated an investigation to determine how such obvious quality inspection discrepancies could have occurred. Teardown inspection and tests were performed on the discrepant RGAs at NSLD. Results of this investigation determined that none of the discrepancies would lead to an inflight failure. The SRB RGAs are Criticality 1R2; the Orbiter RGAs are Criticality 1R3. All of the SRB RGAs installed on STS-35 were new units that had never been flown, and all were processed through a complete Acceptance Test Program (ATP), including Acceptance Vibration Testing (AVT) and Acceptance Thermal Testing (ATT) at Northrop. The AVT limits are equal to the flight limits. The SRB RGAs are shipped to United Space Boosters, Inc. (USBI) where they go through another

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
3 (Continued)		ATP, minus AVT and ATT, prior to installation. After installation and stacking, the SRB RGAs go through checkout/testing prior to launch. The SRB RGAs are designed for zero-g (i.e., conformal coating is provided to protect the circuitry). The RGAs installed on the STS-35/OV-102 Orbiter [Serial Number (S/N) 14, 15, 16, and 17] had a proven flight record. Each had flown 9 times with no inflight anomalies. Additionally, the Orbiter RGAs are produced in a different manufacturing facility with different personnel and engineering. Rationale for STS-35 flight was: • All STS-35 SRB RGAs were new units that had never been flown. • All STS-35 SRB RGAs passed Acceptance Testing, including AVT and ATT. • All Orbiter RGAs had flown 9 times with no anomalies. ***This risk factor was resolved for STS-35.***
4	Orbiter/ET (OV-102/ET-35) misalignment issue. HR No. INTG-051B {C} INTG-052B {C} ***No anomalies or performance problems attributed to Orbiter/ET misalignment were reported on STS-35.***	Post-mate inspection of the STS-35 Orbiter/ET alignment found the Orbiter to be offset approximately 1.5" from the centerline of the ET in the +Y direction at the forward interface EO-1. This misalignment was witnessed after both STS-35 Orbiter/ET mates. No new mating procedures were imposed for the STS-35 mate, and no Operations and Maintenance Instruction (OMI) deviations were required to complete the mate. The concern was that this was the greatest misalignment between an Orbiter and ET recorded to date and the Orbiter/ET interface tolerances were exceeded. This condition could be an indication of structural failure that could lead to the loss of the Orbiter and crew. An investigation was

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
4 (Continued)		performed to determine the source (structural deformation, manufacturing flaw, procedural error, etc.) and impact of the misalignment. The results of this investigation are summarized below. Requirements imposed at the Orbiter-to-ET interface are derived from Interface Control Drawing (ICD) and OMRSD requirements. The ICD maximum interface tolerance requirement is 1.07"; this requirement was exceeded by over 0.4". The OMRSD specifies that the adjustable forward bipod strut is to show 1 to 36 threads after mate operations are complete; the STS-35 strut was originally showing the 1 thread minimum after the first mate and was showing 2 threads after the second mate. Prior to STS-35, the minimum thread exposure was 6, and maximum was 26. The Integration Project reviewed the STS-35 offset configuration for impact on the Space Shuttle Vehicle (SSV) during ascent. Interface loads evaluation found an increase in the Cy Beta load [approximately 10,000 pounds (lb)]. This increase was considered minor (contributing less than 1%) when compared to other flight loads, such as wind effects. The calculated increase was determined not to precipitate a change in the Guidance, Navigation, and Control System (GNCS) nor would it impact the I-loads. Rockwell International (RI) and the Orbiter Project reviewed the OV-102 build process and records for attach point anomalies. OV-102 was built to specification ML0301-0025. Orbiter segments were joined on a scribed grid floor in the plant and shot-in with a theodolite for further accuracy. Resulting measurements were recorded in the Manufacturing Operation Record (MOR) book for future use and comparison. During manufacture of OV-102, it was noted that EO-2 and EO-3 (the left and right aft attach points, respectively) were too close together by 0.46". The EO-2/EO-3 separation tolerance is specified at 0.40", and a waiver for this condition resulted. OV-102 was mated 9 times prior to STS-35. Thread protrusions

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
4 (Continued)		recorded from the previous mates ranged from a minimum of 6 to a maximum of 26 protruded threads. Aft attach bolt inspection showed proper seating at EO-2 and EO-3 indicating correct liner installation. A review of ET-35 manufacturing procedures found no quantifiable inconsistencies that had led to the misalignment with STS-35. However, bipod tolerance verification was not performed with actual flight struts, and there was no verification process with ET-35 in the vertical position. Martin Marietta manufacturing procedures should consistently produce near-nominal centerline offsets at EO-1; however, past history of Orbiter/ET mates demonstrated a large centerline offset variation when examining the thread exposure records (from 1 to 26 threads). The average nominal thread exposure should show a ± 3-thread spread. A review of the STS-35/ET-35 mate process and voice recordings from March 1990 found no real anomalies. OV-102 was yawed in the sling until EO-2 and EO-3 aligned. A +Y offset was first observed as a relative displacement from the centered Ground Support Equipment (GSE). The GSE was bottomed out, and the yoke rotated 0.5", 0.2" above the procedural requirement of 0.3". To remedy this problem, the crane was repositioned in the -Z direction to unload the GSE and to straighten the yoke. Subsequently, the forward bipod adjustable strut was shortened to a point where only 1 thread protruded (it was previously reported that the technician adjusting the strut turned it beyond the 1 thread, but later backed it up so that the 1-thread requirement was met). Previous to this OV-102 mate, 6 protruding threads was the minimum recorded. Optical measurements of ET-35 EO-2 and EO-3 proximity elevations in the mated configuration found a misalignment; EO-2 was determined to be 0.2" higher than EO-3. The effect of this misalignment was calculated, and it was determined that a 0.2" difference at the aft attach points would result in an approximate 0.96" offset at

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
4 (Continued)		EO-1 in the +Y direction. Based on these measurements, the ET Project stated that it did not feel that the manufacturing tooling setup was maintained to drawing tolerances for ET-35 at EO-2 and EO-3. A recommendation was made to verify ET interfaces on ETs at Kennedy Space Center (KSC) prior to Orbiter mate. Additionally, Martin Marietta and the ET Project determined that there was no structural integrity problem with ET-35 in the offset configuration. The optical measurements also determined that the bipod angle was approximately 1.5°, indicating like rotation of the forward separation bolt. The Orbiter Project evaluated this rotation relative to potential interference with the pyro electrical connector and separation functions. This evaluation led to the conclusion that the pyro function would not be affected by contact of the separation bolt and electrical connector. The pyros would fire, and additional roll angle would not impair physical separation. The results of the indepth investigation determined the following: • ICD/engineering drawing tolerances did not account for the total misalignment experienced on STS-35. This determination was based on the fact that worst-case tolerance buildup would only result in a misalignment of 1.002" at EO-1. • ET-35 measured data indicated that EO-2 was 0.2" higher than EO-3, equating to a 0.96" misalignment in the +Y direction at EO-1. • Misalignment was not due to mate operations. The misalignment was due to ET-35 EO-2 and EO-3 misalignment at manufacturing. • Orbiter manufacturing records demonstrated that OV-102 was built to specification. All previous OV-102/ET mated configurations did not result in similar Orbiter/ET misalignment.

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
4 (Continued)		Rationale for STS-35 flight with this condition was: • Additional induced loads and performance variations were within Space Shuttle Program criteria and experience. • The pyro separation system would function properly. • There would be no ET structural degradation due to the misaligned configuration. *This risk factor was acceptable for STS-35.*
5	Liquid Oxygen (LOX) T-0 umbilical foot receptacle 1/4" bolt problem. HR No. INTG-166 {C} *No further problems were experienced on the STS-35 stack prior to launch.*	During a routine quality assurance/engineering walkdown inspection at the pad, the LOX T-0 umbilical foot receptacle was found to be rotated to a position which allowed unlocking of the umbilical left side. Examination found that a 1/4" bolt which prevents the umbilical foot receptacle from rotating had sheared off at the bolt head and nut. The T-0 umbilical was demated, and the nut part was retrieved. Both left and right foot receptacle bolts were replaced. No further problems with bolt shear were experienced during the replacement. *This risk factor was resolved for STS-35.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
6	OV-105 Gaseous Hydrogen (GH_2) Flow Control Valve (FCV) broken poppet. HR No. INTG-151 {AR} ET S.09 {AR} ***No GH_2 FCV anomalies were reported on STS-35.***	During acceptance testing at Wyle Laboratories in 1990, 2 of 3 OV-105 GH_2 FCVs failed. The first failure occurred in May after approximately 11 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 similar manner as 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. Analysis of the poppet which failed in May determined that the failure was due to simple overload. Analysis of the recently failed poppet is in work. The lost poppet material was recovered from the test stand debris trap downstream. The test stand upstream filter was inspected and verified as clean and undamaged in both cases. Rationale for STS-35 flight was: • This type of fracture would not result in the 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 fails open (Crit 1R2).

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
6 (Continued)		• OV-105 FCV poppet failures appeared to be material or process related: 2 of 3 poppets from the same lot failed. • There were no similar FCV poppet failures during testing or flight. ***This risk factor was acceptable for STS-35.***
7	LO_2 fill and drain valve closure in the event of power loss in terminal sequence. HR No. INTG-153 {C} ET P.02 {C} ***Power was not lost during the STS-35 terminal sequence.***	During STS-31 prelaunch activities, a failure in the Launch Complex 39 utility annex resulted in a momentary power fluctuation causing the firing room launch sequence equipment to halt. The concern was that a similar power failure during the last 31 sec prior to launch would cause the firing room to lose control visibility. In the case of a subsequent on-pad abort, the firing room would be unable to control LO_2 fill and drain valves, resulting in the potential for geysering. A procedure was approved and implemented during the STS-31 launch countdown to ensure geyser suppression in the event of a similar power failure during the terminal sequence. This procedure initiated transfer line leak purge at T-80 sec, and the crew would be instructed to open the LO_2 outboard and inboard fill and drain valves using on-board switches. During STS-31 terminal countdown, prerequisite control logic prevented the LO_2 outboard fill and drain valve from closing with the transfer line leak check purge activated, resulting in a momentary delay in the launch. Procedural changes were implemented for STS-35 to ensure that the prerequisite control logic was satisfied. A bypass of the outboard fill and drain valve control logic was performed at T-9 minutes (min). LO_2 transfer line leak check purge was initiated at T-80 sec. If power had been lost to the firing room, verification that the MLP and the pad terminal connection room power lights are on was available at the safing panel. Safing panels and critical GSE are supported with batteries. The LOX Console Operator configured the safing panels so that tank prepress was on, the main fill valve was open, and the ET vent was in the closed position. The

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
7 (Continued)		NASA Test Director, in the event of power loss, would instruct the crew to open the LO_2 outboard, and then inboard fill and drain valves from the Orbiter. After confirmation from the crew that switches were thrown, the LOX Console Operator would activate the safing panel. Transfer line leak check purge was deactivated, and the Tail Service Mast (TSM) vent and drain valves would close. LO_2 tank pressurization would start, and the facility fill valve would go to the open position. LO_2 should then begin draining from the vehicle/ET. Approximately 2 min after power restoration, the LOX Console Operator could cycle the tank prepress with a switch to maintain nominal LO_2 drain pressures. The Launch Process Sequencer would regain control of the launch data bus in 7 to 8 min after power restoration. Full firing room capability would be restored in 30 to 45 min. If this sequence were to occur, the NASA Test Director would have executed a Mode 1 egress after switch action was executed. Mode 1 egress is an unassisted emergency crew evacuation from the vehicle. Concerns were raised relative to the crew/vehicle safety due to the hurried nature of a Mode 1 egress. Inadvertent switch/control activation may result if the crew is too hurried. Since the above procedure for geyser suppression should safely drain LO_2 from the vehicle/ET, egress timing would not be as critical as it would be for other conditions. Crew evacuation from the pad was also questioned in this case. Mode 1 egress requires the use of emergency slides as the primary evacuation method. However, associated risk to the crew could be too great to warrant use of the slide wires. A decision was made to instruct the crew that a call for a Mode 1 egress in this case would be an orderly unhurried egress.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
7 (Continued)		Rationale for STS-35 flight was: • Procedural changes were implemented to control the potential for LO_2 geysering. • STS-35 crew was briefed on procedural changes and was prepared to implement them in case of power loss. ***This risk factor was acceptable for STS-35.***
8	LH_2 leaks on STS-35/OV-102 and STS-38/OV-104. HR No. INTG-006A {AR} INTG-071 {AR} ORBI-306 {AR} ***No LH_2 leaks exceeding the LCC limits at the 17" LH_2 disconnect or in the aft compartment were experienced during the final launch countdown that led to the successful launch of STS-35 on December 2, 1990.***	Significant LH_2 leaks were experienced on STS-35/OV-102 and STS-38/OV-104. Extensive investigation of the leaks was performed. Following is a summary of the investigation findings. LH_2 leaks were discovered during the first launch attempt for STS-35/OV-102. Almost immediately after slow fill operations began, the Hazardous Gas Detection System (HGDS) indicated increasing levels of H_2 in the aft compartment and the TSM. Shortly after LH_2 fast fill operations were initiated, the NASA Test Director was alerted to H_2 concentrations in the aft compartment exceeding the 600-parts per million (ppm) Launch Commit Criteria (LCC) redline. Aft compartment concentrations rose rapidly, reaching 2900 ppm before LH_2 fill was stopped. H_2 concentrations at the 17" disconnect area also rose to a level of concern. The initial conclusion was that a leak existed at or near the OV-102 high-point bleed valve, based on the H_2 levels recorded in the TSM. The launch attempt was scrubbed, and troubleshooting plans were prepared to isolate the leak location. During troubleshooting, H_2 concentrations in excess of 40,000 ppm were recorded at the 17" disconnect area on Leak Detectors (LDs) 54 and 55. Troubleshooting indicated 2 H_2 leaks: at the 1-1/2" TSM flex hose lower flange seal in the area of

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
8 (Continued)		the 17" disconnect, and not at the high-point bleed valve. A small leak was found in the 1/4" LH_2 flex hose that was recently replaced; however, it was determined that this leak would not create the H_2 concentrations levels experienced during the fill and troubleshooting operations. Leak checks of the OV-102 17" disconnect primary seal were performed using helium; no discrepancies were found. Latch shaft seal leak checks were completed with no leaks identified. Visual inspection found no discrepancies. A 30-pounds per square inch gage (psig) mass spectrometer helium leak check was performed; no leaks were identified. Leak checks of the ET/Orbiter disconnect cavity and disconnect interface primary seal were performed with the ET pressurized to 45 pounds per square inch absolute (psia). During this test, no leaks were detected at the primary seal; however, ET/Orbiter disconnect cavity readings rose from 1100 ppm to 7000 ppm. The test included decrementing the ET pressure in 5-psia increments to 0 psia; no leaks were found. A mini-tanking procedure was approved for STS-35/OV-102 to allow further isolation of the leak at the 17" disconnect area and to provide the foundation for a rollback decision. LH_2 fill operations were identical to those used for a launch tanking. Additional instrumentation was located in the 17" disconnect area and aft compartment to isolate a leak source. H_2 concentration limits were set to maintain safety. High H_2 concentrations were recorded similar to those experienced during the initial launch tanking operation. Because of the high H_2 levels, the leak source could not be conclusively isolated to either the Orbiter or the ET. Indications were, however, that the leak source was in the 17" disconnect flapper shaft seals. The decision was made to roll back to the VAB, demate, and remove the ET 17" umbilical for off-line testing. Visual inspection of the ET and Orbiter disconnects found no anomalous conditions. The ET disconnect was removed on June 19, 1990, and sent to RI for testing.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
8 (Continued)		Testing of the ET-side disconnect at RI found no significant leaks. Based on this determination, the Orbiter-side 17" disconnect was removed and sent to RI for leak checks and testing together with the ET disconnect. Results of these tests at RI were inconclusive. The Orbiter-side disconnect was removed from OV-105 and was installed on OV-102 for STS-35. A new 6000-series ET disconnect was installed on ET-35/STS-35. In parallel with troubleshooting operations at KSC, a record review of 17" disconnect manufacturing and build papers was conducted. Parker-Hannifin, the 17" disconnect vendor, determined that the STS-35/ET-35 17" disconnect, S/N 2023, was among a group of 8 disconnects (S/N 2018 through S/N 2025) that were identified as being acceptance tested with an anomalous simulator. The simulator used was not under program control, and the anomalous operation was not investigated. Leak checks to verify seal integrity were performed subsequent to these anomalies with a blanking plate. The results in all cases were good; however, the flapper valve was in the closed position when tested with the blanking plate. Other disconnects in this suspect group included: S/N 2025 on STS-29/ET-36 that had an LH_2 vapor cloud anomaly, S/N 2019 on STS-33/ET-38 that flew with no problem, and S/N 2018 on STS-38/ET-37. Record comparison of S/N 2023 and S/N 2018 determined that similar leak rates were experienced during acceptance testing. This led to a decision to perform a mini-tanking test on the LH_2 side of STS-38/ET-37 to verify the integrity of the LH_2 system. Extensive testing and teardown at Parker-Hannifin of both the OV-102/ET-35 Orbiter and ET disconnects resulted in a non-conclusive determination of the cause of the STS-35 H_2 leak. Teardown revealed small quantities of contamination, mostly in the form of teflon flakes and particles. Glass beads were also found in the shaft seal areas of both disconnects. Marginal seal design coupled with this contamination was the most probable cause of the STS-35 leak.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
8 (Continued)		Because of the H_2 leak on STS-35/OV-102 and the non-conclusive determination of the cause of that leak, a decision was made to perform a mini-tanking test on STS-38/OV-104 prior to launch. This test resulted in a similar H_2 leak condition. LD 54 and 55 saturated above 40,000 ppm during fast-fill operations; however, there were no abnormal H_2 concentrations recorded in the aft compartment. Analysis of the STS-38/OV-104 H_2 leak led to a decision to conduct a second tanking test to further isolate the leak source in the disconnect area. Baggies were installed around portions of the disconnect to isolate potential leak paths. Results of the second test determined that the STS-38/OV-104 leak originated at the ET-side 17" disconnect flange (the interface between the ET 17" disconnect assembly and the 17" feedline to the H_2 tank). No H_2 concentrations were recorded external to the installed baggies (i.e., LD 54 and 55). This test, however, was unable to quantify the actual leak rate. The STS-38/OV-104 investigation that followed focused on potential problems with the 17" flange that could provide a leak path. Molds were taken of the 17" flange gap and compared to manufacturing drawings; no problems were found. Proper torquing of the 48 17" flange bolts was verified to 600 inch-pounds (in-lb). A decision was made to overtorque these bolts by 10%. Eight of the 48 bolts did not move when overtorqued; however, the remaining 40 bolts rotated as much as 30° when 660 in-lb torque was applied. Dye penetrant examination of the 17" flange welds was performed with no anomalous conditions found. Examination of the parent metal around the flange also found no problems. After careful examination of the STS-38/OV-104 17" flange, a third tanking test was planned to determine if the leak could be completely isolated to the flange and to quantify the leak rate. A baggy was installed around the flange circumference and vented through a turbo mass spectrometer and flow meter. In addition, 12 1-ft "rakes" of sensors were distributed around the disconnect area to measure and map any leaks. H_2 concentrations were recorded almost immediately after slow fill

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
8 (Continued)		operations commenced. Once into fast fill, LD 54, LD 55, and a few "rake" sensors began picking up H_2 concentrations. Approximately 3 min into fast fill, H_2 concentrations at LD 54, LD 55, and other sensors jumped to saturation levels above 40,000 ppm. A large vapor cloud was seen emanating from the 17" disconnect insulating foam in the approximate area of the flange. Repeated applications of fast fill operations resulted in a similar vapor cloud and multiple-sensor saturations. It was later determined that the baggy was not in the proper configuration to collect H_2 leaking from the flange; therefore, the baggy leaked. The third tanking test was unable to quantify the flange leak due to the baggy failure. The decision was made to roll back and demate STS-38/OV-104 for further evaluation. Once demated, the STS-38/ET-37 LH_2 17" disconnect and feedline were removed intact and sent to Marshall Space Flight Center (MSFC) for evaluation. Plans were to evaluate the disconnect/feedline with LH_2 cold flow tests to further isolate the leak. After these tests were completed, the disconnect/feedline was to be disassembled for inspection and cause assessment prior to further program flights. However, cold flow tests revealed that the leak did not emanate from the 17" flange as previously thought. Tests conclusively demonstrated that LH_2 leak paths were in the area of the disconnect shaft seals, as seen on the STS-35/ET-35 disconnect. Because of this conclusive finding, teardown of the ET-37 disconnect/feedline assembly was not required prior to the STS-35 launch attempt. Preparations for the second launch attempt of STS-35/OV-102 in September 1990 included additional measurements and inspections to define the condition of the mated 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 OV-102 disconnect was new and would be flown for the first

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
8 (Continued)		time. The 17" flange bolts were torqued to 660 in-lb. A tanking test was not performed on STS-35/OV-102 prior to the second and third launch attempt. The second launch attempt was scrubbed for reasons other than LH_2 leaks. The third launch attempt was made on September 5/6, 1990. Shortly after entering fast fill during tanking, the HGDS identified that the aft compartment H_2 concentration had risen above the 600-ppm LCC limit. Tanking was continued in fast fill, and the aft compartment H_2 concentrations rose to 6800 ppm. The decision was made to scrub the third launch attempt and continue in a troubleshooting mode to further identify the leak source in the aft compartment. During the first launch attempt and the STS-35 tanking tests, aft compartment concentrations rose above 5000 ppm. Indications during troubleshooting were that the leak emanated from the area of the recirculation pump package, primarily in the area of the engine #2 recirculation pump. Removal and replacement of the entire pump package was directed and performed. The pump package was sent to Johnson Space Center (JSC) for testing. No leak path was found during the JSC test effort. During performance of the helium signature leak test after the pump package installation, a small leak was found in the vicinity of the engine #3 prevalve detent cover. While this leak was within specification, it was significantly greater than that witnessed at the 2 other prevalves. Upon removal of the detent cover, the 2 1/2" teflon cover seal was found to be severely crushed. This seal was replaced in January 1990 during inspection for possible MPS contamination. It is believed that the seal was improperly installed at that time. It is also believed that the crushed prevalve detent cover seal was the primary H_2 leak path that resulted in high aft compartment H_2 concentrations during the launch attempts and tanking tests. The LCC for Orbiter/ET 17" umbilical H_2 concentrations was also readdressed for STS-35. After considering the results of the LH_2 leak investigation and extensive

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
8 (Continued)		review of the test data, the following maximum redline requirements were established in the LCC: • No presence of unusual vapors and liquid droplets. The term "unusual vapors and liquid droplets" is defined as: – An obvious blowing leak or a vapor cloud which obscures the disconnect or feedline region for an extended period (> 5 min). – Consistent frequent liquid drops falling or flowing with identifiable vapor trails. • No H_2 concentration greater than 40,000 ppm (4%) on both sensors (LD 54 and 55). If a sensor has been declared failed, the remaining sensor must not exceed 40,000 ppm (4%). • No H_2 concentration greater than 20,000 ppm (2%) on 1 of 2 sensors (LD 54 or LD 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. Because it was uncertain that all leak paths in the aft compartment were fixed, the Level I PRCB directed that the aft compartment LCC be raised. The LCC for aft compartment H_2 concentration was increased to 1000 ppm during fast fill and 300 ppm during stable replenish. Safety concurred with this LCC for STS-35 and concurred that it was safe to fly provided these LCC requirements were not exceeded or waived.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
8 (Continued)		During the fourth launch attempt on September 18, 1990, aft compartment H_2 concentrations again were well above the LCC limit. Extensive inspection and leak checks uncovered another crushed detent seal; this time in the engine #2 prevalve detent cover. This and the other 2 detent cover seals were removed and replaced using an improved installation technique. This technique included further trimming of foam insulation around the detent covers and the use of a mirror to enhance installation visibility. A review of all LH_2 MPS system joints disturbed during the OV-102 MPS contamination investigation (see Integration 2 above) was also accomplished. This review cleared all disturbed joints as potential leak paths. Another tanking test was planned and executed on October 30, 1990. Planning for this test included enhanced instrumentation which consisted of 17 leak detectors, 26 mass spectrometer sense lines, 5 thermocouples, a Carbon Dioxide (CO_2) injection system, and the use of 10 cameras, including 2 Atlas-Centaur Program cameras designed for use in H_2 environments. Portions of the MPS system were also bagged and instrumented to determine leak rates.. The results of this tanking test were excellent. Including minor leaks encapsulated in bags, the total H_2 concentration in the aft compartment was determined to be 145-150 ppm; well within the acceptable limit of 600 ppm. The test cleared STS-35/OV-102 for launch. Rationale for STS-35 flight was: • The Orbiter and ET 17" disconnects installed on STS-35/OV-102 were new. • The tanking test conducted on October 30, 1990 indicated that the leak sources were identified and repaired. • LCC requirements would protect against launching with an excessive H_2 leak both at the disconnect and in the aft compartment. ***This risk factor was acceptable for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
9	Holddown Post (HDP) cracks. HR No. INTG-158B {AR} ***Post-launch inspection of the MLP #1 HDPs found no crack growth on HDP #7 and no cracks on the other HDPs.***	During strain gage replacement on the MLP #2 HDPs after STS-41, a crack was found on HDP #3. The crack was determined to be 2" long and 1/4" deep. Based on this finding, inspection of all MLP HDP 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 on the old HDP design. The new HDP design, currently in use (and includes the 3 HDPs with cracks), 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 and not mechanical loading. Thermal loads applied are the direct result of SRB plume impingement on the MLP deck surfaces, followed immediately by quenching with water from the deluge system. HDPs #3, #4, #7, and #8 experience the highest concentration of plume impingement of the 8 MLP HDPs. Stress analysis indicated that the cracks were in a low-stress area; average stress in the crack vicinity is 6,000 psi. Review of data from strain gages located 24" above the HDP base showed stress values of 8,400 psi. On the average, stress increases with distance from the HDP base and, therefore, stresses at the crack locations identified were estimated to be less. HDPs were designed for a minimum stress of 85,000 psi and, based on stress calculations at the crack site, the resulting Factor of Safety (FOS) is above 10. It was not believed that there was an appreciable reduction in this FOS due to the observed surface cracking.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
9 (Continued)		Rationale for STS-35 flight was: • The discovered crack on STS-35/MLP #1 was in a low-stress area of HDP #7. • Cracks are a result of thermal cycling, and not mechanical loading. • Similar cracks were experienced on the HDPs of older design; 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-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Main Landing Gear (MLG) uplock hook deformation. HR No. ORBI-182 {AR} ***No MLG anomalies were reported on STS-35.***	The MLG uplock hooks on OV-102 were found to have permanent deformation due to overload caused by misrigging. The OV-102 Left-Hand (LH) and Right-Hand (RH) hooks were found to have yielded 0.019" and 0.063" outside of the allowable rigging specifications. The suspected cause of the gear uplock hook yield conditions was the original rigging of OV-102 at RI/Palmdale. As a result, changes to the rigging specifications were identified and were incorporated. The RH uplock hook on OV-102 was removed and replaced, and both the RH and LH MLG mechanisms were rerigged to the new specification. Both main gear doors were positioned to within 0.020" of the pre-rigging position. The excessive preload caused by misrigging no longer existed. The door latches, latch linkages, and stops were all within specification. The door and gear proximity switches operated correctly, the booster bungee gaps were within specification, and the MLG strut upstops made contact with the strut. Nondestructive Evaluation (NDE) of the RH and LH uplock hooks was performed, and no cracks were found. Fracture mechanics stress analysis indicated a positive safety margin for the MLG uplock hook. The stress analysis concluded that no other mechanism/structure was affected by the overload condition. Proper operation of the door and uplock features was verified during the Orbiter Processing Facility (OPF) testing, after rework was completed. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2	MLG oversize uplock fitting mounting holes. HR No. ORBI-182 {AR} ***No MLG anomalies were reported on STS-35.***	As a result of observed motion, the RH uplock fitting was removed for fastener inspection. The holes in the RH inboard fitting and intercostal were found to exceed the drawing tolerance. It was determined that the hole condition was not related to wear. The worst-case holes will be inspected between flights. The oversize holes were not considered to be a safety-of-flight issue. ***This risk factor was not a safety concern for STS-35.***
3	Nose Landing Gear (NLG) axle bearing nut. HR No. ORBI-179 {AR} ***The NLG axle bearing nuts were found to be properly in place during postflight inspection.***	During installation of the nose wheel assembly on STS-35/OV-102 at 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. The OV-102 axle assembly was shipped to Menasco for further investigation. At Menasco, the retainer nuts were removed by milling the inner diameter to preserve the nut threads. Inspection of the bearing and bearing race found no degradation. The nut thread crest was not sheared, but was found to be rolled approximately 0.009" at the 12-o'clock position. The housing threads were unaffected. The bearing and bearing grease were found to be in excellent condition. Magnetic particle inspection of the housing verified that there were no cracks. A 2" long by

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)		0.22" wide area on the inside of the axle housing was determined to be untempered martensite. The piston assembly was reworked to eliminate the untempered martensite condition at the axle contact area. (See this section, Orbiter 4, for further details concerning the untempered martensite problem.) Stress analysis verified slapdown loads as the cause of the nut cocking. Slapdown loads of 90,000 pounds force (lbf) or greater provide radial forces on the retainer nuts sufficient to displace the top of the nut and cause it to "jump" 1 thread. Slapdown loads sufficient to cause a retainer nut to "jump" 2 or more threads would fail the NLG structure and result in a catastrophic event. OV-102 nuts were found with the top of the nut or 12-o'clock position displaced by 1 thread. Radial forces at NLG slapdown result in elastic deformation of the NLG housing at the nut ends. The radial forces also generate an axial component on the top part of the nut through the bearing race. The combination causes crossthreading of the first thread at the 3- and 9-o'clock positions; the threads at the 6-o'clock position were in place and in good condition. Rationale for STS-35 flight was: • New retainer nuts were installed on the OV-102 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. ***This risk factor was resolved for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4	Untempered martensite in the NLG. HR No. ORBI-179 {AR} *No NLG anomalies were reported on STS-35.*	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 3.) 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. Formation of martensite was determined to be caused by contact of the rotating Inconel 718 axle with the housing during touchdown. Menasco determined the depth of untempered martensite to be 0.41" by successively removing a thin layer, polishing to 125 rms, and performing etch inspection. The total depth removed was 0.061" which included the overtempered (softened) zone surrounding the untempered martensite. The reworked OV-102 piston/axle housing was subjected to magnetic particle testing. No cracks were found in the housing. The assembly was refurbished to print (new bearing nuts, etc.). The axle was reworked to remove the centerline ridge that causing contact with the housing as a result of the slapdown loads. The reworked axle, with the centerline ridge removed, would not contact the housing under anticipated slapdown load conditions. Therefore, additional martensite formation should not occur. Rationale for STS-35 flight was: • The OV-102 NLG piston/axle housing was reworked to remove the areas of untempered martensite. • Stress analysis determined that a positive safety margin existed with 0.1667" thickness remaining in the cleaned and polished area of the housing after the martensite removal and blending operations. *This risk factor was resolved for STS-35.*

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Upper diagonal strut damage. HR No. ORBI-179 {AR} ***No anomalies attributed to upper diagonal strut damage were reported on STS-35.***	The OV-102 upper diagonal strut vertical thrust structure, aft fuselage, contained a deep gouge and scuff marks on the boron/epoxy reinforcement, and there was possible damage to the TI 6AL4V tube. The strut carries tension or compression loads. Boron/epoxy provides strength and stiffness, and minimizes the strut weight. Fabrication requires heat and pressure cure. Radiographic inspection at 0° and 90° was performed; no damage to the titanium wall of the strut was found. Analysis was performed using 6.0 loads and STS-35 flight-specific loads. The strut had positive margins of safety for ultimate loads, assuming that the damaged strip is nonfunctional. The strut was repaired and approved for unrestricted use by blending the defect and filling it with epoxy. ***This risk factor was resolved for STS-35.***
6	Titanium fitting tee cracks. HR No. ORBI-188 {C} ORBI-050 {AR} ***No further hydraulic leaks were experienced on STS-35 after replacement of the fittings.***	Twenty-five cycles into the landing gear functional testing of STS-35/OV-102 in the OPF, a hydraulic leak was observed behind the NLG. The leak was traced to a 3/8" x 3/8" x 3/8" tee swaged fitting that had 3 cracks in the area of the swaging tool marks. Experience showed this to be a generic problem associated with titanium fittings throughout the hydraulic subsystem. This failure occurred after approximately 11 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 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 similar manner as 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.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)		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. Analysis of the poppet which failed in May determined that the failure was due to simple overload. Analysis of the recently failed poppet is in work. The lost poppet material was recovered from the test stand debris trap downstream. Inspection of the test stand upstream filter verified a clean and undamaged condition in both cases. Rationale for STS-35 flight was: • Extensive testing by RI concluded that the crack would not propagate when exposed to flight environment stresses. • The leaking fitting was replaced with a titanium fitting, and leak check was satisfactory. ***This risk factor was resolved for STS-35.***
7	Emergency egress slide. HR No. INTG-102 {C} ***The emergency egress slide was successfully reinstalled after performing the waste water collector Detailed Test Objective (DTO).***	Due to a DTO for the improved waste collector on STS-35, the emergency egress slide must be removed; Inflight Maintenance (IFM) allows for reinstallation of the slide on orbit. The slide must be reinstalled prior to deorbit. Inability to reinstall the slide results in possible loss of a crew member if rapid emergency egress is required. The crew may be unable to reinstall the slide while on orbit due to structural deformation of the crew cabin. The crew should check for relative distortion and pin tightness before removing the pins. If the pins are binding, the egress slide will not be removed on orbit. The IFM tools that are on board to assist the crew to reinstall the slide on orbit can be utilized as a backup if the original mounting pin cannot be reinstalled. Analysis indicated adequate slide pack tolerance to allow for the reinstallation.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)		If the slide cannot be reinstalled, alternate egress methods exist: installation into the original mounting postlanding; alternate mounting location on the side hatch; use of Sky Genie through the side hatch or the overhead window; or ground crew assistance when available. Alternate egress methods can increase crew escape times postlanding. There is a Program requirement for postlanding emergency egress capability within 1 min following rollout. The added risk of increased egress time if alternate methods are used is considered minimal. Rationale for STS-35 flight was: • If the slide cannot be reinstalled, alternate egress methods exist. ***This risk factor was acceptable for STS-35.***
8	Increase in the ET-35 (STS-35) LH_2/LO_2 17" disconnect flapper tip loads. HR No. INTG-035 {AR} ***No anomalies attributed to the higher loads were reported on STS-35.***	The flapper tip loads are measured during ATP by the supplier and again at KSC to verify minimum preload force. The OMRSD requirement for the ET tip load measurement at KSC is 55-lb minimum (no maximum specified), and the tip loads must remain within 10 lb of the value measured at ATP. Flapper tip loads measured on ET-35 were 82.6 and 79.6 lb for the LH_2/LO_2 disconnects, respectively. These values were both 12.6-lb higher than the ATP values. This problem was documented in Corrective Action Report (CAR) MMO167, and the condition was approved for flight by Waiver WK1714R2. Detailed inspection of the hardware was conducted in August 1984, which was prior to standardizing the valve actuation techniques and the slave unit configuration. The lack of standardization could explain the 12.6-lb change in the tip load value from that measured at ATP. Prior to standardization, variances in actuation rate alone could result in tip load variations in excess of 10 lb. Standardization

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		incorporates a pneumatic system that functionally duplicates the orbiter actuation system. Adjustments to the slave unit were also incorporated that may influence the ATP values. ET-35 tip load data was within both water flow test and flight experience. LH_2 tip loads as high as 97 lb were successfully tested in water flow. Flight tip loads of 88 and 86.1 lb were flown on ET-16 and ET-30. During flow conditions, tip loads, if measurable, would increase beyond the 300-lb range. The increased ET-35 tip loads were minimal when these forces were considered. Both units demonstrated the minimum 450 pounds per square inch (psi) actuation requirement. Cycling in both the open and closed position during tip load measurements was nominal. The valves met the specification times during post-mate cycling. No corrective action was deemed necessary. Rationale for STS-35 flight was: • ET-35 tip load data was within both water flow test and flight experience. • Analysis and testing determined that no corrective action was deemed necessary. ***This risk factor was resolved for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	OV-102 Engine Interface Unit (EIU) Power-On Reset (POR) anomaly. HR No. INTG-165 {C} ORBI-066 {AR} *No further PORs occurred after the EIU replacement on STS-35.*	During EIU testing, position 1, S/N 23, the 60-Kilobit (Kbit) data path dropped out for 1.1 sec, and the subsequent read of the Bite Status Register (BSR) indicated POR. This anomaly occurred once on April 27, 1990. The concern was that a simultaneous POR-A and POR-B in the last 30 sec prior to MECO would result in the General Purpose Computer (GPC) closing prevalves on running engines, causing 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-35 crew prior to flight. During the third scrub turnaround, 2 additional PORs were experienced on EIU S/N 23. Removal and replacement of EIU S/N 23 was completed. Retest of the new EIU was successful. The subject POR was characteristic of previous occurrences (7 units, 9 vehicle flows since January 1983). POR was transient and self clearing, and troubleshooting did not reproduce the problem. Rationale for STS-35 flight was: • Each occurrence was a single POR (POR-A or POR-B). • No single POR would fail more than one data command/data channel. • POR has not occurred in flight.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9 (Continued)		• A single POR prior to T-0 would result in a launch hold or launch abort. • Simultaneous reset of both channels requires 2 failures and has never been experienced. • 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 7.9×10^{-12}. ***This risk factor was resolved for STS-35.***
10	STS-35/OV-102 Freon Coolant Loop (FCL) Flow Proportioning Valve (FPV) anomaly. HR No. ORBI-275A {C} ***No further FCL #1 flow degradation was reported after the launch of STS-35. On-orbit FCL operations were nominal.***	During STS-35/OV-102, processing of FCL #1 experienced a sudden shift in the flow rates. While the FPV was in the payload position, the interchanger flow decreased 250 pounds per hour (lb/hr), the payload heat exchanger flow increased 200 lb/hr, and the aft coldplate network flow increased 15 lb/hr. The interchanger and payload flow rates equalized at approximately 1300 lb/hr. When the FPV was switched to the interchanger position, the interchanger flow had decreased from 2280 lb/hr to 1925 lb/hr. Flow changes indicated an increase in the ΔP on the interchanger leg of the FCL. Any increase in the ΔP in one leg would result in more flow being directed through the other parallel flow legs. Valve cycling did not improve the problem. Occurrences of shifting flow rate continued in a random manner. Preliminary analysis and troubleshooting indicated that the problem existed in the FPV, with the most probable cause being blockage of an internal orifice or outlet filters in the FCL. This anomaly would have resulted in a violation of the LCC if it occurred prior to launch, or violation of Flight Rule requirements for minimum FCL flow rates if it occurred on-orbit. Both rules protect against the next worst-case failure which is

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10 (Continued)		loss of one FCL (leakage or blockage). The LCC launch minimum is 2150 lb/hr for the interchanger flow rate (generic LCC rule), and the Flight Rule on-orbit minimum flow rate is 2025 lb/hr for the interchanger. Loss of 1 FCL requires an abort to orbit/first day primary landing site, with power down of 3 GPCs. Consequently, Program Management decided to replace the FPV and pump inlet filter at the launch pad. Post-removal inspection found material missing from the FPV inlet screen. Further inspection of the FPV found pieces of the inlet screen in the valve spool. The orifices in the valve spool were partially blocked by pieces of the inlet screen. Contamination products were also found. The composition of the contamination is under investigation. FCL pump package inlet filters were removed, inspected, and found to be very clean. This finding ruled out migration of contamination to the inlet screen. Metallurgical examination determined that the inlet filter screen failed due to corrosion. The cross-sectional area of the screen was reduced by the corrosion, and the load on the screen increased from the higher ΔP due to reduced flow area. The final failure occurred when the remaining screen area could no longer sustain the load. A high chlorine concentration was found in the corroded screen areas. Freon samples met requirements for chloride content, <0.3 ppm. The outlet screens were less severely corroded than the failed inlet screen. The FPV and FCL pump package filters were replaced. Drying and freon servicing of FCL #1 was completed. Flow testing indicated rates better than previously seen on OV-102. Because of the contamination found and associated problems with the humidity separators, JSC Safety, Reliability and Quality Assurance (SR&QA) initiated a program to review all Orbiter fluid systems and to make recommendations for improved preventive maintenance. This effort is not yet complete.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10 (Continued)		Rationale for STS-35 flight was: • Flight rules dictate that loss of 1 FCL due to freon flow rate degradation or other anomaly requires a next primary landing site reentry on 3 GPCs. • Freon flow rates are monitored by the Mission Control Center (MCC). If flow rates degrade, Flight Controllers can detect decrease in flow and plan reentry to avoid the potential of less than a 3-GPC reentry on a single FCL. ***This risk factor was resolved for STS-35.***
11	Aluminum rivets installed in the wing assembly without proper heat treatment and corrosion protection. HR No. ORBI-277 {C} ***No anomalies attributed to faulty wing assembly rivets were reported on STS-35.***	While removing the tape used to protect areas during tile work on OV-105 wings, the head of a 5/32" 2000-series rivet came off a clip that holds a RH wing tip stiffener. There was a concern that multiple rivet failures could lead to structural failure of the wing assembly. Scanning Electron Microscope (SEM) analysis of the failed rivet found that the failure was due to stress corrosion cracking near the rivet head. Stress corrosion is indicative of improper heat treatment; however, Alcoa Aluminum indicated that there is no reliable test to determine that the proper heat treatment was performed. Stress corrosion is also indicative of a corrosive environment and a high tensile load. The greater concern was that the rivet failure was indicative of a generic problem, and other Orbiter wing assemblies could contain rivets that had a similar level of stress corrosion. There has been only one other recorded rivet failure on a wing assembly. There have been 24 additional rivet failures recorded in various applications. There are over 5 million 2000-series rivet applications in the Orbiter fleet.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)		The OV-105 wing assembly was manufactured at Grumman in 1985 (under a 1974 contract) as a spare. That contract did not require the application of Koropon, an anti-corrosion protection, to each rivet as current manufacturing specifications require. Additionally, most rivets used on all Orbiter wing assemblies were purchased at the same time, and there is no lot traceability. Metallurgical analysis of 164 removed rivets found no evidence to conclude that a single "bad" lot of rivets caused the cracking. There was no indication of eutectic melting, no continuous grain boundary precipitates, and no unusual elements or phases. The results of these analyses indicated some lot-to-lot variation in resistance to intergranular cracking. These variations were determined not to be sufficient to cause cracking; however, variations explained the intergranular nature of the cracks. RI drilled out 88 additional rivets from the failed rivet area on the wing assembly and the opposite wing assembly. Seventy-six rivets from WA-18 and Line Item #36 test articles were also examined. Four rivets had similar stress corrosion: 3 from the OV-105 wing assemblies and 1 from Line Item #36. Two WA-18 rivets in the identical application as the OV-105 problem rivets were found with small cracks. A stress corrosion crack was found 180° around a rivet that was next to the failed rivet. Similar stress corrosion was found on the rivet removed from the opposite wing in same position as the failed rivet. The only common factor found was a gap beneath the clip. It was not certain that these gaps existed prior to rivet installation. Grumman requires discrepancy reports to be written against discovered gaps over 0.030". Gaps under 0.030" are dealt with through standard repair such as liquid or metal shimming. A review found that no Discrepancy Reports (DRs) were written and there was no record of standard repair on any of the failed rivets. There was, however, a material review report written against installation problems noted on both OV-105 wing tips. RI requires gaps over 0.006" to be identified on a DR; gaps under 0.006" are dispositioned with standard repair. Here too, there is no record of RI action relative to gap identification.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)		Stress analysis indicated that a positive safety margin of 0.97 against buckling during ascent would exist even if both OV-105 wing assembly rivets with stress corrosion failed. Rivets are designed for shear loading; stress corrosion cracking is only possible when rivets are in tension. However, this particular application puts the rivet in tension. Tension loading is amplified by the 0.028" gap witnessed at the clip on the left OV-105 wing. The OV-105 rivet failure investigation concluded that the OV-105 anomaly was caused by wing tip assembly difficulties that resulted in a gap of 0.028" beneath the riveted clip. The rivet failure mechanism was intergranular cracking from high sustained stress and natural environmental exposure. This problem may exist on all vehicle wing tips. Stress analyses demonstrated that no structural or tile problems would result if the wing tip clip rivets failed. Assessment of rivet lots showed variations in microstructure; however, no out-of-specification conditions were found. The JSC SR&QA position was that OV-102 was safe to fly without inspection; however, inspection of wing tip rivets should be undertaken after STS-35. Additionally, rivet applications in all areas of the Orbiter wing will be examined to determine if any others are in tensile load. The failed rivets will be replaced with hi-lok fasteners on all Orbiters as soon as practical. Rationale for STS-35 flight was: • A similar condition, if it existed on OV-102, was acceptable. • There was no historical evidence of a generic 2000-series rivet problem. – Large areas of tile were removed from OV-102 during the post-Challenger down period; no similar rivet anomalies were discovered.

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
11 (Continued)		– Several structural modifications were implemented in the fleet including numerous rivet drillouts with no anomalies noted. • Stress analysis demonstrated a positive safety margin in the area of the rivet failure. Other rivet applications are in shear and, therefore, should not experience loads that could lead to failure. ***This risk factor was acceptable for STS-35.***
12	External leak on STS-35/OV-102 1/4" LO_2 flex hose. HR No. ORBI-306 {AR} INTG-151 {AR} ***No indications of either LO_2 or LH_2 flex hose leakage were reported on STS-35.***	During 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 OPF Oxygen (O_2) system decay checks. The leaking flex hose and the LH_2 1/4" flex hose were replaced. Leak checks were performed; however, this did not include a repeat of the helium signature test. Retest results for the new flex hoses were good. There are 6 flex hoses of varying sizes used in the LO_2 and LH_2 sides of the MPS on each Orbiter. All are manufactured by Coast Metal Craft. These flex hoses accommodate the vibration and structural deflections between the forward thrust structure and the LH_2/LO_2 17" disconnect pneumatic panels. Four flex hoses, including the LO_2 1/4" 20-psi regulator sense line, have Crit 1/1 failure modes associated with external leakage. Of the 6 hoses, 5 are constructed using 321 CRES bellows covered with 321 CRES braid. The sixth is identical except the bellows are constructed with Inconel 718. There was an unusually high number of problems with welds in these and similar flex hoses. Rationale for flying with similar flex hoses was based in part on weld failures being screenable during acceptance testing.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
12 (Continued)		Failure analysis found a weld bridge between the weld joint and the first convolute of the flex line convolutes. The weld bridge created a stress riser that led to cracking. A crack through the first of 2 plies in the first convolute was determined to be the leak source. Further analysis found fatigue crack initiation sites in the third, fourth, and fifth convolutes. In the area of the leak, bird-caging of the wire braid surrounding the convolutes was observed. This condition is indicative of mishandling; flight environment alone is not sufficient to cause bird-caging. The wire braid is employed primarily to constrain the convoluted hose from vibration or other radial forces. When the braid is damaged to the point of bird-caging, the constraining forces are no longer present, and the convolutes are free to move in any direction. A result of this investigation was to establish inspection criteria for bird-caging for all vehicles. It was determined through failure analysis that external handling damage produced the bird-cage condition in the braid and deformed the convolutes. This localized condition permitted unconstrained motion of the first 5 convolutes of the flex hose. The weld bridge created a localized stress riser at the first convolute resulting in accelerated fatigue crack growth and a leak path. The unconstrained localized motion also caused additional fatigue crack initiation at several places in the deformed convolutes. The conclusion was that this flex hose probably would not fail if the bird-cage condition did not exist. Other sense lines (OV-102 LH_2, OV-104 LO_2, and OV-103 LO_2) were returned to RI for examination prior to the first STS-35 launch attempt. No problems were identified.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
12 (Continued)		Rationale for STS-35 flight was: • The 1/4" LO_2 and LH_2 sense lines on OV-102 were new. Leak and decay checks were successfully completed. • Inspection of other flex lines was completed, and no bird-caging existed. ***This risk factor was resolved for STS-35.***
13	Display Unit (DU) solder joint investigation. ***No DU failures attributed to the solder joint issue were experienced on STS-35. The Payload Data Display Systems that failed on STS-35 were not a subject of this investigation.***	During STS-36, Cathode Ray Tube (CRT) #4, the aft 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 with a cracked solder joint and was attributed to stress, and 1 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.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER 13 (Continued)		There were 2 failure mode theories: • The capacitor was physically too close to a screw hole used to mount a board stiffener. This hole could act as a heat sink during soldering that led to poor solder wetting. • Because the capacitor was glued to the board prior to soldering, it was conjectured that the adhesive expanded and contracted during thermal cycling causing stress at the solder joint. Final disposition of this problem was made prior to STS-35 flight. It was confirmed that there were 3 deflection amplifier pages in STS-35/OV-102 DUs which were from the suspect lot. CRT #1 DU had 2 suspect pages, and CRT #3 DU had 1. JSC SR&QA recommended changeout of CRT #1 prior to STS-35 because CRT #1 pages had over 17,000 hr of operation. However, JSC SR&QA also stated that redundancy of DUs was sufficient to cover a potential failure if CRT #1 was not replaced. The final decision was not to replace CRT #1. Rationale for STS-35 flight was: • Redundancy was sufficient to cover CRT failure. ***This risk factor was acceptable for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14	Ku-band broken lock motor wiring. HR No. ORBI-305A {C} ***No Ku-band anomalies were reported on STS-35.***	Ku-band antenna deployment assembly S/N 105 was undergoing testing when the lock jaws failed to fully unlock. The lock motor power wire going to the sequencing switch terminal board was broken at the exit point from the Room-Temperature Vulcanizing (RTV). This was the first occurrence of this failure mode. The concern was that deployment assembly S/N 106 on OV-102 had the wire bundle routed similar to S/N 105. Failure analysis found that the broken wire was caused by wire flexing during gimbal lock/unlock sequences together with inadequate strain relief. The investigation determined that the assembly drawing did not specify a preferred method for wire bundle routing. Retrofit of the lock motor resulted in the wire bundle being rerouted. A preferred method of routing the wire bundle was established; the assembly drawing was changed to clearly reflect this preference. If a similar failure should occur on orbit, the result would be failure to either lock or unlock the Ku-band antenna. In the event that the antenna could not be unlocked and deployed for the mission, communications and radar capabilities would be lost. The S-band and Ultrahigh Frequency (UHF) systems are backup in this case for communications, and ground and star tracking could be used in the event a rendezvous is required (none was scheduled for STS-35). If the failure resulted in the inability to lock the Ku-band antenna in place for reentry, the assembly would be jettisoned. Rationale for STS-35 flight was: • A video of S/N 106 during lock and unlock sequences was reviewed. No flexing of the wire bundle at the RTV interface was noticeable. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
15	Auxiliary Power Unit (APU) Gas Generator Valve Module (GGVM) issue. HR No. ORBI-031 {AR} ORBI-184 {AR} ***No APU anomalies attributed to GGVM failures were reported on STS-35.***	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. 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 Flight Readiness Review (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 the test provides some confidence that no PCV seat material is missing. Leak tests of STS-35/OV-102 APU/GGVMs was performed at the pad. The STS-35 MSE, L-2 Edition, May 28, 1990, identified concerns with APU #1, S/N 311, because it experienced the highest SOV wet time and PCV cycle time of the 3 APUs on OV-102. Because of the STS-35 launch delay, and due to a

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
15 (Continued)		shortage of APU/GGVM resources, S/N 311 was removed from OV-102 and installed on STS-38/OV-104. S/N #310, with new SOV and PCV valves installed, was designated for OV-102, APU #1. APU #3, S/N 310, was installed and was hot-fired at the pad. Rationale for STS-35 flight was: • STS-35/OV-102 APU/GGVMs did not exceed the life-limit criteria by the launch date. • Liquid leak tests were performed and provided additional confidence that no PCV seat material was missing. • LCC will not allow a launch with an APU in high speed. ***This risk factor was acceptable for STS-35.***
16	OV-102 GH_2 FCV power "on" violation. HR No. INTG-151 {AR} ***No GH_2 FCV anomalies were experienced on STS-35.***	Data review at KSC showed that, during post-tanking checkout, the STS-35/OV-102 GH_2 FCVs were energized longer than the OMRSD limit [were on approximately 10 hours (hr)]. This occurred during the ET flight pressure leak checks at the pad prior to rollback of STS-35/OV-102 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.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
16 (Continued)		The concern was for the health of the valve solenoid and the nitrite rubber O-ring. Continuous FCV O-ring operating temperatures range from −65°F 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. LH2 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 removed hardware was analyzed to determine if the unplanned energizing adversely affected the valve components. Results of this analysis indicated no adverse effects on the solenoids; however, O-ring resistancy was reduced. Rationale for STS-35 flight was: • All OV-102 GH_2 FCV O-rings and solenoids were replaced. ***This risk factor was resolved for STS-35.***
17	OV-102 FCL flow rate degradation. HR No. ORBI-275A {C} ***No further FCL #1 flow degradation was experienced after the launch of STS-35. On-orbit FCL operations were nominal.***	FCL #1 flow rate had steadily decreased since OV-102 power-up at the pad. In 1983, both FCLs decayed by approximately 100 lb/hr to below the OMRSD lower limit. The FCL pump inlet filters were replaced; minor contamination was found. FPM inlet filters were also replaced; holes in the filters due to braze material contamination were found. In July 1989, FCL #2 payload flow rate decreased below the OMRSD lower limit. The Radiator Flow Control Assembly (RFCA), FPM, flow meters, and pump inlet filters were replaced. Examination of the filters revealed aluminum alloy, brass, and stainless steel particles, plus oxidation products and some non-metallics. In May 1990, FCL #1 interchanger flow rate dropped

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
17 (Continued)		below the OMRSD lower limit with the FPM in the payload position (see this Section, Orbiter 10). The FPM was replaced. The inlet filter was found to be corroded and broken. (See STS-28 MSE Postflight Edition, November 14, 1989, Section 4, Orbiter 5 for further details.) This problem only affected flow rate prelaunch and on ascent, because the restriction was at the radiator bypass valve inlet filter. During countdown and until the payload bay doors are opened, the normal configuration is to bypass the radiators. When radiator flow is initiated, the bypass valve is changed to the radiator flow position; the controller positions the FCV to adjust the radiator outlet temperature. FCL criticality is 1R2. However, this failure did not result in total loss of flow in the loop. LCC flow rate requirements for launch would be met with the current rate of decrease in flow rate and minimum FCL #1 on time. The FPM and the FCL pump package filters were replaced. The inlet filter was found to be corroded and broken. The FCL was not backflushed to remove all remaining contamination. This resulted in clogging of the bypass valve inlet filter on the RFCA. Options for STS-35 launch formed the rationale for STS-35 flight and were: • Normal (Bypass) Configuration -- The STS-35 FCL #1 LCC flow rate was reduced to 1800 lb/hr (approved at the November 21, 1990 Level II PRCB). Recent analysis showed that at 1000 lb/hr (with the other loop flow normal) a 5°F margin remained before component overheating during ascent. If the flow degradation rate remained constant, the LCC would be met.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
17 (Continued)		• Bypass Valve to Radiator Position -- Radiators would be at approximately 100°F. Payloads worked to determine if there was a thermal constraint. • "Alternate" Bypass Mode -- The Mission Operation Director worked on a procedure to set the radiator FCV to the full-hot position and turn off the controller, then position the bypass valve to "radiator". This bypasses the radiator by flowing through the FCV instead of the bypass valve. Any of these 3 options were acceptable under the present circumstances; the FCL was not a constraint to launch unless the LCC limit was reached. ***This risk factor was acceptable for STS-35.***
18	Wing struts below minimum wall thickness. HR No. ORBI-277 {C} ***No anomalies attributed to wing struts were reported on STS-35.***	Postflight inspection of OV-103 after STS-31 revealed a damaged strut tube in the LH wing. This particular strut was replaced. Failure analysis showed that the buckled area occurred in a portion of the tube that was below minimum wall thickness (0.012" instead of the required 0.018" minimum). RI believes that the buckling was caused by outside forces and was not due to flight loads. 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 that had 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. 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

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
18 (Continued)		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-103 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 minimum), then their margins of safety were positive. The thin wall areas were believed to be due to improper chemical milling. RI is continuing the investigation into the origin of this problem. Rationale for STS-35 flight was: • OV-102 has experienced the highest loads of any Orbiter with no strut failures. • Margins of safety are based upon design loads; actual loads are less. • OV-102 wing components are sturdier than the other vehicles. • A buckled strut would still carry the load. ***This risk factor was acceptable for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
19	OV-102 air lock dents. HR No. ORBI-077 {AR} *No anomalies attributed to the air lock damage incident were reported on STS-35.*	During recent work on the Payload Specialist #1 seat, seat position #6, the headrest was accidentally dropped. The headrest landed in the air lock where inspection found 6 dents in the wall. Two of the 6 dents were superficial, and 3 others were up to 0.014" deep. The worst dent was determined to be 0.52" deep and needed to be repaired prior to launch. The other 5 dents will be repaired after STS-35. The worst dent was repaired with an adhesive fill under a bonded 0.020" doubler. Analysis determined that installation of this doubler would secure the air lock wall against potential leaks. *This risk factor was resolved for STS-35.*
20	Outer ring movement on Preload Indicating (PLI) washers on LH_2 Orbiter/SSME interface on OV-103. HR No. ORBI-306 {AR} *No anomalies attributed to loose PLI washers were reported on STS-35.*	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 #1 LH_2 line, 7 each on engines #2 and #3. 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 #3 line were still loose. Breakaway torque was also verified to be 300-600 in-lb on all engines. OV-104 was also examined during this investigation. Engine #2 F.1 joint was found with 1 loose outer ring, engine #3 with 2. All joints were previously leak checked with zero leakage reported.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
20 (Continued)		PLI washers are used to ensure proper bolt preload of 2500 lb (minimum). Washer assemblies at each bolt comprise a compressible inner and outer concentric ring sandwich between flat washers. The inner ring is crushed against the outer, impeding rotation when proper preload is applied. The 36 bolts are tightened in a star pattern in 25 in-lb increments. The outer ring is checked for movement/rotation before proceeding to the next bolt. However, there is no accurate procedure other than the technicians' touch for verification that the outer ring does not rotate. Stress analysis performed by RI/Downey determined that the calculated load for an outer ring gap of 0.0015" is greater than 2500 lb. Cryogenic effects were determined to be insignificant. Calculated bolt load at 360 in-lb torque is approximately 6000 lb. The investigation concluded that the OV-103 installation was 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-103 passed leak checks. OV-102 installation was accomplished with the same procedures as OV-103 and OV-104. All OV-102 joints passed leak checks with zero leakage recorded. PLI washer outer rings were verified not to rotate freely at installation, and the preload was maintained. Rationale for STS-35 flight was: • All OV-102 joints passed leak checks with zero leakage recorded. • PLI washer outer rings were verified not to rotate freely at installation, and the preload was maintained. ***This risk factor was acceptable for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
21	MPS joint weld issue. HR No. ORBI-306 {AR} ***No indications of LH_2/LO_2 leakage from MPS welded joints were reported on STS-35.***	H_2 leaks on OV-102 and OV-104 raised concern for the potential of undetected weld defects in MPS joints. Undetected defects 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. Because of weld defect concerns, a JSC/RI investigation was initiated. The 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 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 an LO_2 bleed line]. One 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 determined 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
<u>ORBITER</u>		
21 (Continued)		Given the above interpretations, fracture analysis was performed to determine the magnitude of the potential for flaw growth. A nominal flaw depth of 90% of 0.035" wall thickness was assumed. The analysis determined that flaw growth was not predicted for line exposure to proof-pressure tests and subsequent mission environments. Findings of the JSC/RI investigation did not identify generic concerns with welded MPS lines. Rationale for STS-35 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-35.***
22	Cracks in APU dynatube fittings. HR No. ORBI-103 {AR} ***No indications of APU dynatube fitting leaks were reported on STS-35.***	During acceptance testing of APU S/N 307, helium leak checks were performed. A leak, 2×10^{-6} standard cubic centimeters per second (sccs), was found at the GGVM bypass dynatube fitting. Acceptance test allowable leak rate is 1×10^{-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. This dynatube fitting was replaced prior to installation of S/N 310 on STS-35/OV-102. No cracks were found on 2 other APUs, S/N 312 and S/N 207.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
22 (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 which reduces the susceptibility to stress corrosion. APUs are helium leak checked prior to and after acceptance testing, as well as after initial installation in the Orbiter. Critical flaws are considered screenable by the helium leak checks. STS-35/OV-102 APUs had not experienced similar leaks during acceptance and post-installation testing. GGVM dynatube fittings are lockwired on installation. Rationale for STS-35 flight was: • The APUs are helium leak checked before and after the ATP. • The APUs are leak checked after initial installation in the vehicle. • S/N 310 dynatube fitting was replaced and properly leaked checked. No leakage was detected on the other OV-102 APUs, and there was no leakage history. • The design is failure tolerant; i.e., the dynatube fitting has a sealing surface outboard of the stress corrosion cracking area, and the fittings are lockwired in place. *This risk factor was resolved for STS-35.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
23	OV-102 20-psi helium regulator leak. HR No. ORBI-306 {AR} ***No indication of hydrogen leakage through the 20-psi helium regulators was reported on STS-35.***	During leak check activities following the first STS-35 launch attempt, 1 of 2 20-psi helium regulators was found to have a leak of 1 x 10^{-4} sccs. The regulator was removed 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 is on STS-37/OV-103 and has 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 plies 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 is 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 include reverse repressurization of the diaphragm and overstress during proof-pressure testing. Plastic deformation of the diaphragm is believed possible during proof-pressure testing. Because of this, the potential exists that all 20-psi regulator diaphragms are, at a minimum, wrinkled. 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 HGDS and would result in a scrub prior to launch. The regulator is used post-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

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
23 (Continued)		isolation valve is upstream of the 20-psi regulator. The isolation valve could be closed if the regulator failed open. Rationale for STS-35 flight was: • This was the first diaphragm failure in the program history. • OV-102 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-35.***
24	Fuel Cell (FC) separator plate plating defects. HR No. ORBI-282A {C} ***No FC anomalies were reported on STS-35.***	During recent refurbishment of FC S/N 109, plating blisters were found on 6 separator plates. These blisters are 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 recently found was attributed to the presence of microcracks in the nickel layer. Corrosion through 1 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

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
24 (Continued)		integrity. This corrosion site developed over a 31-month period of exposure 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. Corrosion pits could potentially develop, however, if material/magnesium impurities are present at the blister site. The concern associated with blistering was based on the potential for explosive mixing of H_2 and O_2 through the separator plates, resulting in loss of vehicle and crew. Indications 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-35/OV-102 FCs passed these tests. FCs used for the qualification test program operated for 2000 hr with no problems. A historical review found that FCs had successfully operated 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. STS-35/OV-102 FCs had less than 900 hr of total operation prior to flight. Investigation continues 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.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
24 (Continued)		Rationale for STS-35 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 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-35.***
25	Inertial Measurement Unit (IMU) #2 "power conditioner fail" Built-In Test Equipment (BITE) indication. HR No. ORBI-051 {C} ***No IMU anomalies were reported on STS-35. IMUs #1 and #3 were reported to have differing data on the first day; however, a successful attitude adjustment was made to correct the problem.***	Post landing on STS-41, IMU #2 indicated a "power conditioner fail" BITE 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 believed 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. This failure mode is Crit 1R3. There is no workaround; entry 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.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
25 (Continued)		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-35 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-35.***
26	Gaseous Nitrogen (GN_2) Shutoff Valve (SOV) weld failure on OV-105. HR No. ORBI-071 {C} ORBI-111 {C} ***No GN_2 SOV anomalies were reported on STS-35.***	A GN_2 SOV inlet fitting weld failed during installation on OV-105. Investigation into the failure determined that an improper installation technique had been 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 0.020" minimum required. Due to the lack of weld penetration, and 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

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
26 (Continued)		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, had 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-35 flight was: • This was not a generic problem and was limited to the OV-105 installation assembly. • The "double-wrenching" technique was not used on OV-105. • The valves are pressure tested before each flight. • The original GN_2 SOV lot, including those on STS-35, were cleared for flight. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
27	APU Gas Generator (GG)/Fuel Pump (FP) heater system "B" failed on during STS-41. HR No. ORBI-104A {C} ***No GG/FP heater problems were experienced on STS-35.***	During Flight Day (FD) 4 checkout of the STS-41/OV-103 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 min after switchover from system "A" heaters to system "B". 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 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, with a detonation temperature of 350°F, would detonate and result in APU fuel line rupture, hydrazine leakage, fire, and potential loss of crew and vehicle. Cycling APU GG/FP heaters off at 100°F is designed to protect against fuel line overtemperature. Troubleshooting at KSC determined that there was a short-to-ground between the fuel line heater and the water valve heater wires. This short was believed to be at a location where a clamp is used to secure the wiring to the fuel line. Further troubleshooting is in work to isolate the short. It was 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. 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-35 flow.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
27 (Continued)		Action was assigned to the Orbiter Project at the STS-38 FRR 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: • Prelaunch tests will be conducted during tanking for flight on all APU heaters. This test will include switching between "A" and "B" heaters. • There will be a new APU high-temperature FDA limit set at 150°F. The current limit is 180°F. This will provide an additional minute of response time to the crew. The ground monitoring system has been changed to alert the APU console operator when temperatures reach 130°F to enhance response awareness. • All APU reconfigurations will be performed in Acquisition of Signal (AOS) conditions only to allow ground monitoring of any failure. • On FD 1, heater reconfiguration will be performed early, at 6-hr Mission Elapsed Time (MET), to allow verification of system "B" heaters. • Any APU heater anomalies detected during Loss of Signal (LOS) conditions will result in the crew powering down all APU GG/FP heaters. Heater reconfiguration will follow for the failed heater, and the remaining heater strings will be reactivated. • To enhance response time, "booties" will be installed on APU heater switches for quick recognition. Additionally, the crew's orbit pocket checklist was updated to reflect crew response procedures in the event of an APU heater FDA.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
27 (Continued)		• Ferry heater cycle checkout is performed prior to the ferry flight. The heater configuration for ferrying was the same as the reentry heater configuration. Rationale for STS-35 flight was: • The above listed procedural changes were made in the event of an APU heater/thermostat failure. • No APU thermostats were changed during the STS-35 flow. • 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 MCC during operation. High-temperature limits were lowered to enhance response time. ***This risk factor was resolved for STS-35.***
28	LH_2 prevalve detent cover seal spring. HR No. ORBI-306 {AR} ***No LH_2 prevalve anomalies were reported during STS-35 countdown or ascent.***	During STS-35/OV-102 processing, a crack was found in the OV-102 prevalve #3 detent cover seal spring. The seal is a teflon jacket with an internal stainless steel spring made of 301 CRES. During the H_2 leak investigation, cracks were found on 2 adjacent spring webs; the webs were not continuous, and the cracks were self limiting. The cracks were believed to have been caused by H_2 embrittlement. No evidence of fatigue or corrosion was found. There was evidence of H_2 exposure and microtears that could have occurred during the forming of the spring. 301 CRES is usually not susceptible to H_2 embrittlement; however, this spring was harder than

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
28 (Continued)		normal. Five additional seals from OV-102 were inspected, and 1 partially cracked web was found. The concern was potential H_2 leakage into the aft compartment. The HGDS would detect excessive H_2 leakage during prelaunch. Additionally, the STS-35 tanking test data verified that MPS system leakage was well within the LCC limits. Rationale for STS-35 flight was: • STS-35 tanking test data verified that the MPS system leakage was well within the LCC limits. • The aft compartment HGDS would detect excessive leakage during prelaunch. ***This risk factor was acceptable for STS-35.***
29	Tire pressure problem. HR No. ORBI-018 {AR} ORBI-185 {C} ***No tire problems were reported after the landing of STS-35.***	The pressure on the RH NLG tire, S/N NWA-016, was projected, at the recorded leak rate, to decrease to the minimum allowable landing pressure as early as December 14, 1990. The OMRSD specification for landing is 310 psi for the NLG (OMRSD V51AGO.075). ***This risk factor was acceptable for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	SSME High-Pressure Fuel (HPF) duct weld proof-pressure test failure issue. HR No. ME-D3 (All Phases) {AR} *No SSME anomalies were reported on STS-35.*	SSME HPF ducts are manufactured by RI and proof-tested upon delivery to Rocketdyne, Canoga Park. One of these ducts recently failed the proof-pressure test by rupturing at a weld. Because of this issue, a complete review of the HPF duct fabrication pedigree was performed on STS-36/OV-104, STS-31/OV-103, and STS-35/OV-102. This resulted in removal and replacement of a High-Pressure Fuel Turbopump (HPFTP) on STS-36/OV-104. The Material Review Board (MRB) cleared all HPF ducts on STS-31/OV-103 and STS-35/OV-102. All build records and x-rays were reviewed, and no significant issues were identified. *This risk factor was resolved for STS-35.*
2	Titanium HPFTP inlet defects. HR No. ME-D1 (All Phases) {AR} *No SSME anomalies were reported on STS-35.*	A review of titanium HPFTP inlets was initiated following the recent HPF duct proof-pressure test failure at Rocketdyne, Canoga Park. (See this Section, SSME 1.) During the review, weld x-rays on the STS-36/OV-104 engines were evaluated, and defects were detected on the inlet of engine #2027, HPFTP #4008. Program Management directed that this pump be removed and replaced prior to STS-36 flight. To support the STS-35/OV-102 flight, the inlets of the HPFTP scheduled for flight on STS-35 were examined. This included HPFTP #4008 that was removed from STS-36/OV-104. The defects on HPFTP #4008, along with a suspected defect on HPFTP #6009, were removed and repaired by polishing. All build records and x-rays were reviewed, and the MRB cleared all STS-35/OV-102 HPFTPs for flight. *This risk factor was resolved for STS-35.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
3	Heat exchanger weld contamination (generic). HR No. ME-B3 (All Phases) {AR} ***No SSME anomalies were reported on STS-35.***	During leak testing of a new powerhead build, a crack was found in the heat exchanger coil. The leak was first detected at weld #2 at a rate of less than 0.0002 cubic centimeter per second (cc/sec), 1-to-2 orders of magnitude higher than previous experience. It was later confirmed that the leak was in the Haynes 188 fitting side of weld #2, 0.010" in the heat-affected zone. A crack, 0.060" long at the inner diameter and 0.025" at the outer, was found to be the source of the leak. Propagation of the crack occurred under additional strain, probably during handling or adjacent heating. This heat exchanger had previously passed the heat exchanger coil proof test. Investigation found evidence of a high-temperature fracture at the surface. Auger analysis revealed a small amount of tin at the fracture surface. The cracked weld, along with another weld, had been identified during fabrication as being wider than normal welds. A wide weld is indicative of significant heating or rewelding. It was noted that this heat exchanger coil was one of the first coils to be welded at the new Wintec facility and new equipment/procedures were considered to be a factor. The coil was installed in the heat exchanger liner assembly at Rocketdyne. A leak was detected at Rocketdyne near weld #2 during normal liner assembly mass spectroscopy tests, and it was isolated by removal of the sleeve. Initial fabrication x-rays and dye penetration tests did not detect the crack. Subsequent to finding the leak, single-wall x-ray did detect the crack. Metallurgical failure analysis found the crack to be in the weld heat-affected zone, approximately 0.014" from the fusion line. Tin and zinc were also found. Investigation and analysis of preweld tools, fixtures, solvents, and procedures revealed no source for elemental tin and zinc. Silicon molds used to verify final inner diameter weld width were found to have tin and zinc as constituents of the mold catalyst. Twenty percent of all outlet welds that did not meet minimum weld width are rewelded to within requirements after a deionized water rinse and isopropyl alcohol swab. Weld width data from 92 outlet welds were examined. The

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
3 (Continued)		cracked fitting was found to have 2 pre-ream molds and was rewelded; the widest documented weld had 3 pre-ream molds and was subsequently rewelded. Inspection of 41 additional welds from 21 units, using dye penetrant, found no additional anomalies. A series of laboratory tests concluded that contaminants promoted crack propagation only when plastically deformed at elevated temperature (>2000°F). The failure investigation determined that the crack was due to Liquid Metal Embrittlement (LME) caused by the combination of stress, heat, and contamination from the weld mold. Fracture mechanics analysis determined that the heat exchanger coil green run tests would screen for LME. Rationale for the flight of STS-35 heat exchangers was: • Green run tests will screen for cracks; if it doesn't fail, it will have an indefinite life. All OV-102 heat exchangers passed multiple proof, acceptance, leak, and hot-fire tests. The brittle nature of the fracture is expected to be detectable by testing. Critical flaw size in embrittled material can be screened by 1 hot-fire test. • Low likelihood of producing cracks; 41 welds from 21 units were inspected and did not have cracks; 43 weld samples were prepared with no cracks. • Review of OV-102 heat exchanger fabrication records found no indication of wide welds and only 3 rewelds; all were within specification and were leak checked. • There was no known contamination or cracks found within the history of 50 hot-fired heat exchanger coils. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
4	Fuel Preburner (FPB) LOX post fatigue. HR No. ME-B2 (All Phases) {AR} ***No SSME anomalies were reported on STS-35.***	Eddy current measurements on engine #2030 increased with LOX post support pins installed. Failures in engines #0204 (in 1981) and #0009 (approximately 10 years ago) were attributed to High-Cycle Fatigue (HCF) cracking (austenitic-to-martensitic transformation under high strain conditions). Offset fuel feed holes increased the flow-induced loading. LOX post support pins were installed to arrest the damage sensitivity as the reading increase indicated transfer to martensitic. Engine #2030 FPB injector was hot-fired for 14 starts and 4210 sec operating time without support pins. Elements in engine #2030 were from the same material lot as elements in engines #0204 and #0009. Pins were installed during overhaul prior to operation on engine #2030 to arrest further damage accumulation. Acceptability of engine #2030 was verified by margin testing on engine #2206 (tested to over 20,000 sec). An FOS of 2.0 was demonstrated. Rationale for STS-35 flight was: • Extensive hot-fire time on other units with support pins installed produced no concern. • There was no history of preburner LOX post failure and no occurrence of damage. • OV-102 engines had pins installed since the original fabrication and no problems were experienced during 17 tests and greater than 4600 sec of operation. • This was not an issue for STS-35 because all 3 engines were pinned from the start. ***This risk factor was resolved for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
5	Engine #0213 spark igniter potential failure. HR No. ME-B-2S {AR} ME-C3S {AR} ***No SSME anomalies were reported on STS-35.***	Engine #0213 hybrid igniter failed on the B1 test stand at Stennis Space Center (SSC) during mainstage. This failure mode did not preclude satisfactory performance of the redundant 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 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 an HCF fracture. Stress analysis showed susceptibility to 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 Flight Readiness Test (FRT). This failure was attributed to moisture on the igniter tip. Rationale for STS-35 flight was: • A benign environment exists during the initial start phase. During this phase there is low vibration, and engine ignition occurs by 0.6 sec. • Redundant igniters are in all chambers. • 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-35.***

C-2

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
6	Low-Pressure Fuel (LPF) duct tripod cracks. HR No. ME-D3 (All Phases) {AR} ***No SSME anomalies were reported on STS-35.***	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 has experienced 69 starts with 18,527-sec total run time; 12,414 sec of operation were 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, had 89 starts and 31,705-sec run time with 12,517 sec of operation at or above 104%. S/N 4917969 had one leg break 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. Tripod leg cracks found to date were only on LPF ducts with high operating time above 104%. This indicated that the legs may be more sensitive to fatigue at operations greater than 104%; however, the data base to support this conclusion was not substantial. Engine dynamic loads are a function of (Power Level)^N, where $N = 0.5$ to 2 (as N increases, the percentage of fatigue damage assigned to high-power level operation increases). Correlation of failures seen to date indicated $N = 2$. The current Deviation Approval Request (DAR) limit for the LPF ducts is based on $N = 0.5$. OV-102 SSME duct tripods have less than 200 sec operation at greater than 104%.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
6 (Continued)		Hot-fire tests on engine #0213 indicated that the tripod leg failures were due to resonance with HPFTP synchronous frequencies. LPF ducts were instrumented with strain gages, accelerometers, and frequency pressure transducers for this hot-fire test. Test procedures require ramping the engine power level from 65% to 104%, increasing 1% every 5 sec. For power functions of N = 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; HPFTP at 600 Hz that approached the duct resonance of 615 Hz. Here, there was a high response potential at flex joint "C" at higher speeds. The failure mechanism was 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-102 SSMEs were considered safe for flight based on low exposure to 104% or greater operations. An effort is underway to redefine the life-limit and inspection criteria for future LPF duct use. Rationale for flight of LPF ducts on OV-102/STS-35 was: • Duct failures to date had greater then 12,500 sec of hot-fire operation at 109% power level; maximum OV-102 duct operation at greater than 104% was 200 sec. • All OV-102 HPFTPs had synchronous frequencies less than 590 Hz at 104%; below the LPF resonant frequency. ***This risk factor was acceptable for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
7	Pogo precharge pressure software anomaly. HR No. ME-C1C Rev. A {AR} ***No SSME anomalies were reported on STS-35.***	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 Fault Identification (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-35 flight was: • A patch to AR01 fixed this problem for STS-35. ***This risk factor was acceptable for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
8	High-Pressure Oxidizer Turbopump (HPOTP) bearing etching/corrosion issue. HR No. ME-C1 (All Phases) {AR} ME-C2 (All Phases) {C} ***No SSME anomalies were reported on STS-35.***	HPOTP bearings 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 had misinterpreted the drawing requirements and allowed etched bearings to be installed in HPOTPs. A total of 107 bearings were identified as having been processed through this inspector so far. Forty of the 107 bearings were designated as turbine-end bearings; 30 were inspected with 2 etched and 28 unetched. The remaining 67 were designated as pump-end bearings; 39 were inspected with 21 etched and 18 unetched. Only 5 suspect bearings were installed on pumps. There were 2 each in HPOTPs on engines #2031 and #2026 at the KSC engine shop. The fifth was determined to be installed in bearing position #2 on HPOTP #9309R1 on engine #2027, ME #3 on STS-38/OV-104. There were no suspect bearings on STS-35/OV-102. Material analysis of an etched bearing was performed to determine the worst-case effects. Dissection found the average etch depth was 0.0002" with the deepest at 0.0005". Stress analysis indicated that an etch depth of 0.001" 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 was of concern because it could 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 OV-104 engines were installed using this new process. The source of the chloride contamination is still under investigation.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
8 (Continued)		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. HPOTP #9309R1 will be torn down and inspected following STS-38. Rationale for STS-35 flight was: • No suspect bearings were installed on STS-35/OV-102 HPOTPs. ***This risk factor was resolved for STS-35.***
9	Rigid fuel bleed duct stress corrosion cracking, engine #2029. HR No. ME-D3 (All Phases) {AR} ***No SSME anomalies were reported on STS-35.***	A leak was detected at the rigid fuel bleed duct insulation expansion joints during engine #2029 post-acceptance test leak checks. The leak was isolated to a 0.1"-long stress corrosion crack. Contamination embedded in the crack near the outer surface was determined to be grinding wheel constituents (silicon, calcium, and sulfur). There was a second stress corrosion crack found adjacent to the first, measuring 0.020" long, that had similar contamination. There was no evidence of fatigue propagation in the through crack. The outer surface of the rigid fuel bleed duct showed signs of mechanical polishing and etching. There was no other corrosion or cracks found in the duct. Engine #2029 rigid fuel bleed duct was initially fabricated in April 1982. At that time, there were no corrosion inhibitor requirements for the duct. Stress corrosion cracking in a low-pressure fuel duct found in 1985 led to a requirement for duct inspection and the application of corrosion inhibitors to all insulated 21-6-9 CRES ducts. This requirement became effective in 1989 with STS-28 engine ducts. Engine #2029 rigid fuel bleed duct was refurbished in March 1988. At that time, a

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
9 (Continued)		visual and dye-penetrant inspection found corrosion pitting near the duct mount bracket, near the location of the recent leak. Analysis of surface replicas revealed a stress corrosion crack at one pit and recommended sanding and further inspection. Material review disposition specified blending to remove pitting, etching, penetrant inspection, and verification of duct wall thickness (> 0.027"). Stress corrosion was not specifically addressed in this disposition, and no leak check was required. It was concluded that the stress corrosion crack and leak path existed during the March 1988 refurbishment. The leak resulted from a 0.10" long stress corrosion crack with intermittent corrosion. An adjacent partly-though 0.20"-long corrosion crack was similarly contaminated. No other corrosion or cracks were found in the duct. Contamination induced by the blending process (silicon, calcium, and sulfur grinding wheel constituents) prevented discovery of the stress corrosion crack during subsequent inspections. Engine #2029 flew on STS-30 and STS-34, and it is believed that the leak was either too small for detection or it was contained by the duct insulation. The finding of this refurbishment discrepancy led to the review of all flight duct material review dispositions that identified corrosion. Six flight units were dispositioned for rework due to corrosion in a similar manner as the engine #2029 rigid fuel bleed duct. Three were identified on STS-35/OV-102 engines: a low-pressure fuel duct and rigid fuel bleed duct on engine #2012, and an articulating fuel bleed duct on engine #2028. Only the rigid fuel bleed duct on engine #2012 was determined to be unacceptable for flight and was replaced. No corrosion was identified during refurbishment of the replacement duct.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
9 (Continued)		Rationale for STS-35 flight was: • All STS-35/OV-102 insulated 21-6-9 CRES ducts were inspected for corrosion and refurbished with corrosion inhibitor. • Corrosion was identified and removed during refurbishment of the engine #2012 LPF duct and engine #2028 articulating fuel bleed duct. This was verified by inspection. • Engine #2012 rigid fuel bleed duct was replaced. No corrosion was identified during refurbishment of the replacement unit. • All engines passed the helium signature test (insulation closeout was removed on engine #2012 LPF duct and engine #2028 articulating fuel bleed duct). ***This risk factor was resolved for STS-35.***
10	HPF duct flange radius cracking. HR No. ME-D3 (All Phases) {AR} ***No SSME anomalies were reported on STS-35.***	Sustained load cracking in the HPF duct flange fillet radii was first identified in 1986. There are 2 flanges on the duct; at the F-4 joint, the interface between the duct and the HPFTP, and at the F-5 joint between the duct and the main fuel valve interface. The basic cracking mechanism was attributed in 1986 to hydride formation from hydrogen diffusion within the basic 5AI-2.5Sn titanium material in regions of high sustained tensile stress. Sustained tensile stress was reduced by lowering preload at each joint. Recently, cracks were found on an HPF duct flange that was installed with lower preload. These cracks were found during post-proof test inspection after the duct insulation system was repaired.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
10 (Continued)		Evaluation of the cracked duct determined that crack formation at the flange was consistent with that associated with high sustained tensile stress. There were numerous initiation sites within the flange fillet radii. Most cracks were 0.010" long; however, some cracks joined in a stair-step pattern to a length of 0.025". No machining tears or anomalous surface conditions were identified. To determine if HPF duct flange cracking exists, OMRSD requirements direct preflight and postflight dye penetrant examination. Preflight examination must occur within 45 days of flight. The 45-day interval is based on maximum sustained crack growth for 90 days and engine testing with cracked ducts. Crack propagation was not demonstrated during hot-fire testing of 4 HPF ducts with flange cracks. Ducts with identified cracks during the preflight dye penetrant examination were removed prior to flight. Rationale for STS-35 flight was: • All HPF duct flanges on STS-35 SSMEs passed the preflight dye penetrant examination. • Dye penetrant examination is adequate to detect cracks that could affect flange preload or duct performance. • If undetected cracks exist, no adverse effects are anticipated based on hot-fire tests with cracked duct flanges. *This risk factor was acceptable for STS-35.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
11	HPOTP first stage turbine disc cracking. HR No. ME-C1 (All Phases) {AR} ***No SSME anomalies were reported on STS-35.***	Dye penetrant inspection of a high-time, developmental HPOTP first-stage turbine disc 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 the cracks were found, was the fleet leader with 21,908 sec and 52 starts; it had been removed from the flight program for a long time. 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. SEM inspection of the fractures indicated a crystallographic appearance. The fracture mode showed the effects of H_2 influence that indicated probable Low-Cycle Fatigue (LCF) or sustained load crack propagation. Structural analysis indicated a cyclic strain range overwhelmingly dominated by thermal shock at shutdown caused by H_2 cooling of the hot disc. Peak strain was determined through tests to follow a minimum of 40 to 100 sec of operation, or when the disc reaches steady-state high operational temperature. Evaluation of the correlation of LCF analysis to this failure mode indicated that the worst-case thermal shock strain range is insufficient to result in cracking without H_2 embrittlement. Rationale for STS-35 flight was: • The cracked first-stage turbine disc was the first seen in the program. • HPOTPs on OV-102 had less than 33% of the total cracked disc time, mainstage starts, or starts greater than 80 sec prior to flight. • There was an extensive, safe HPOTP operational history compared to OV-102 HPOTPs; >44 discs with more starts and operating time, >24 discs with twice the starts and operating time. ***This risk factor was acceptable for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
12	Preburner fuel duct failed during proof test. HR No. ME-D3 (All Phases) {AR} ***No SSME anomalies were reported on STS-35.***	A preburner fuel duct, Part Number (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 is made from INCO 903 seam-welded tubing. 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 a problem with the 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 the 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 were 115 to 120 INCO 903 constructed ducts in the inventory. A tube assembly on engine #2026 was determined to have come from the same heat treatment lot as the failed preburner fuel duct. Heat treatment of this tube assembly was verified. All INCO 903 parts on STS-35/OV-102 engines were inspected and tested for proper heat treatment and hardness. This effort was completed with no problems found.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
12 (Continued)		Rationale for STS-35 flight was: • All suspect INCO 903 parts were verified to have proper heat treatment and hardness. *This risk factor was resolved for STS-35.*
13	HPFTP S/N #4405R1 liftoff seal anomaly during a 10,000 (10K)-sec certification run. HR No. ME-D1 (All Phases) {AR} ME-D3 (All Phases) {AR} ***No SSME anomalies were reported on STS-35.***	Post-test data review of HPFTP S/N #4405R1 10K certification test run #11 indicated that a liftoff seal behaved abnormally. Further review found that indication of this condition was seen in test run #8 data and subsequent runs. The component drain line temperature was recorded to be near ambient instead of the normal range of 250° to 300° Rankine (R). The data review also indicated that the liftoff seal tried to close during the HPFTP S/N #4405R1 operation. Indications were that hot gas reacted with the coolant flow to form an ice plug that blocked the drain plug. These indications were similar to those witnessed prior to the engine #0208 test stand failure in 1982 that destroyed the engine. If a failure similar to the engine #0208 incident occurred in flight, it would result in a catastrophic event. In the case of the HPFTP S/N #4405R1, however, there was no significant change in coolant liner pressure as was experienced during the engine #0208 failure, and HPFTP S/N #4405R1 performed nominally through the test. HPFTP S/N #4405R1 was sent to Rocketdyne and was disassembled. The failure was attributed to a cracked plug weld which allowed cryogenic H_2 to get into the liftoff seal. This leak increased backpressure to the liftoff seal, allowing it to cycle open and closed. The cracked weld was a Class II weld, requiring only visual inspection and dye penetrant for acceptance. Indications were that this crack was not fatigue related. Pump bearings were found discolored (normal), and there was no damage either to the bearings or bearing cage. Evidence of pitting was found in the pump case from liftoff seal rubbing as it repeatedly opened and closed.

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
13 (Continued)		The investigation into this anomaly determined that the cracked plug weld leading to liftoff seal cycling is a Criticality 3/3 failure mode and would not result in the catastrophic event experienced with the engine #0208 failure. Rationale for STS-35 flight was: • The test firing was completed with no decrease in HPFTP or engine performance. • There were no similar anomalous indications in any flight pump data, including those HPFTPs on STS-35/OV-102 engines. • The 10K HPFTP anomaly is a Crit 3/3 failure mode. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1	Linear Shaped Charge (LSC) lot acceptance test failures. HR No. B-70-16 Rev. B {C} ***LSC functioning was not required on STS-35.***	During lot acceptance testing of a new Range Safety LSC lot, lot AAV failed to sever 2 of 12 witness plates. The concerns were failure of an LSC to terminate the thrust of the Solid Rocket Motor (SRM), if required, and the failure of lot AAV implied a potential failure in lot AAT. Lot AAT was installed on STS-35 and was fabricated using the same process as lot AAV. Lot AAV failure was attributed to separation voids [voids approximately 0.010" wide were found between the explosive (HMX) and the copper sheath]. A failure investigation team found separations ranging from 0.008" to 0.015" between explosive and copper sheath on 10 of 10 LSC samples from lot AAV. Separations of 0.001" or less were found on 10 of 28 LSC samples from lot AAT. Two samples from lot AAV and 3 samples from lot AAT were fired in the LSC cord-level acceptance test configuration. All 5 samples had explosive/copper sheath separations; all samples severed the witness plate. In the LSC manufacturing process, the final forming was changed from swaging to rolling between lots AAR and AAS. The change was made to eliminate a cosmetic imprint left on the cord during the swaging process. The process change was qualified during delta qualification of lot AAS. It is now believed that the rolling process is not as effective as the swaging process. However, comparison of the witness plates of lot AAT versus AAV revealed that during the LSC cord-level acceptance test all 99 witness plates (1 per LSC segment) from lot AAT severed nominally. Severance of 165 witness plates (1 per segment) from lot AAV was less robust. Stress analysis results showed that the motor case will fail at 900-psi chamber pressure if the thickness is less than 0.275", and at 400 psi a through crack of 5.0" will propagate to the field or factory joint. Analysis indicated a large margin of safety for lot AAT based on the performance of the AAT lot acceptance test. Acceptance test severance for lot AAT was greater than 0.500". Additionally, a

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1 (Continued)		conservative analysis was performed of limited LSC cutting performance; the analysis showed that lot AAV with voids present had sufficient cutting capability to rupture the case. Rationale for STS-35 flight was: • Lot AAT was manufactured and tested as a separate lot from AAV. • Review of lot AAT build paper and acceptance criteria verified proper processing. • Visual and dimensional inspection, and nondestructive testing of samples and flight hardware, showed no anomalies for lot AAT. ***This risk factor was resolved for STS-35.***
2	Hydraulic system Quick Disconnect (QD) spring anomalies. HR No. A-20-04 Rev. C-DCN3 {C} B-20-09 Rev. C-DCN3 {C} ***No anomalies attributed to the SRB QDs were reported on STS-35.***	During visual inspection of a 3/4" QD spring sealing surface prior to installation, a crack was observed. This 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 of 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.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
2 (Continued)		Metallography analysis of the spring crack indicated the potential for formation of small particles and flakes. These particles and flakes could be released from the cracked surface and form contamination in the SRB hydraulic system. If a flake was released from the high-pressure fill QD, it could migrate to the hydraulic reservoir bootstrap cylinder and become trapped between the cylinder and wall. This could result in internal leakage; however, no hydraulic system performance degradation would result. A loose flake emanating from the low-pressure QD could migrate to the hydraulic reservoir low-pressure side and into the hydraulic pump. This condition would result in minor degradation of hydraulic pump operation, but within operating parameters. Materials and processing testing and analysis were performed in an effort to create a crack in a spring and to understand the failure mode. This included sustained load testing, cyclic load testing, elevated temperature salt water bath, and hydrogen sulfide environmental testing. Only the hydrogen sulfide environmental test created a failure. PH 17-7 cracking in a laboratory hydrogen sulfide environment was expected; it is a known phenomena from the literature. The only possible introduction of H_2 into QDs prior to delivery to USBI is chemical descaling of the spring prior to electropolish using a solution of nitric acid only as a backup method. There are no processing steps that introduce H_2 in the refurbishment processing evaluation. There are no operations or interactions in flight system processing that would introduce H_2 into QDs. A review determined that 1/4" QDs used in the SRB hydraulic fuel system were similar in design and manufacturing processes. Inspection of 116 1/4" QD springs found no indication of notches or cracks. Based on finding unacceptable 3/4" QD springs, all eight 3/4" QDs on STS-35 SRBs were removed and replaced. Three of the removed springs had notches. No

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
2 (Continued)		cracks were found on any removed QD spring; therefore, no contamination resulting from associated flaking was considered present in the STS-35 SRB hydraulic systems. Rationale for STS-35 flight was: • All suspect QD springs were removed from STS-35 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 is present. • Review of 116 1/4" springs revealed no notches or cracks. ***This risk factor was resolved for STS-35.***
3	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} ***No SRB hydraulic pump failures were experienced on STS-35.***	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 with the test equipment during the previous test operations. 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.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
3 (Continued)		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-35 flight was: • The failure, due to test operator error, is believed to be an isolated case. • STS-35 SRB hydraulic pumps passed initial proof-pressure acceptance tests without incident. However, there is 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. • Leakage was not detected during any of the refurbishment or acceptance checkout operations. The pumps on STS-35 had flown only one time. Therefore, the pumps on STS-35 had more than adequate fatigue life to safely support the STS-35 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-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
4	Actuator brackets found cracked during refurbishment of aft skirt. HR No. BN-05 Rev. B {AR} *No anomalies attributed to SRB actuator brackets were reported on STS-35.*	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 existed for some time. Aft skirt S/N 015 had not flown since 1985 (on STS-20) and was flown only one other time on STS-14. The actuator bracket, splice plate, upper ring, and splice fittings are 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). 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 gaping to occur at the fitting interface and torque loads applied during installation could cause 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. Stress analysis performed as part of this investigation found that flight environment loads are 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 USBI has 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-35 SRBs could not be inspected.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
4 (Continued)		Rationale for STS-35 flight was: • A splice fitting could fracture 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-35.***
5	Improper spotface on SRB Thrust Vector Control (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} ***SRB TVCs and CVFAs operated nominally on STS-35.***	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. Because of these finding, all CVFAs and piston accumulators in the fleet, including those on STS-38 SRBs, potentially had 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 result in an aft skirt fire. Investigation into the origin of this problem determined that it was the direct result of using an improperly sized spotface tool during CVFA refurbishment operation. The origin of the improper piston accumulator spotface has not been identified. The spotface tool cuts an O-ring bevel and shoulder for dynatube MS 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-35 SRBs, O-ring extrusion into the 0.020" gap was insufficient to violate the seal.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
5 (Continued)		CVFAs and other hydraulic system components on STS-35 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 the 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. MSFC Dynamics Laboratory assessed the potential for fitting displacement in the seal area due to vibrational loads to be negligible. It was also learned during this analysis that the fittings are lockwired in place after preload is applied. Rationale for STS-35 flight was: • All STS-35 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-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
6	SRB electrical cable integrity through the rooster tail. HR No. B-70-02 Rev. C {C} B-70-16 Rev. B-DCN4 {C} ***No SRB anomalies attributed to electrical cables in the rooster tail were reported on STS-35.***	During STS-35 SRB processing, the RH SRB electrical cable area known as the "rooster tail" (aft end of systems tunnel) was found to contain approximately 2 gallons of water. Several cables and connectors, including the Range Safety System (RSS), were under water. This problem was first observed during retest after a silver-zinc battery was used to replace a lithium battery in the RSS. During the retest, the RSS exhibited low signal strength. The problem was isolated to the rooster tail area. Upon opening the rooster tail area, approximately 2 gallons of water drained out. Visual inspection indicated that the water had been in the area for a long time. Water accumulation occurred because a vent/drain hole had not been properly reopened after the launch delay. The rooster tail area was dried out. The RSS connector, which was the only circuit that indicated a problem, was removed, dried out, and tested to be within specifications. Other connectors/cables in the rooster tail, including the watertight Operational Flight Instrumentation (OFI) cables, joint heater power cables, and the motor/joint Ground Environment Instrumentation (GEI) cables, were examined for water and corrosion contamination. The OFI cables were designed to be watertight, and no concern existed regarding the flight worthiness of these cables. An insulation resistance test was performed on the joint heater power cables, and the results were within specifications. The motor/joint GEI cable system was powered up for the last 2 months, and no problems/anomalies were identified. Additional confidence testing was conducted to verify the OFI system. The STS-35 LH SRB showed no signs of water.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
6 (Continued)		Rationale for STS-35 flight was: • There was no concern for Crit 1R watertight cables (environment within specification). • The RSS cables were dried out and tested to be within specification. • The joint heater power cables successfully passed insulation resistance testing. (Failure of this test would occur if there was water in the connector.) • The motor/joint GEI cables showed no signs of signal deterioration. • Reruns of previous OFI circuit tests showed no changes. ***This risk factor was resolved for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1	STS-35 potential SRM thrust imbalance. HR No. BC-06 Rev. B {AR} ***Thrust imbalance for STS-35 was reported to be within the CEI specification limits.***	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 Configuration End Item (CEI) specification limits. Waivers were written for Flight Sets #10, #11, and #12 to allow 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 on STS-35 and STS-38. Data relative to measured thrust imbalance during STS-31 indicated that it was in the range of 20,000 lbf. For STS-35, DR #400253 identified the worst-case thrust imbalance at the 103-sec time interval to be 143,000 lbf. ***This risk factor was acceptable for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2	Putty on inner gasket of test motors. HR No. BC-02 Rev. B {AR} BC-03 Rev. B {C} BI-02 Rev. B {C} ***No putty was found on the inner gaskets of STS-35 SRMs.***	Putty was found on the inner gasket of Test Evaluation Motor (TEM)-5 and TEM-6 at the postfire inspection. Putty in this location could impair gasket resiliency and allow blowby. The putty could also 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, Delta L-1 Update Edition, April 23, 1990, Section 4, SRM 4.) Rationale for STS-35 flight was: • The recommended LCC of 100°F and igniter heater setpoint of 110°F provide the required tracking factor with a potential TEM-5/TEM-6 type condition for the STS-35 as-built condition. The required tracking factor of 1.4 was met, and the tracking factor is 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 the bolt preload was not affected, an overfill condition was not created, and the seal crown footprint was unaffected. • The design is safe because redundant seals exist and function as designed (10 tests; SRM, DM-6, TPTA, JES, and NJES experience), STS-35 leak test results 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-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
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} ***A single blowhole through the igniter joint putty was experienced on both STS-35 SRMs. Again, there was damage to the cadmium plating and sooting. There was no damage to the Gask-O-Seal elastomer. This was an expected occurrence due to previous flight experience.***	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 with 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-31 MSE, L-1 Update Edition, April 23, 1990, Section 5, SRM 1.) A redesign effort on the igniter-to-dome joint is in work to delete the joint putty. An investigation team is working on changing the gasket retainer material from cadmium-plated steel to stainless steel. A Level II action item is pending for the SRM Project Office to review a design change to remove cadmium plating from the Gask-O-Seal. Putty blowholes were also experienced on STS-41 and STS-38 SRM igniter joints. These occurrences were within the SRM experience base. It is believed that the change in the putty layup procedure, and reduction of putty used to alleviate the problem described in SRM 2 above, led to the increased incidence of blowholes. Rationale for STS-35 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).

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
3 (Continued)		• 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-35.***
4	SRM igniter insulation FOS issue. HR No. BI-05 Rev. B {C} ***No anomalies attributed to SRM igniter insulation were reported on STS-35.***	Recent Thiokol analysis indicated that outer diameter insulation adjacent to the igniter nozzle insert did not meet the 1.5 safety factor against erosion on several previous flights. Deviation RDW-0565 was approved prior to STS-26 because the measured FOS at station 8 was 1.46 on static test motor Development Motor (DM)-9. Measurements on flight igniters since STS-26 revealed an FOS at station 8 of 1.07 on STS-29. Others also measured low. After the deviation was approved, the igniter chamber was thickened to allow the phenolic nozzle insert to meet the structural safety factor. Consequently, the insulation became thinner. The thermal environment imposed on the insulation at station 8 is non-erosive because it is subjected to igniter flow for approximately 0.4 sec. After that time flow stagnates, and the radiant heat from combustion gases during motor burn results in insulation charring with insignificant surface erosion. This phenomenon was confirmed by post-fire inspection of all test and flight motors. There were no reuse issues on the igniter chamber recovered from DM-9 or from any other static test or flight motor since STS-26. The question of measured safety factor lies in the accuracy of the postflight measurements. Postflight inspection at

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
4 (Continued)		KSC requires a visual inspection for irregular insulation erosion patterns or other abnormal indications. Char depth measurement is taken at KSC at station 8 adjacent to the nozzle insert. Once the igniter is returned to Thiokol, remaining virgin insulation thickness is measured at station 8 after removal of all char material. This is accomplished by drilling through the insulation to the metal chamber and using a depth gage. The angle of the drilled hole was determined to greatly affect the resulting measurement (records show some cases of postflight insulation thickness greater than preflight). The igniter chamber metal is then inspected for indication of heat effect after the insulation is removed. Results of these postflight inspections indicated adequate insulation performance. Inspection of igniter chamber metal showed no heat effect and indicated acceptability for reuse. Measurements of remaining insulation thickness, however, were inaccurate. This implied that the resulting calculated FOS was not meaningful. FOS calculations had ranged from a low of 1.07 to a high of 48. Analysis demonstrated that there was no safety-of-flight issue relative to igniter insulation at station 8. With the insulation char layer totally removed and the steel chamber exposed at 97 sec after ignition, the steel would reach melting temperature of 2800°F at station 8. There would be no resulting debris inside the motor, and igniter retention was not compromised. However, the igniter would not be reusable, and the effects would be easily detected by missing material and bluing. Rationale for STS-35 flight was: • STS-35 prefire insulation thicknesses were within the SRM data base. • Igniter insulation erosion safety factors based on pre- and postflight measurements did not indicate a safety-of-flight issue.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
4 (Continued)		• Visual inspections of postfire insulation revealed no anomalous effects. • All SRM igniter chambers were successfully refurbished. ***This risk factor was acceptable for STS-35.***
5	Debris in the SRM nozzle flex bearing cavity. HR No. BN-06 Rev. B DCN64 {AR} ***No TVC problems were experienced on STS-35.***	Recent inspections 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 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 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. 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 occurred 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

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
5 (Continued)		ring with additional damage probable to the snubber segment. In the case where the bolt remains lodged through splashdown, the snubbers would be inhibited from performing their splashdown function, resulting in significant damage to nozzle components. The dry-fit bolt was removed from the STS-35 nozzle. Borescope inspection of STS-35 nozzle cavities was performed prior to launch with snubbers in place. No further foreign material was found. Borescope inspections of nozzle flex bearing cavities will be performed on all completed nozzles at KSC and Thiokol in the near future, and will be performed as part of a mandatory preshipment inspection for all future flight nozzles. Rationale for STS-35 flight was: • All known foreign material was removed from STS-35 SRM nozzles. • The STS-35 nozzle was returned to print and was safe to fly. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
6	Aft dome factory joint internal insulation voids. HR No. BC-10 Rev. B-DCN71 {C} ***STS-35 SRM aft dome factory joints have not yet been dissected to inspect for insulation voids. There was a terminated blowhole in the polysulfide insulation on the RH nozzle-to-case joint. While this was the first occurrence in a flight motor, it was not unexpected because of ground test experience and does not pose a joint performance concern.***	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. Because of these findings, aft dome factory joints from STS-36 and STS-31 SRMs were dissected. Similar insulation voids were identified. It was determined, however, that these SRMs, and others based on visual postflight inspections, met the erosion safety factor. Post-fire insulation samples were taken from all SRMs. These samples had small, entrapped air voids. None of these were considered to be folds, bulges, or thin spots in the insulation. Voids were always localized and were surrounded by rubber-tearing vulcanized bonds that did not propagate or communicate. Voids are in compression during motor operation. The minimum SRM insulation erosion safety factor over the aft dome joint was determined to be 3.46 based on post-fire evaluation of 400 insulation samples. It is relatively certain that all SRMs had similar aft dome factory joint insulation voids. 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 same insulation technique had 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.

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
6 (Continued)		Rationale for STS-35 flight was: • Post-fire 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-35 insulation was concluded to have a safety factor greater than 2.0 over all factory joints. ***This risk factor was acceptable for STS-35.***
7	SRM Ignition Initiator (SII) leak test. HR No. BI-01 Rev. B {C} ***No SII anomalies were reported on STS-35 SRMs.***	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. 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 that 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 period. Seal use history revealed that no leak test failures had occurred in RSRM motors to date. The current SII and seal configuration was used since STS-7 (282 samples demonstrate a reliability of 99.75 percent). All showed no seal violations or seal distress for this configuration. (Redesigned Solid Rocket Motor 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 hole in several SII disassemblies. Joint

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
7 (Continued)		seal design provided a high confidence that the seals would function properly. Testing showed shoulder seals provided reliable sealing at low torques. Tests on initiator pressure port plugs showed a similar configuration would seal even with anomalous assembly. Tests without the primary seal indicated the secondary seal functioned as designed. Rationale for STS-35 flight was: • STS-35 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-35.*
8	STS-41 Stat-O-Seal rubber damage. HR No. BN-03 Rev. B {AR} ***Similar damage was not experienced on STS-35 SRMs.***	Stat-O-Seals on STS-41 were found to be damaged at disassembly. Although grease was 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 also a common occurrence. Potential failure of igniter inner secondary seals is a Criticality 1R function. Rationale for STS-35 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-35.*

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ET</u>		
1	LH_2 recirculation line primer debonds. ***No problems were experienced with the STS-35 LH_2 recirculation line after a replacement line was installed.***	During 17" umbilical replacement after STS-35/ET-35 was first scrubbed, epoxy primer debonds were discovered on the LH_2 recirculation line. Post-tanking inspections of ET-35 and ET-37 found local primer debond at flange ends. Failure analysis revealed the most probable cause to be improper cleaning resulting from the geometry and fastener interference. A replacement recirculation line was installed on ET-35. Verification of primer adhesion was initiated on ET-35 and subsequent units. Thermal Protection System (TPS) was removed within 3" of each flange. Primer adhesion was verified by a wet tape test. Any debonded primer was removed and replaced. Pre-molded Super-Light Ablator (SLA) panels were bonded and closed out with BX250 foam. Rationale for STS-35 flight was: • Wet tape test confirmed good primer adhesion on ET-35. Material data base showed SLA on primed stainless steel that passed the wet tape test would meet cryostrain requirements with an FOS >2. • Repairs successfully passed acceptance tests. • TPS closeout was accomplished per a validated process. The TPS configuration provided adequate heating and ice/frost protection. ***This risk factor was resolved for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ET</u>		
2	Foreign object found on ET-35. HR No. T.02 {AR} ***No anomalies attributed to the metallic slag found on ET-35 were reported on STS-35.***	During the normal KSC weekly walkdown inspection of STS-35, a piece of metallic slag with evidence of a caulking-like material (facility type material) was found embedded in the ET-35 foam insulation. The metallic slag dimensions were determined to be 1-3/4" x 3/8" x 1/8". The depth of penetration into the foam was measured to be 0.21". Since there is a minimum requirement of 1" for the ET foam insulation, no rework was required. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
PAYLOAD		
1	Broad Band X-Ray Telescope (BBXRT) failure could lead to loss of vehicle. ***No problems with the BBXRT Argon system were reported on STS-35.***	The BBXRT Project identified a new failure potential resulting in liquification/ gasification of argon and leading to possible loss of vehicle and crew. The concern was that there was a failure mode which could result in excessive liquid argon spillage resulting in loss of BBXRT structural integrity. This structural degradation was critical because BBXRT would not be able to sustain reentry loads, possibly breaking loose and resulting in damage/loss of the vehicle and other payload elements. Argon was used to thermally stabilize the BBXRT lenses. It was stored as a solid at 30° Kelvin (K) to 78°K in 2 dewars in the BBXRT cryostat subsystem. When argon is frozen, the dewar inner tank had only a few torr (1 torr = 1/760 of an atmosphere) of pressure inside and no pressure on the outside due to the surrounding guard vacuum. During ground operations, vacuum pumps maintained the argon in the solid state condition. Redundant vent valves, controlled by 2 switches, provided a means to vent increased pressure to ensure that the solid argon did not liquify. However, the primary vent valves were normally closed, preventing a vent path if power was lost to the vent switch/valve. An emergency vent line was included in the BBXRT design to back up the primary vent path and provide redundant capability for venting the argon dewars. The emergency line would vent liquid/gas argon from the dewar if the internal pressure exceeded 20 psi. The BBXRT Project determined that an ice block could form in the emergency vent line and result in unstable pressure buildup in the dewar. Recent testing indicated that argon gas ice blockage was a credible and repeatable event; therefore, the BBXRT cryostat design was determined to be only single-fault tolerant. Meetings at Goddard Space Flight Center (GSFC) and JSC on May 4 and 5, 1990, resulted in identifying corrective action to resolve this issue. First, an approved revision to the STS-35 LCC required verification that the BBXRT argon was solid prior to launch. Specific dewar pressure and temperature limits to maintain solid argon were calculated. Second, the incorporation of 3 low-pressure 5-psi relief valves in the BBXRT argon line closeout caps was approved to ensure pressure is maintained in

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
PAYLOAD		
1 (Continued)		the argon system below the argon triple point. These caps were installed on the BBXRT. Existing relief mechanisms relieve at 20 psi above the argon triple point. As stated above, this could lead to freezing and ice blockage in the argon lines. It was determined that venting Argon into the payload bay, either when closed or open, would not result in a hazardous condition. Rationale for STS-35 flight was: • There was no concern with venting argon into the closed payload bay due to the projected small vent rate and quantity. • Low-pressure vent valves were qualified and installed. • LCC changes were incorporated to ensure solid argon at liftoff. • Multiple failures were required prior to entering a hazardous state; the BBXRT met the dual-failure tolerant requirement prior to launch. ***This risk factor was resolved for STS-35.***
2	Astro payload support structure/pallet assembly hardpoint lockwire issue. ***No structural anomalies associated with this issue were experienced on STS-35.***	During final walkdown inspection of the payload bay in the OPF, a technician noticed a nut without lockwire near a payload latch area. Evaluation of this situation determined that lockwire should have been applied to that nut. Further inspection confirmed the following: • 6 pallet-to-payload fasteners without lockwires. • 2 hut mounting nuts without cotter pins. • 4 optical sensor package nuts without cotter pins.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
		• 2 bonding strap bolts without lockwires (not structural). A processing paper review indicated that lockwires were not installed on the payload support structure/pallet assembly at hardpoints 8, 9, 16, and 17. Further investigation found that payload support structure assembly drawing notes specified either cotter pins or lockwire installation on all hardpoints; however, part numbers for hardpoints 8, 9, 16, and 17 were missing from the drawing. Because no part numbers were listed, the KSC assembly procedure did not include lockwire installation for these hardpoints. Visual inspection of all 7 hardpoints securing Astro to the pallet was performed. Hardpoints 5, 12, and 21, the other 3 of 7 used hardpoints for Astro, were lockwired per the assembly drawing. Lockwire was installed on hardpoints 9 and 16. Hardpoint 8 was accessible; however, the bolt lockwire hole and nut castellation groove were not aligned. Lockwire was installed only across the castellations on hardpoint 8. Hardpoint 17 was only accessible with one hand; therefore, lockwire installation could not be performed. A decision was made to epoxy the nut in place for this mission. Validation testing of the epoxy method for securing hardpoint 8 was performed, and the epoxy was deemed adequate. KSC payload operations implemented a system whereby separate line items are required in installation procedures for torquing and lockwiring. Flagging of all safety-critical items in the OMRSD is required; all procedures with flagged items require KSC payload management review and design center approval. This process was implemented late in the Astro flow.

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
PAYLOAD		
2 (Continued)		Rationale for STS-35 flight was: • Epoxy concept was demonstrated by validation tests to be adequate for hardpoint 17. • Lockwire installation on hardpoint 8 was determined to be acceptable for flight. • Other hardpoints had lockwires installed per drawing prior to launch. ***This risk factor was acceptable for STS-35.***

RESOLVED STS-35 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
GFE		
1	Unrestrained camcorder during STS-35 ascent/descent. HR No. CCH/T-001 {AR} ***No problems associated with use of the tethered camcorder were experienced on STS-35.***	The STS-35 crew requested authorization to use a Cannon camcorder during the STS-35 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 that 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 was used by the MS #1 who is located behind the commander. Crew members were taught proper and safe use of the camcorder tethering system in training exercises and briefings. The camcorder was used during orbit reentry and landing only. The camcorder, including the harness assembly and tether assembly, was stowed during launch. ***This risk factor was acceptable for STS-35.***

SECTION 5

STS-38 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-38/OV-104 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 risk factor in this section are listed beneath the anomaly title. Where there is no baselined HR associated with the anomaly, or if the associated HR has been eliminated, none is listed. Hazard closure classification, either Accepted Risk {AR} or Controlled {C}, is included for each HR listed.

SECTION 5 INDEX

STS-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 #3 did not initiate cooling of APU lube oil at the required temperature on STS-35 (IFA No. STS-35-17). This anomaly was attributed to the presence of wax in the lube oil instead of a problem with the controller.***	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 reach 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" determined replacement was required. Rationale for STS-35 flight was: • Redundant controllers exist. • Controllers are checked every flight. • OV-102 set points and temperature sensors were verified as functional. ***This risk factor was acceptable for STS-35.***

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2	Flash Evaporator System (FES) water supply accumulator heater system biased low. IFA No. STS-38-02 HR No. ORBI-276B {C} ***No FES anomalies were reported on STS-35.***	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 #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 was the suspected cause of this anomaly. FES heater #1 was removed and replaced at Kennedy Space Center (KSC). Rationale for STS-35 flight was: • This appeared to be an unique failure; not generic. • FES equipment was not disturbed or changed since STS-32. • Multiple levels of redundancy exist. A total of 5 failures are necessary to lose crew/vehicle: loss of one radiator coolant loop, radiator heater, or ammonia boiler, and loss of all four water heaters. ***Not a safety concern for STS-35.***

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 ***No anomalous accelerometer indications from APU instrumentation were reported on STS-35.***	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 included connector and accelerometer checkouts. APU #3 was removed and replaced due to life-time cycle limits and isolated the anomaly to a broken coax in connector 50P3 pin A. Rationale for STS-35 flight was: • APU integrity is monitored by many other measurements. ***Not a safety concern for STS-35.***

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Vacuum cleaner short circuit. IFA No. STS-38-04A HR No. ORBI-301 {C} *No problems with the vacuum cleaner were reported on STS-35.*	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. Prelaunch vacuum cleaner checkout will be modified to include a check for shorts on all 3 phases. The vacuum cleaner was removed and replaced with a stock vacuum cleaner that passed the new checkout procedures for shorting condition on all 3 phases. Rationale for STS-35 flight was: • The vacuum cleaner on STS-35 was checked for shorts in all 3 phases and functioned nominally. *Not a safety concern for STS-35.*

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	APU Exhaust Gas Temperature (EGT) instrumentation interaction with injector tube temperature instrumentation. IFA No. STS-38-05 HR No. ORBI-106A {AR} ***No anomalous injector tube temperature indications were experienced on STS-35 APUs. APU #2 injector temperature was biased approximately 50°F; however, this was a known condition prior to launch.***	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. Rationale for STS-35 flight was: • Ground loops were verified during turnaround. • Injector temperature measurements are utilized for hot restart. Injector temperatures are backed up by water tank pressure for hot restart conditions. ***Not a safety concern for STS-35.***

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	Right vent doors 1, 2 purge position failure. IFA No. STS-38-06 HR No. ORBI-178A {AR} *No vent door anomalies were reported on STS-35.*	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 vents #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. Troubleshooting isolated the failure to the Power Distribution Unit (PD). The PDU was replaced, and test indicates satisfactory performance. Rationale for STS-35 flight was: • Criticality for failure to purge is Crit 3/3 during nominal mission. *Not a safety concern for STS-35.*

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 R5D failed "off" indicating low P_c and was deselected by Redundancy Management (RM) on STS-35 (IFA No. STS-35-20). Investigation by the crew determined that helium was present in the crossfeed line of R5D. This condition is indicative of bubbles in the feedline. Manual firing of the thruster restored R5D to operational status.***	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 6-8 psi pressure decay in a 2-hr period. Post-decay mass spectrometer test identified no leakage. Troubleshooting of thrusters R1U, R3D, RF3L, and R4U degradation is complete with no further problems identified. Rationale for STS-35 flight was: • All STS-35 thrusters were inspected with no problems noted. • There is multiple redundancy; the system can tolerate 2 failures (Crit 1R3) during normal flight and 1 failure during abort modes (Crit 1R2). ***Not a safety concern for STS-35.***

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8	Continuous "tire press" Fault Detection and Annunciator (FDA) message following landing gear safing. IFA No. STS-38-08 HR No. ORBI-018 {AR} ***No similar anomaly was reported on STS-35.***	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 found no other problems. This anomaly was closed as unexplained. Rationale for STS-35 flight was: • There was no effect on tire performance. • Inflight confirmation of a leaking tire requires a decrease in both pressure measurements of one tire that is inconsistent with changes in pressure of the other Orbiter tires and cannot be explained by instrumentation inaccuracy or variations in temperature. ***Not a safety concern for STS-35.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
9	Transient smoke detector event indication anomaly. IFA No. STS-38-09 HR No. ORBI-300 {C} ***No smoke detector anomalies were reported on STS-35.***	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 CB 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. Rationale for STS-35 flight was: • Redundancy is provided by 2 smoke detectors in each avonics bay. • A false alarm is identified if there is no concurrent increase in smoke concentration or there is no alarm from the redundant detector. ***Not a safety concern for STS-35.***

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1	Unidentified debris observed between Mission Elapsed Time (MET) 26 seconds (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} *There were no similar observations on STS-35. Two pieces of instafoam were observed falling from the Right-Hand (RH) SRB aft skirt at liftoff; however, this is a known problem that the SRB Project is actively working.*	A review of launch films revealed debris exiting from both the Left-Hand (LH) and 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 conditions 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>		
1 (Continued)		Rationale for STS-35 flight was: • The aluminum tape is designed for ground-based functions. • No loss of thermal curtain function was witnessed on STS-38 SRBs. • Loss of viton-coated nylon is a normal flight occurrence. *Not a safety concern for STS-35.*

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} ***There was no report of lost Instafoam from the ETA on STS-35. However, Instafoam was seen falling from the RH SRB aft skirt at liftoff.***	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 process 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 is considered unlikely due to the location and density of the foam (3 - 4 lb/ft^3). Rationale for STS-35 flight was: • Instafoam is not required for thermal protection. • Debris damage is unlikely due to the location and density of the foam. ***Not a safety concern for STS-35.***

STS-38 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>MOD</u>		
1	Unexpected General Purpose Computer (GPC) #3 talkback indication. IFA No. STS-38-MOD-01 HR No. ORBI-066 {AR} *No GPC anomalies were reported on STS-35.*	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. Rationale for STS-35 flight was: • This anomaly was most likely a procedural error. • There was no evidence of hardware or software problems. ***Not a safety concern for STS-35.***

SECTION 6

STS-32 INFLIGHT ANOMALIES

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

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

SECTION 6 INDEX

STS-32 INFLIGHT ANOMALIES

SECTION 6 INDEX - CONTINUED

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1	Auxiliary Power Unit (APU) #3 lubrication oil outlet pressure high. IFA No. STS-32-02 HR No. ORBI-036 {AR} *APUs #2 and #3 had indications of wax in the lubrication oil on STS-35 (IFA No. STS-35-17 and STS-35-19). APU #2 will be removed and replaced due to life-limit constraints. APU #3 as well as Water Spray Boilers (WSB) #2 and #3 will require a hot oil flush to remove the wax.*	Higher than expected lube oil outlet pressure was observed on APU #3, STS-32/OV-102. The lube oil pressure was 75 to 95 pounds per square inch absolute (psia) for the first 10 minutes (min), then returned to normal (55-60 psia) at full operating temperature. The pressure was nominal for ascent and landing. This anomaly was similar to the APU #1 anomaly on STS-33 (IFA No. STS-33-01). It was determined that the high pressure was caused by the presence of hydrazine in the gearbox. Hydrazine reacts with the lube oil to form pentaerythritol and hydrazides which liquify at approximately 200°F. This contamination/wax buildup collects on the lube oil filter and causes partial blockage. Heating the oil causes the contaminant to liquify and clears the filter. Continued operation disperses the contaminant throughout the gear box. It is believed that seepage of hydrazine past the seals into the gearbox is due to the postlanding gearbox pressure being below the seal cavity pressure. The worst-case criticality is 1R2 for gross hydrazine leakage into the gearbox and loss of the APU. The Launch Commit Criteria (LCC) allows for lube oil outlet pressures up to 110 psia. The Flight Rules do not address lube oil pressure, and the temperatures were normal; therefore, this anomaly did not result in a violation of the Flight Rules. The most probable effect of wax buildup is an increase in lube oil outlet pressure for a period of 10 min, which does not affect operation of the APU. The solution to this problem is postlanding pressurization of the gearbox above the pressure of the seal cavity. Hot oil flushing techniques were developed which allow the wax to melt and the system to be flushed. A Requirements Change Notice (RCN) was issued to change the Operational Maintenance Requirements and

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)		Specifications Document (OMRSD); it included new procedures for postlanding pressurization of the gearbox to no less than 5 psia greater than the seal cavity pressure. APU #3 was removed; the system was flushed, sampled, and reserviced. ***Not a safety concern for STS-35.***
2	Right-Hand (RH) Orbital Maneuvering System (OMS) "no-back" device moved during ascent. IFA No. STS-32-04 ***No similar anomalies on were reported on STS-35.***	The RH OMS yaw actuator, Serial Number (S/N) 117, drifted 0.112° during the first 50 seconds (sec) after the launch of STS-32. The "no-back" device is designed to prevent back-drive or movement of the actuator during powered operations. After the initial movement, the "no-back" device held the actuator as required for the remainder of ascent. During entry, movement of 0.048° was recorded. Failure of the "no-back" device could result in positioning the OMS engine nozzle into the air flow. The dynamic pressure of the air flow will cause a nozzle to deform. Review of previous flights using this yaw actuator was completed. On STS-28, there was indication of 0.082° movement. The previous mission employing S/N 117, STS-61C, recorded movement of 0.098°. Movement in each of the 3 missions reviewed occurred during ascent when the highest vibration environment is experienced. Because the occurrences of movement during ascent were relatively consistent, there was no indication of degradation. Movement similar to that witnessed on STS-32/OV-102 was not considered detrimental to actuator function since no significant problem exists until 1.5° of movement occurs during launch. If movement occurs during major modes 102 and 103 of OPS-1 (ascent software), the software will automatically power-up the OMS

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)		controller primary channel if the gimbaled angle reaches 0.7° and commands the actuator to the stow position. In either software mode, an alarm will sound alerting the crew to manually select the backup channel. An RCN changed the acceptable limit for actuator movement to 0.2°. No troubleshooting of this anomaly was performed. *Not a safety concern for STS-35.*
3	Gaseous Oxygen (GOX) Flow Control Valve (FCV) #2 opened sluggishly. IFA No. STS-32-06 HR No. INTG-150A {C} ORBI-248A {C} ***No GOX FCV anomalies were reported on STS-35.***	The GOX FCV #2 exhibited sluggish operation when first cycled open at T+61 sec. This FCV took approximately 0.75 sec to open versus the nominal operation of 0.2 to 0.4 sec. Sluggish operation of this FCV was only experienced during the first open cycle; all other closed-open/open-closed cycles were nominal. Investigation of sluggish GOX FCV operation previously experienced found contamination as the cause of the slowness. This was the first instance of sluggish GOX FCVs since STS-29. All GOX FCVs on OV-102 were replaced with new reshimmed valves during the STS-35/OV-102 turnaround process. Poppet installation, leak and functional tests, and signature tests were completed. GOX FCVs are scheduled for replacement with fixed orifices.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)		Rationale for STS-35 flight was: • FCVs are redundant; sluggish action of one FCV will not affect External Tank (ET) ullage pressure. • Most likely cause of sluggish behavior was transient contamination. • No FCV had failed open or closed to date. • FCVs on STS-35/OV-102 passed all turnaround tests and leak checks. *Not a safety concern for STS-35.*
4	Humidity separator "B" water bypass. IFA No. STS-32-07A HR No. ORBI-254 {C} ORBI-321A {C} *A small amount of water was found near the air outlet of humidity separator "B" on STS-35 Flight Day (FD) 6. It was decided to allow the water to air dry. Followup inspections found no additional water for the remainder of the mission.*	During changeout of the Lithium Hydroxide (LiOH) canister, the crew discovered uncontained water. The LiOH canister was wet, and water was coming from the humidity separator "B" exit port. At the time, humidity in the cabin was nominal, and there was no indication of a humidity separator problem. The crew switched from humidity separator "B" to "A" and initiated free fluid cleanup procedures. Approximately 2 gallons of water was collected. The separator was removed and sent to the vendor for failure analysis. Some "white residue" was found on the heat exchanger outlet. The replacement humidity separator was installed and retested. Crew procedures were available should a problem occur during STS-35 flight, and the separators are redundant. Furthermore, the mission could be terminated early if other efforts to control cabin humidity fail. The cause of the failures was believed to be due to high humidity and an accumulation of debris on the condensing heat exchanger. *This anomaly was not a safety concern for STS-35.*

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	Humidity separator "A" water bypass. IFA No. STS-32-07B HR No. ORBI-254 {C} ORBI-321A {C} ***A small amount of water was found near the air outlet of humidity separator "B" on STS-35 FD 6. It was decided to allow the water to air dry. Followup inspections found no additional water for the remainder of the mission.***	On FD 6, the crew found approximately 8 ounces (oz) of water around humidity separator "A". Inflight maintenance procedures (towel and bag) were initiated to collect additional water that might escape from humidity separator "A". On FD 7, the crew reported more water coming from humidity separator "A". Cleanup procedures were again initiated, and approximately 2 cups of water were collected. It was postulated that this water escape resulted from the high level of crew activity. Separator "A" leakage is most likely due to carryover of water from humidity separator "B". Water was not collected for the remainder of the flight. The humidity separator was removed and sent to the vendor for failure analysis. The replacement humidity separator was installed and retested. Crew procedures were available should a problem occur during STS-35 flight, and the separators are redundant. Furthermore, the mission could be terminated early if other efforts to control cabin humidity failed. ***Not a safety concern for STS-35.***
6	Flash Evaporator System (FES) topping duct "B" string heater failed. IFA No. STS-32-14 HR No. ORBI-276B {C} ***There were no FES heater anomalies on STS-35.***	After activation of the FES topping duct heater "B", the aft duct temperature did not increase at the correct rate. This occurred during inflight checkout of redundant heater strings on FD 7. FES topping duct heater "A" was selected and operated nominally for the remainder of the mission. Worst-case effects occur with loss of all 3 redundant heater strings, resulting in loss of the FES. Flight Rule 9-71B states: if the FES topping duct cannot be maintained at 100°F or more, the FES is considered lost and results in a minimum duration mission. Troubleshooting determined that Remote Power Controller (RPC) #34 was defective. The unit was removed, replaced, and retested successfully.

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)		Rationale for STS-35 flight was: • FES redundancy. • Heater redundancy. *Not a safety concern for STS-35.*
7	Inertial Measurement Unit (IMU) #1 was deselected by Redundancy Management (RM) due to Y-axis transients. IFA No. STS-32-15 HR No. ORBI-051 {C} ***No IMU anomalies were reported on STS-35. IMUs #1 and #3 were reported to have differing data on the first day; however, a successful attitude adjustment was made to correct the problem.***	IMU #1, S/N 24, was deselected by RM on FD 6 due to 7 occurrences of erratic Y-axis accelerometer transients; however, IMU #1 continued to track the redundant IMU set after deselection. The crew was able to reselect IMU #1 prior to IMU alignment. After IMU alignment, all 3 IMUs were operating nominally. IMU #1 was deselected prior to crew sleep periods to avoid waking the crew should an alarm occur. No further problems were reported with this IMU for the remainder of the mission. As a precaution, the crew continued to deselect IMU #1 prior to their sleep periods. Playback data indicated that the failure was caused by multiple Y-axis velocity transients. No previous problems associated with this IMU were reported. Additionally, there was no indication that this was a generic problem. IMU #1 was replaced. The removed S/N 24 unit was sent to Johnson Space Center (JSC) for failure analysis. The IMU system has triple redundancy. *Not a safety concern for STS-35.*

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8	Hydraulic systems #1 and #2 unloader valves exhibited anomalous operation. IFA No. STS-32-16 HR No. ORBI-052 {C} ***No unloader valve anomalies were reported on STS-35.***	Approximately 1 hour (hr) prior to circulation pump #2 deactivation during the launch scrub turnaround, there was a significant increase in cycling of unloader valve #2. Approximately 45 min after deactivation of circulation pump #2, all bootstrap fluid pressure was lost. Because of a similar anomaly on STS-28 (IFA No. STS-28-23), the decision was made to replace the hydraulic system #2 unloader valve prior to STS-32 launch. The hydraulic system #1 unloader valve leaked excessively when the accumulator pressure fell below 2300 psia. This was an internal hydraulic leak with hydraulic fluid on the high-pressure accumulator side leaking to the low-pressure return side. Prior to STS-32 flight, it was known that all 3 OV-102 unloader valves were experiencing out-of-specification leakage [see STS-32 Mission Safety Evaluation (MSE) Report, Postflight Edition, July 20, 1990, Section 4, Orbiter 9]. This condition was waived (WK 1547) for 1 flight only, STS-32, with the understanding that hydraulic accumulator pressures would be closely monitored during prelaunch activities. It was believed that contamination in the unloader valve pilot area caused the leakage. System #1 unloader valve failed during testing and was removed, replaced, and successfully retested. Systems #2 and #3 data were acceptable. There were no constraints to STS-35 launch. There was no indication of anomalous unloader valve operation on OV-102 during the STS-35 turnaround. Bootstrap pressure can be maintained with increased circulation pump operation as was done on STS-32. Additionally, loss of one hydraulic system is an acceptable risk. ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	WSBs #2 and #3 regulator pressure decaying slowly. IFA No. STS-32-17 HR No. INTG-072A {C} ***No WSB pressure anomalies were reported on STS-35.***	Gaseous Nitrogen (GN_2) regulator pressure on WSB systems #2 and #3 had pressure decay rates of 0.11 pounds per square inch per hour (psi/hr) over a 16-hr period; the allowable decay rate is 0.06 psi/hr. Monitoring of the pressure decay rate throughout the mission found that it approached 0 by the end of the mission. There is a tank in which GN_2 is stored for use in pressurizing the WSB water storage tank. GN_2 is routed from this tank through a regulator and relief valve prior to entering the water storage tank. Pressure loss is measured between the regulator/relief valve and the water tank. GN_2 pressure loss was seen before and was been attributed to leakage overboard of GN_2 through an improperly reseated relief valve. The relief valve "burps" on ascent and should reseat when pressure is greater than 28 pounds per square inch (psi). Other possible GN_2 leaks could occur at the pressure transducer port or at the GN_2 vent Quick Disconnect (QD); however, both failure modes are very unlikely. The other possible cause for the observed loss of pressure was loss of water in the storage tank. If water loss was the problem, a reduced mission duration was possible. At the decay rate originally recorded, only 7.5 min of APU operation would be available due to APU bearing temperature constraints. It was believed that the pressure decays were due to the GN_2 relief valves not being fully seated, and were not due to water leaks. The poppets in the relief valves were removed and replaced. GN_2 24-hr decay check on system #2 indicated no leakage; decay check on system #3 indicated leakage of 0.06 psi/hr that was within OMRSD limits. ***This anomaly was resolved for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
10	Avionics bay "3A" smoke detector transient alarm and associated lights. IFA No. STS-32-19 HR NO. ORBI-259A {AR} ***No smoke detector anomalies were reported on STS-35.***	On FD 8, the crew reported a smoke alarm with siren from avionics bay "3A", sensor "3A". The alarm cleared itself in about 6 sec. Playback data indicated that there was no increase in the smoke concentration readings. A successful fire/smoke detection test was subsequently performed indicating that there was no problem with the detection system. It was concluded that an intermittent fault in the smoke detection electronics most likely caused the alarm. Sensor 3A operated nominally for the remainder of the day after the initial alarm. There were several previous instances of smoke detection failures where no cause could be found. Sensor "3A" annunciated several additional times during the crew sleep period on FDs 9 and 10. Each time, the crew checked for increased smoke concentrations; no increase was noted. A decision was made to pull the sensor circuit breaker to avoid nuisance alarms during the crew sleep period. Continued nuisance alarms prompted a decision to open the circuit breaker during reentry. This resulted in the loss of smoke detection redundancy in avionics bays #1 and #3. Safety concurred with this plan based on the fire/smoke detection/suppression test for avionics bays #1, #2, and #3 and after evaluating the risk of loss of redundancy against possible crew distraction during reentry and landing. Sensor "3A" was removed and replaced. The defective unit was sent to the vendor for failure analysis; however, it was damaged during transportation and could not be analyzed. The replacement unit was successfully tested.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10 (Continued)		Rationale for STS-35 flight was: • Sensor "3A" was removed and replaced. • The replacement unit was successfully tested. *This risk factor was resolved for STS-35.*
11	Waste water dump line/nozzle blockage. IFA No. STS-32-21 HR No. ORBI-254 {C} *The waste water dump function on STS-35 degraded to the point of failure (IFA No. STS-35-5). Repeated attempts to clear the line were unsuccessful. Alternate methods for storing waste water were used in order to continue the mission.*	During free fluid disposal on FD 10, the crew did not get any suction through the collection wand. A later attempt at dumping the waste tank was also unsuccessful. Troubleshooting and visual inspection of the dump nozzles using the Remote Manipulator System (RMS) determined that there was no icing in the dump nozzles. It was suspected that a restriction formed in the waste dump line. Inspection at Dryden found charred material around the urine dump nozzle face. Blockage was previously seen; however, there was more than usual. A sample of the material was taken for analysis. A sample taken from the orifice indicated that some potassium was among the charred material; everything else was normal. "Mucky junk" was flushed from the dump line. Troubleshooting confirmed that the dump line was clogged. The line was removed, and the replacement line/nozzle installation was completed. Leak checks and heater/insulation installation were completed. *Not a safety concern for STS-35.*

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
12	Backup Flight Computer (BFC) General Purpose Computer (GPC) errors - Input/Output (I/O) terminal B. IFA No. STS-32-22 HR No. ORBI-066 {AR} ORBI-334 {AR} *No GPC anomalies were reported on STS-35.*	GPC #5, where the Backup Flight System (BFS) software was resident, registered numerous GPC error code 41s (illegal engage/I/O term B). Real-time data analysis indicated that the GPC #5 "Term-B" discrete was toggling. If the GPC #5 discrete is toggling or fails hard "0", the backup BFS/GPC cannot gain control of the 8 flight buses. The error was the result of the BFS detecting no I/O terminate B discrete when the engage discretes are not present. The error was logged approximately 43 times before the GPC was halted. As a result, the BFS was moved from GPC #5 to GPC #2 and reinitialized. This left STS-32 with 3 primary and 1 backup GPC for entry and reduced fault tolerance to a single failure. The BFS software is normally loaded into GPC #4 in the event that GPC #5 is determined bad. Because of the preflight concern with Kemet capacitors in the GPC #4 Input/Output Processor (IOP), it was decided before launch to use GPC #2 as the alternate BFC. KSC was able to recreate the problem; however, when the Breakout Boxes (BOBs) were installed, the problem did not recur. BFC connector J4 and IOP connector J5 were inspected. Power was cycled to the GPC. The GPC output switch was moved, wires were wiggled, and the problem would not repeat. Additional troubleshooting proved to be nonproductive. BFC #2 and the IOP #5 were removed and replaced. A confidence test was completed on IOP #5 input receiver (DI 13). If this anomaly occurred prior to launch, the launch would has been held until the cause of the problem was determined, or the launch would have been scrubbed if it was determined that the BFC/GPC was bad.

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
12 (Continued)		To date, there have been several IOP circuitry failures; however, there were no failures involving erroneous I/O terminate discretes. None of the IOP circuitry failures were determined to be generic in nature. There were 2 BFC transmit circuitry failures identified involving I/O terminate discretes. Both failures occurred during acceptance testing, and both are still under investigation. Rationale for STS-35 flight was: • Launch would occur only if there was a good 5-GPC set. • There is built-in failure tolerance and redundancy (2-fault tolerant). • Proven crew workaround procedures were in place. ***This risk factor was acceptable for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
13	WSB #3 controller "A" overcooling. IFA No. STS-32-23 HR No. ORBI-170 {C} *WSB #3 operations on controller "A" did not initiate lube oil cooling as required at 250°F on STS-35. (IFA No. STS-35-17). This problem was attributed to the presence of wax in the lube oil. During reentry operations, WSB #3 while under the control of controller "A" was reported to be overcooling the lube oil.*	WSB #3 went into the heat exchanger mode early and dumped excessive water while operating on controller "A". The crew switched to controller "B", and operation continued nominally. There were no previous WSB failures of this type. Troubleshooting confirmed the failure of controller "A". The defective controller was removed, replaced, and successfully retested. A redundant controller is also available. In addition, redundant hydraulic systems are available if both controllers in one system malfunction. *Not a safety concern for STS-35.*
14	Main Propulsion System (MPS) Liquid Hydrogen (LH_2) outboard fill and drain relief valve leak. IFA No. STS-32-25 HR No. ORBI-306 {AR} *No MPS fill and drain valve problems were reported on STS-35.*	The MPS outboard fill and drain valve PV11 was found with a blowing leak during STS-32/OV-102 postflight inspection. The leak was heard and felt at the 6:30-o'clock valve position. Helium tank pressure decrease confirmed the leak. Investigation indicated some contamination in the system. The PV11 valve is redundant, and a second failure (PV12 or PV13) would required to cause a hazardous condition during main stage. PV11 was removed and replaced. (See Section 4, Integration 2, for additional information.) *This risk factor was resolved for STS-35.*

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
15	RH stop bolt was found slightly deformed on the STS-32 centering ring of the forward ET attach/separation assembly. IFA No. STS-32-26 HR No. INTG-051B {C} *Postflight inspection of STS-35/OV-102 found the RH stop bolt bent approximately 5° from center (IFA No. STS-35-22). Damage was worse than that seen on STS-32 and STS-38, but not as bad as experienced on STS-34. The bolt was removed and sent to the vendor for analysis.*	Postflight inspection of STS-32/OV-102 found the RH stop bolt slightly deformed (not bent condition) on the centering ring of the forward ET attach/separation assembly. This anomaly was similar, but not nearly as bad as that seen previously on STS-34. Deformations or flat spots, similar to that seen on STS-32, were found on other flight and qualification bolts. A bent stop bolt was first found on STS-34/OV-104. STS-34 postflight inspection at Dryden found the RH stop bolt to be bent, forward and inboard. This bolt, located on the centering ring of the forward ET attach/separation assembly, was found compressed into the centering mechanism. It is used to restrict side motion at the attach/separation assembly between the ET and Orbiter and is not considered a structural bolt. Indications were that the assembly sustained a side load. The moment required to bend this bolt is in excess of 10,000 inch-pounds (in-lb). The force required to obtain this moment is 900 pound (lb). A side load of this magnitude could lead to early uncontrolled separation of the Orbiter from the ET. There was no indication that a side load occurred on either the STS-34 or STS-32 flights. The most probable cause of the STS-34 anomaly was determined to be improper sequencing of the ET/Orbiter mating procedure resulting in a yaw moment that could bend the bolt. Sequencing employs Ground Support Equipment (GSE) (H72-0590) that could produce the required loads. Improper sequencing would not lead to early, uncontrolled separation of the ET and Orbiter. However, a bent bolt extended into the airstream could result in excessive localized heating during

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
15 (Continued)		reentry. An additional cause could be the use of the Orbiter transporter, which moves the Orbiter from the Orbiter Processing Facility (OPF) to the Vehicle Assembly Building (VAB). Since the transporter was first used on STS-32, it is possible that the bolt was deformed at the Orbiter-to-transporter mate. There were no anomalies recorded during STS-32 ET/Orbiter mating. A bent stop bolt is a criticality 3 failure. Analysis indicates that a moment of 430-2100 in-lb could locally deform the bolt end. This moment could be generated by either side-to-side movement during normal handling or by the small, pyro-initiated rocking motion at separation. The rocking motion was first seen during review of pyro qualification test film. The bolts used in the qualification tests also exhibited similar local flat spots. (Note that the rocking motion was not sufficient to cause the bolt bending experienced on STS-34.) New mating procedures were developed to alleviate this problem. These procedures were formalized and will also be used for Orbiter-to-transporter mate. The STS-32/OV-102 ET attach/separation assembly was removed at Dryden and sent to Rockwell International (RI)/Downey for failure analysis. ***This risk factor was resolved for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
16	Pilot seat down drive motor did not operate. IFA No. STS-32-27 HR No. ORBI-256C {AR} ORBI-340 {AR} ***The pilot seat down limit switch failed to allow the seat to be driven down on STS-35 (IFA No. STS-35-23). The seat motor and limit switch will be investigated by the vendor.***	During preparation for descent, the pilot attempted to make seat adjustments. The seat would drive up, but not in the down direction. Repeated attempts to lower the seat failed. The forward and back drive was not used or tested. The most probable cause of this failure was a defective down limit switch. Ground tests showed that the seat was operating nominally. The flight effect was crew inconvenience if seat height cannot be adjusted. ***Not a safety concern for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	Main Combustion Chamber (MCC) aft end debond found on engine #2022. IFA No. STS-32-ME-01 HR No. ME-B5 (All Phases) {C} ***Postflight ultrasonic inspection of STS-35 engine #2024, MCC #2013, found a debond less than 3/64" in diameter. This debond was considered to be within acceptable limits, and no action will be taken.***	On engine #2022, a 5/64" diameter MCC debond was detected post STS-32 flight by ultrasonic inspection. It was located 1/2" from the edge and in line with nozzle tube #664. An aft region internal fuel leak is a criticality 1 failure and is assumed to be rapid and extensive, resulting in High-Pressure Fuel Turbopump (HPFTP) cavitation and Liquid Oxygen (LOX) rich operation. The test history of this engine included 16 starts and 4650 sec. The debond was limited to an area between adjacent feed slots in line with nozzle tube #664 (2 affected channels). Postflight leak checks verified no leak at the bond line. There was no fabrication or assembly history found which was indicative of a problem. The failure was consistent with previous bond line failure assessment. The debond initiated at the aft end of the feed slots, resulting most likely from an undetectable flaw or marginal bond in this region. The defect could then propagate as a result of start/shutdown transients (highest strain to bond line). A proof test screens gross bond deficiencies. Also, post-proof ultrasonic inspection detects debonds. Current data on this type of condition indicates that the propagation rate is slow and stable; there is a low probability of a massive bond line failure. The MCC was returned to Canoga for rebuild/repair prior to reuse. Engine #2022 was replaced by engine #2012. (The same type failure was seen on engine #2031, post STS-29 flight. See STS-30 MSE, Postflight Edition, August 25, 1990, Section 5, SSME 1, for additional related information.) ***This risk factor was resolved for STS-35.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
2	Gouges found in the MCC throat area of engines #2024 and #2028. IFA No. STS-32-ME-02 STS-32-ME-03 HR No. ME-B5 (All Phases) {C} ***No similar anomalies were found on STS-35 SSMEs.***	Gouges were found in the MCC throat area on engines #2024 and #2028. It was thought that these gouges were introduced when the leak check throat plug was installed prior to flight. The gouges should have been found prior to flight and polished-out. Imperfections of this type in the MCC throat area could cause localized hot spots leading to burnthrough. The gouge found on engine #2028 measured approximately 2" long x 0.080" wide x 0.009" deep with some raised metal. The gouge was caused by the engine horizontal installer during removal/installation after STS-28. The engine #2028 MCC liner was repaired using a cell plating process to deposit copper in the gouge area. A NASA/contractor team were formed to revise procedures or modify equipment to eliminate or minimize engine handling damage. In addition, the launch and landing site personnel were counseled on the importance of hardware inspections. The MCC gouge on engine #2024 was noted 6" out from the throat area at the 6-o'clock position. The gouge measured approximately 0.250" long x 0.24" wide x 0.10" deep. The gouge was caused by a "B" nut which is tethered to the upper throat plug. The "B" nut is used to cap the upper throat plug bleed valve. Investigation using an MCC proof-load test article and an upper throat plug found that the "B" nut swinging on its tether could inflict gouges of the dimensions noted. The engine #2024 MCC liner was repaired by reducing the stress concentration in the area of the gouge. ***This risk factor was resolved for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Upper strut Ethylene Propylene Diene Monomer (EPDM) cover partially missing. IFA No. STS-32-B-01 HR No. B-30-06 Rev. C {C} C-60-03 Rev. B {C} ***No similar anomalies were reported on STS-35 Solid Rocket Boosters (SRBs).***	During postflight inspection at Kennedy Space Center (KSC), both left and right SRBs were missing some of their EPDM and Room-Temperature Vulcanizing (RTV) Q3-6077 materials from the upper strut location. The upper strut EPDM was partially missing, and the unprotected areas showed heat effects. A 5" section of EPDM cover was missing on the RH side, and a 4" section of EPDM cover was missing on the Left-Hand (LH) aft sides of the upper struts. The Q3-6077 high-temperature silicone that covers and protects the PR-855 foam from heat damage was missing below the lost EPDM rubber on both RH and LH struts. The PR-855 foam showed heat effects on both the LH and RH struts. Specific heat effects on the RH SRB included: 2 cables (A-bus power and upper strut firing line) were found with heat discoloration on the outer YR-364 tape; 5 sealant caps and PR-1422 were eroded. Thermal analysis indicated that the YR-364 tape should protect the cables. It is possible that damage was caused by aeroheating during descent. The previous worst-case damage was a 3" tear in the EPDM cover on STS-27 and a small piece (1/2" x 1/2") missing on STS-28. No previous heat effects were found on the cables. The edges of the EPDM covers are typically charred and frayed. The areas under investigation included: evaluation of the EPDM bond line, evaluation of the Q3-6077 failure mode, analysis of heat damage to the PR-855, evaluation of the extent of the heating effects on the cables, and design evaluation of the upper strut closeout.

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1 (Continued)		Rationale for STS-35 flight was: • Thermal protection of cables through ascent is assured by blast tape; the Thermal Protection System (TPS) provides additional protection. • Cables are routed in different bundles in the same area. ***Not a safety concern for STS-35.***
2	External Tank Attachment (ETA) ring aft Instrument and Electronics Assembly (IEA) cover sooted. IFA No. STS-32-B-02 HR No. B-60-24 Rev. C {C} ***No sooting was found on the IEA covers on STS-35 SRBs.***	The ETA ring aft IEA middle cover was sooted on the aft inside surface. The function of the IEA was not affected. Minor sooting was confined to a small area on the cover. The problem was attributed to the installation of "larger" Hi-Lok fasteners, preventing proper fit of the cover. As a result of an inspection of this same ETA ring following STS-29, several oversized holes were drilled to accommodate 5/16" Hi-Lok fasteners in place of the normally-used 1/4" size. Since the larger fasteners were longer, the fasteners protruded and held the cover up approximately 0.1", thus allowing a hot-gas path into this area. Build paper on all ETA rings was checked to verify proper installation. This condition was peculiar to an STS-32 ETA ring. The ring was returned to proper configuration prior to reuse. ***Not a safety concern for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
3	Broken fastener found on STS-32 LH SRB upper strut fairing. IFA No. STS-32-M-03 HR No. INTG-081A {AR} INTG-134A {AR} *No fastener problems were found during disassembly of STS-35 SRBs.*	During postflight disassembly of the STS-32 SRBs, a broken fastener was found in the LH upper strut fairing or "milk-can". Proper fastener material properties and heat treatment were confirmed by analysis of the failed fastener. Material analysis concluded that the failure was due to torsional overload. This conclusion led to the determination that overtorquing occurred prior to launch, based on the fact that there are no torsional forces exerted on this fastener during flight. Water impact loading would have resulted in shear failure. There were no problems reported during installation of the STS-35 upper struts. *This risk factor was resolved for STS-35.*

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Right SRM Safe and Arm (S&A) gasket depression on secondary seal. IFA No. STS-32-M-01 HR No. BC-03 Rev. B {C} BI-02 Rev. B {C} ***No problems were found with either SRM S&A seal on STS-35.***	During postflight inspection of the right SRM S&A gasket, a small depression was found in the crown of the secondary seal aft face. The crown of the right SRM S&A gasket secondary seal was depressed inward at the 0° location. The depression measured approximately 0.050" circumferentially x 0.026" radially x 0.0025" deep. This gasket was previously flown on STS-26R (RSRM-1); however, no anomaly was detected during STS-26 postflight inspection since the gasket was not inspected within 1/2 hour after removal from the joint. The gasket was later inspected in accordance with the old gasket inspection requirements for reuse on the STS-32 mission. An additional inspection of this seal was performed when an igniter seal void was discovered for the STS-28 mission. This supplemental gasket inspection required a 3-hr compression test in a plexiglas fixture, with a post-compression touch inspection within 1/2 hr after removal from the fixture. No defects were found. This procedure was documented in the latest release of STW7-2790, (Rev D) for gaskets to be reused, which meets the definition of a "used" gasket. This gasket did not meet the criteria of a used gasket since it was not touch inspected for approximately 20 hr after STS-26 disassembly. *The following corrective actions were implemented:* • Reviewed pedigrees of all gaskets installed on flight and test motors. • Replaced gaskets having no touch inspection performed within 1/2 hr after undergoing 3 days of compression with gaskets that have been touch-inspected properly.

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1 (Continued)		• Created new S&A gasket dash numbers to preclude use of gaskets already accepted per the old requirements. • Changed the reuse inspection specification to require every gasket to have a documented 3-day compression with a touch test within 1/2 hr prior to each reuse. ***This risk factor was resolved for STS-35.***
2	Raised areas found on the igniter inner Gask-O-Seal. IFA No. STS-32-M-02 HR No. BC-03 Rev. C {AR} ***No problems were experienced with STS-35 Solid Rocket Motor (SRM) Gask-O-Seals.***	During postflight assessment of the right SRM igniter inner gasket, raised areas of rubber were found along both sides of the gasket on the outer primary seals. This condition was limited to the void and cushion areas (nonsealing surfaces) intermittently around the circumference of the outer primary seals. The largest area found measured approximately 0.20" circumferentially. This condition was possibly caused by air trapped between the rubber and retainer during the molding process. The seal footprint was not affected by this anomaly. A new baseline was implemented which controls the molding process and adhesive application. The process requires mold bumping to reduce the likelihood of trapped air. Vents also were added to the mold. ***This risk factor was resolved for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ET</u>		
1	Review of the ET separation photos from STS-32 showed 4 Spray-On Foam Insulation (SOFI) divots in the bipod area. IFA No. STS-32-T-01 HR No. INTG-008B {AR} INTG-037B {AR} INTG-081A {AR} ***STS-35 ET separation photos illustrated 11 circular TPS divots in the intertank-to-hydrogen flange area. Divots ranged from 7" to 10" in diameter (estimates).***	Postflight review of STS-32 ET separation photos found 4 SOFI divots just forward and underneath the bipod area. These divots were similar to those seen in STS-28 ET separation photos. On STS-28, divots were seen above the RH bipod spindle. Divots seen on STS-32/ET-32 were into the Isochem layer. Additionally, these divots had the same appearance as divots noted on flights before the implementation of the intertank TPS vent hole modification. No Orbiter damage was attributed to the existence of these divots. Vent holes are drilled through the TPS in the intertank. It was thought that, due to tolerance stackup of the TPS (i.e., thicker areas of TPS), vent holes were not drilled to the proper depth. Additionally, a review found that the vent holes may not have been drilled at the proper angle. Rationale for flight of STS-35/ET-35 was that the vent holes on the intertank were verified for proper depth. ***This risk factor was resolved for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
KSC		
1	RH aft Integrated Electronic Assembly (IEA) bent pins. IFA No. STS-32-K-03 HR No. B-00-17 Rev. B-DCN5 {C} ***No bent pins were found during postflight inspection of STS-35 elements.***	During postflight assessment, a connector in the RH aft IEA was found to have 2 bent pins. Pin #64, which is wired as a spare, was bent 90° flat and was nearly touching pin #58. Pin #66, which was a wired shield, was bent 180° into a hook shape. Since these pins are wired spares, they are not checked out during final functional testing after final mating of the cable to the IEA. The pins were bent during cable mating to the IEA. Adjacent pins in the connector control vital Thrust Vector Control (TVC) functions. However, the signal paths are redundant and, by design, redundant functions cannot be routed through the same connector. Also, Operational Maintenance Requirements Specification (OMRS) system check verified every functional path within all circuits. Rationale for STS-35 flight was: • The File II and File V system check verified every functional path within the IEA after final connector mating. • By design, redundant functions were not be routed through the same connector. • Adjacent pins, if shorted together, would not cause loss of a critical function. • OMRS system check verified every functional path within all circuits. ***This risk factor was resolved for STS-35.***

STS-32 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>MCC</u>		
1	State vector uplink incident. IFA No. STS-32-MOD-01 HR No. ORBI-066 {AR} ***No uplink errors were reported on STS-35. There was, however, an error in a patch to the BFS software prior to launch (IFA No. STS-35-I-01). Because of the relocation of STS-35 to pad B for launch, launch pad locational data was required to be updated in both the PASS and BFS. During ascent, a difference of 143 ft was witnessed in the positional data sent from the BFS and PASS. Post-ascent evaluation of telemetry data identified an error in the sixth digit of the longitude string of the BFS patch.***	An off-nominal maneuver was observed in all 3 axes during the crew sleep period on FD 9. The vehicle began to roll nose-to-tail following a state vector update that was later determined to be erroneous. The vehicle roll rates were arrested by manual intervention by the crew. A new state vector was uplinked later to allow the crew to reselect the auto-track mode. An investigation team was established to determine the cause of this incident and to make recommendations for corrective action to prevent future occurrences. The findings indicated that the state vector uplink incident was caused by an operator error. The erroneous operator procedural response was clearly outside the "trained to" and commonly expected procedures for this scenario. The basic system design and procedures associated with every aspect of this incident are mature and sound. The ground system and onboard system hardware and software worked as designed. Based on the investigation team findings, the following was implemented for STS-36 and subsequent missions in order to preclude recurrence: • The Integrated Communications Officer (INCO) console handbook was updated to add cautionary notes prohibiting the use of manual execute (override) in the presence of an indicated data reject message.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
MMC		
1 (Continued)		• Handshakes between uplink owners and the INCO console are required when passing direction cosine matrix numbers. • Handshakes between the INCO console and the INCO backroom console are required for all critical commands and command troubleshooting. • The command load owner and flight director are notified of all manual loads or line-by-line corrections. Command load owner and flight director concurrence is required. Additional long-term corrective actions are under consideration. ***This risk factor was resolved for STS-35.***

SECTION 7

STS-35 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-35/OV-102 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-35 INFLIGHT ANOMALIES

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Backup Flight System (BFS) software patch for pad B definition of longitudinal location was incorrect. IFA No. STS-35-I-01 HR No. ORBI-066 {AR}	The relocation of STS-35 from pad A to pad B required an I-load software patch to include the pad B location definition. During ascent, a difference of 143 feet (ft) was witnessed in the positional data sent from the BFS and the Primary Avionics System Software (PASS). Post-ascent evaluation of telemetry data identified an error in the sixth BFS software patch for pad B. An error was found in the sixth digit of the longitude string. An investigation into the factors that led to the incorrect longitude string determined that the error was caused by software developers at Rockwell International (RI)/Downey who incorrectly read the Change Request (CR). CR 90365 was faxed to RI/Downey and was used as the authority for the I-load software patch. All who read CR 90365 interpreted the longitude position as -1.40709036E+00; the correct value was -1.40709836E+00. Verification of this value was not made at RI/Downey prior to incorporation into the I-load. Verification could have been made through comparison with the electronic data set associated with CR 90365. The modified BFS I-load passed STS-35 certification testing. Pass/fail criteria for downrange position at Main Engine Cutoff (MECO) command is ±600 ft; a value of 322 ft was observed and accepted during testing. The worst-case effect for downrange position at MECO command if the position is >±600 ft would be either the External Tank (ET) would land outside the predicted footprint or there would be insufficient propellant to continue the mission. To ensure that, in the future, conditions will not exist for a similar error to occur, all CRs that require I-load patches must be accompanied by the associated electronic data set for verification.

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Left Reaction Control System (RCS) drain panel heater "A" was not at normal temperature. IFA No. STS-35-04	On-orbit, the left RCS drain temperatures indicated that the heater did not cycle at the expected 56.6°F. The temperature on heater "A" went down to 52°F before the crew was instructed to switch to heater "B". The "B" heaters operated nominally after switchover. Data analysis determined that the "A" heater cycled once normally prior to this failure. On-orbit troubleshooting included switching back to heater "A" and allowing the left RCS drain temperatures to drop to 40°F which confirmed the failure of heater "A" to cycle properly. Due to the attitude of the vehicle, the RCS drain temperature did not go below 40°F for the remainder of the mission. The Shuttle Operational Data Book (SODB) limit is +20°F; RCS oxidizer freezes at +12°F, and the fuel freezes at −60°F. This is believed to be an isolated failure, with no indication of a generic problem. The most likely cause of this anomaly was a failed thermostat. There were no reported problems with the left RCS drain panel thermostat on previous OV-102 missions. RCS heaters are Crit 2R3 components. RI is currently investigating if there are any Crit 1 applications of this thermostat.

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2	Degradation of waste water dump function. IFA No. STS-35-05 HR No. ORBI-254 {AR}	Water from the waste water storage tank is periodically dumped overboard into space during a nominal mission. A gradual degradation of the waste water dump rate was noted during the first 3 waste water dump cycles. The line was completely blocked on the fourth dump. Inflight maintenance was performed with no success. Waste tank offload into Contingency Water Container (CWC) and urine collection devices was required for the remainder of the mission. A decision was made to magnifest additional CWCs on all flights. Similar problems on STS-32/OV-102 led to removal and cleaning of the last 22" of the dump line. It is believed that the current blockage is upstream of this section. Troubleshooting of the STS-35 anomaly found that the waste water dump line filter had deteriorated and was the root cause of the line blockage. The filter assembly has 3 filters (coarse, medium, and fine) and is replaced after every 3 flights. It was determined that the filter material, polyurethane, will deteriorate after approximately 8 years. A spare filter assembly obtained from the logistics stockroom was found to show similar signs of deterioration. This spare filter, that had never been used (still in shipping package), was manufactured in 1980.
3	-Z star tracker Serial Number (S/N) 006 failed 2 initial self-tests. IFA No. STS-35-10	On the initial power-up, the -Z star tracker S/N 006 failed the first 2 self-tests. Position errors were observed on the first self-test software cycle. All subsequent software cycles indicated the correct Built-In Test Equipment (BITE) star position. The -Z star tracker passed the third self test and 5 additional self-test cycles. Performance thereafter was nominal.

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)		Initial evaluation of this anomaly determined that the star tracker electronics may not have responded quickly enough to star acquisition during the first 2 self-test cycles. It is believed that this slow response time could be a function of warmup time. Minimum warmup time is >15 minutes (min); however, the STS-35/OV-102 star trackers had power on for 25 min prior to the first 2 self-test cycles. This slow response time condition was seen during laboratory tests on other units; however, this was the first occurrence in flight. This anomaly is a Crit 1R3 failure mode. The -Y star tracker and the Crew Optical Alignment Sight (COAS) have redundant functions to the -Z star tracker.
4	Payload Bay Door (PLBD) environmental seal debond. IFA No. STS-35-16	Postflight inspection of OV-102 found a 24" piece of the environmental seal teflon material hanging loose between panels #1 and #2 on the right PLBD, at the top. The loose seal was cut off prior to the ferry flight to preclude further loosening or damage. There was no apparent damage internal to the payload bay. This was a first time occurrence in this area. A 6" splice segment of the PLBD-to-aft bulkhead environmental seal debonded on STS-41/OV-103. Evaluation of STS-41 problem determined the cause to be insufficient application of the seal (bad etching and bonding). Investigation into the cause of the STS-35 anomaly continues.

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	Water Spray Boiler (WSB) #3A operation was abnormal during ascent and entry. IFA No. STS-35-17 HR No. ORBI-121 {AR}	During ascent, WSB #3A did not initiate spray cooling until Auxiliary Power Unit (APU) #3 lube oil return temperature reached 277°F. WSB cooling operations should begin at 250°F. During reentry operations, WSB #3A slightly overcooled the lube oil. Preliminary analysis of this anomaly indicated that the presence of wax in the APU #3 lube oil may have caused the spray bar to freeze. The lube oil temperature increased until the spray bar thawed, and proper cooling commenced. A hot oil flush will be performed during the STS-40/OV-102 turnaround process. WSB #3A operations will be tested after the hot oil flush.
6	Window W-1 has a 0.15" diameter chip. IFA No. STS-35-18 HR No. ORBI-009 {AR}	Postflight inspection of the OV-102 windows revealed a chip in window W-1, measuring 0.15" in diameter and approximately 0.0109" in depth. A "spider web" type crack formation was found to be radiating from the impact point. During the crew debrief, it was determined that the crew first noted the chip on Flight Day (FD) 6. It is believed that the chip occurred during ascent. Window W-1 will be removed and replaced prior to the next OV-102 mission. Further examination of the window will be performed to attempt to determine the cause of the chip.
7	WSB #2 was subjected to abnormally large quantities of wax. IFA No. STS-35-19 HR No. ORBI-121 {AR}	During ascent and entry, indication of a large amount of wax was noted in the APU #2 lube oil system. This condition subjected WSB #2 to wax in the lube oil; therefore, WSB #2 will require a hot oil flush during the STS-40/OV-102 turnaround processing. APU #2 will be removed and replaced during this process due to Gas Generator Valve Module (GGVM) life-limit criteria.

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8	RCS vernier thruster R5D failed "off". IFA No. STS-35-20 HR No. ORBI-056 {C}	During orbital maneuvering, RCS vernier thruster R5D exhibited low Chamber Pressure (P_c) and was deselected by Redundancy Management (RM). Data evaluation indicated that helium was present in the crossfeed line. A similar failure was seen on STS-9. Vernier thruster R5D was successfully hot fired on orbit to flush out the helium. Evaluation of the hot-fire data indicated some gas ingestion during the first pulse and none in the 4 subsequent pulses. RM was reset following nominal performance during the hot fire.
9	Orbiter/ET Liquid Oxygen (LO_2) aft attach/separation hole plugger did not fully extend. IFA No. STS-35-21 HR No. ORBI-302A {AR}	The Orbiter/ET LO_2 aft attach/separation hole plugger did not complete its stroke. One of the 2 pyros was jammed between the plugger and the rim of the hole. The other pyro device was not found and may have escaped. The ET umbilical door closed and latched normally, and no debris was found on the runway after the ET doors were opened. Similar hole plugger failures occurred on STS-29 and STS-34. The concern was 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 likelihood of escaping fragments preventing the ET umbilical door from closing was determined to be remote. The ET doors may be recycled in flight in the event that closing or latching is obstructed. The Orbiter performs a maneuver at ET separation, moving away from the ET and escaping possible debris prior to ET umbilical door closure.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10	Right-Hand (RH) stop bolt was found bent on the STS-35 centering ring of the forward ET attach/separation assembly. IFA No. STS-35-22 HR No. INTG-051B {C}	Postflight inspection of STS-35/OV-102 found the RH stop bolt bent approximately 5° from center. Bending of stop bolts was previously experienced on STS-34, STS-32, and STS-38. Damage to the STS-35/OV-102 stop bolt was worst than that seen on STS-38, but not as bad as the STS-34 anomaly. The left and right stop bolts restrict side rotation of the centering ring during Orbiter/ET mate. They are not designed to carry any mating or flight loads. A review of all ground operations was undertaken. It was determined that mating and demating operations have the physical capability to bend the stop bolts. Mating procedures were modified after STS-34 to control the yoke position and to preclude the potential for bolt damage during mating operations. No stop bolts were reported bent during the next several missions. However, there were no requirements implemented for demating operations. Because of the recent reports of bent stop bolts on STS-38 and STS-35 and both flows required Orbiter demate from the ET, a modified demate procedure was developed and submitted for approval. New mate/demate Ground Support Equipment (GSE) with improved visual and digital readout will be available in mid-1991. Additionally, a more robust stop bolt design is in evaluation for future use. There were no anomalies recorded during the STS-35 ET/Orbiter mating process. Misalignment of the ET attach points, EO-2 and EO-3, was not considered a contributor to this anomaly. A bent stop bolt is a Crit 3 failure. Analysis conducted during the investigation of previous instances of bent or damaged stop bolts determined that a moment of 430-2100 inch-pounds (in-lb) could locally deform the bolt end. This moment could be generated by either side-to-side

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10 (Continued)		movement during normal handling or by the small, pyro-initiated rocking motion at separation. The rocking motion was first seen during review of pyro qualification test film. The bolts used in the qualification tests also exhibited local flat spots similar to those seen on the STS-32 stop bolts. The rocking motion, however, was determined not to be sufficient to cause the bolt bending experienced on STS-34.
11	Pilot seat down-limit switch failure. IFA No. STS-35-23 HR No. ORBI-340 {AR}	During the ingress and prelaunch operations, the pilot attempted to make seat adjustments. The pilot seat failed to drive down. This was a repeat of an STS-32 anomaly (IFA No. STS-32-27). The seat, however, worked properly on orbit. The down-limit switch was replaced during the STS-35 turnaround process. Further troubleshooting will be performed at KSC. This anomaly was unique to OV-102.

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Heat-effected Carbon Cloth Phenolic (CCP) seen on the left Solid Rocket Motor (SRM) at nozzle joint #3. IFA No. STS-35-M-01 HR No. BN-03 Rev. C {AR}	During postflight inspection of nozzle joint #3 of the left SRM by Thiokol Corporation (TC), a 1.5" gas path was observed through a Room Temperature Vulcanizing (RTV) void at 195° of the CCP. Surface heat effects and associated sooting resulted. Heat effects to the virgin CCP was approximately 1.0" radially past the char line and appeared to be on the surface only. Soot reached the primary O-ring, approximately 12" circumferentially in both directions from the 195° position. There was no blowby erosion or heat effects to the primary O-ring. No metal nozzle components were affected. The RTV contributes as a thermal barrier only and is not considered a seal in the nozzle joints. RTV below the char line is a design goal only and is not a performance requirement. To date, all RTV backfill nozzle joints met all design requirements. There were no flight or static test motor nozzle joints which have exhibited primary O-ring heat effects, erosion, or blowby. This was the first occurrence of heat-affected CCP in nozzle joint #3. Heat-effected CCP, silica cloth phenolic, and glass cloth phenolic were seen in joint #2 (nose inlet bearing/cowl) of STS-36, QM-7, and PVM-1, all with no O-ring heat effects. Gas paths and soot in nozzle joints were within the experience base of 26 flight SRMs and 7 static test nozzles. TC plans to conduct an aero/thermal analysis of nozzle joint #3 to determine the gas volume and flow characteristics associated with the STS-35 anomaly.

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ET</u>		
1	ET Thermal Protection System (TPS) divots found at the intertank-to-hydrogen flange. IFA No. STS-35-T-01 HR No. INTG-008B {AR} INTG-037B {AR} INTG-081A {AR}	During review of ET photographs taken by the STS-35 crew after separation, 11 circular TPS divots were observed. All were located on the intertank-to-hydrogen flange. Six divots were on the left side of the tank; the other 5 were on the right side. The divots were estimated to be from 7" to 10" in diameter. The ET intertank flanges are closed out after the splice/mate of the LO_2 tank to the Liquid Hydrogen (LH_2) tank. The intertank flanges are then manually sprayed with BX-250 foam, which is bonded to existing tank foams by an isochem adhesive. Adhesive voids cannot be detected by visual inspection, and there is currently no available, nondestructive test to verify proper adhesion. Review of similar photographs from STS-28 found a large divot in the ET intertank acreage TPS. This divot was estimated at 23" x 15" and >1" in depth. There was no conclusion reached as to the cause of the STS-28 divot; however, when possible, all crews are asked to photograph the ET after separation. The divots seen on STS-35 were the first experienced since STS-28. Review of the STS-35 ET photographs by Martin Marietta Corporation (MMC) determined that the divots did not expose the ET tank metal. The "whitish" color is believed to be fresh BX-250 foam that was not exposed to the weather. Worst-case analysis performed by MMC indicated that bare ET tank metal of the divot sizes seen on STS-35 would not result in structural or thermal problems.

SECTION 8

BACKGROUND INFORMATION

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

APPENDIX A

LIST OF ACRONYMS

10K	10,000
AFB	Air Force Base
AFD	Aft Flight Deck
AMOS	Air Force Maui Optical Site
AOA	Abort Once Around
AOS	Acquisition of Signal
APU	Auxiliary Power Unit
AR	Accepted Risk
ATP	Acceptance Test Procedure
ATT	Acceptance Thermal Testing
AVT	Acceptance Vibration Testing
BB	Barrier-Booster
BBXRT	Broad Band X-Ray Telescope
BFC	Backup Flight Computer
BFS	Backup Flight System
BITE	Built-In Test Equipment
BOB	Breakout Box
BSR	Bite Status Register
C	Controlled
CA	California
CAR	Corrective Action Report
CB	Circuit Breaker
cc/sec	Cubic Centimeters Per Second
CCP	Carbon Cloth Phenolic
CDMS	Command and Data Management System
CEI	Configuration End Item
CG	Center of Gravity
CO_2	Carbon Dioxide
COAS	Crew Optical Alignment Sight
CR	Change Request
CRES	Corrosion Resistant Steel
Crit	Criticality
CRT	Cathode Ray Tube
CVFA	Check Valve Filter Assembly
CWC	Contingency Water Container

APPENDIX A

LIST OF ACRONYMS - CONTINUED

DAR	Deviation Approval Request
DDU	Dedicated Display Unit
DM	Development Motor
DR	Discrepancy Report
DSC	Designated Signal Conditioner
DTO	Detailed Test Objective
DU	Display Unit
EAFB	Edwards Air Force Base
EGT	Exhaust Gas Temperature
EIU	Engine Interface Unit
EPDM	Ethylene Propylene Diene Monomer
EPDS	Electrical Power Distribution System
EST	Eastern Standard Time
ET	External Tank
ETA	External Tank Attachment
F	Fahrenheit
FASCO	Flight Acceleration Safety Cutoff System
FC	Fuel Cell
FCL	Freon Coolant Loop
FCS	Flight Control System
FCV	Flow Control Valve
FD	Flight Day
FDA	Fault Detection and Annunciator
FEC	Field Engineering Change
FES	Flash Evaporator System
FID	Fault Identification
FMEA/CIL	Failure Modes and Effects Analysis/Critical Items List
FOS	Factor of Safety
FP	Fuel Pump
FPB	Fuel Preburner
FPM	Flow Proportional Module
FPV	Flow Proportioning Valve
FRR	Flight Readiness Review
FRT	Flight Readiness Test
FSP	Flight Safety Panel
ft	Feet

APPENDIX A

LIST OF ACRONYMS - CONTINUED

GEI	Ground Environment Instrumentation
GFE	Government Furnished Equipment
GG	Gas Generator
GGVM	Gas Generator Valve Module
GH_2	Gaseous Hydrogen
GN_2	Gaseous Nitrogen
GNCS	Guidance, Navigation, and Control System
GO_2	Gaseous Oxygen
GOX	Gaseous Oxygen
GPC	General Purpose Computer
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
H_2	Hydrogen
HCF	High-Cycle Fatigue
HDP	Holddown Post
HGDS	Hazardous Gas Detection System
HPF	High-Pressure Fuel
HPFTP	High-Pressure Fuel Turbopump
HPOTP	High-Pressure Oxidizer Turbopump
HPV	Helium Precharge Valve
HR	Hazard Report
hr	Hour
HUT	Hopkins Ultraviolet Telescope
Hz	Hertz
I/O	Input/Output
ICD	Interface Control Drawing
ID	Inside Diameter
IEA	Instrument and Electronics Assembly Integrated Electronic Assembly
IFA	Inflight Anomaly
IFM	Inflight Maintenance
IMCS	Image Motion Compensation System
IMU	Inertial Measurement Unit
in-lb	Inch-Pounds
INCO	Integrated Communications Officer
INTG	Integration
IOP	Input/Output Processor
IPS	Instrument Pointing System
IRS	Integrated Radiator System
JSC	Johnson Space Center

APPENDIX A

LIST OF ACRONYMS - CONTINUED

K	Kelvin
kbit	Kilobit
KSC	Kennedy Space Center
L-2	Launch Minus 2 Days (Review)
lb	Pound
lb/hr	Pounds Per Hour
lbf	Pounds Force
LCC	Launch Commit Criteria
LCF	Low-Cycle Fatigue
LCN	Logic Change Notice
LD	Leak Detectors
LH	Left-Hand
LH_2	Liquid Hydrogen
LiOH	Lithium Hydroxide
LME	Liquid Metal Embrittlement
LO_2	Liquid Oxygen
LOS	Loss of Signal
LOX	Liquid Oxygen
LPF	Low-Pressure Fuel
LPFTP	Low-Pressure Fuel Turbopump
LPOTP	Low-Pressure Oxidizer Turbopump
LPS	Launch Process Sequencer
LSC	Linear Shaped Charge
LSFR	Launch Site Flow Review
MCC	Main Combustion Chamber
	Mission Control Center
MCF	Major Component Failure
ME	Main Engine
MECO	Main Engine Cutoff
MET	Mission Elapsed Time
min	Minute
MLG	Main Landing Gear
MLI	Multi-Layer Insulation
MLP	Mobile Launch Platform
MPE	Mission-Peculiar Equipment
MMC	Martin Marietta Corporation
MMT	Mission Management Team
MOR	Manufacturing Operation Record
MPE	Mission-Peculiar Equipment
MPS	Main Propulsion System

APPENDIX A

LIST OF ACRONYMS - CONTINUED

MRB	Material Review Board
MS	Mission Specialist
MSE	Mission Safety Evaluation
MSFC	Marshall Space Flight Center
N_2	Nitrogen
NASA	National Aeronautics and Space Administration
NDE	Nondestructive Evaluation
NEOM	Nominal End-of-Mission
NLG	Nose Landing Gear
NSI	NASA Standard Initiator
NSLD	NASA Shuttle Logistic Depot
NSRS	NASA Safety Reporting System
O_2	Oxygen
OD	Outside Diameter
OFI	Operational Flight Instrumentation
OMI	Operations and Maintenance Instruction
OMRS	Operational Maintenance Requirements Specification
OMRSD	Operational Maintenance Requirements and Specifications Document
OMS	Orbital Maneuvering System
OPB	Oxidizer Preburner
OPF	Orbiter Processing Facility
OPO	Orbiter Project Office
ORBI	Orbiter
OSMQ	Office of Safety and Mission Quality
OSP	Optical Sensor Package
OV	Orbiter Vehicle
oz	Ounce
P/N	Part Number
PAR	Prelaunch Assessment Review
PASS	Primary Avionics Software System
P_c	Chamber Pressure
PCV	Pulse Control Valve
PLBD	Payload Bay Door
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
PRLA	Payload Retention Latch Assembly

APPENDIX A

LIST OF ACRONYMS - CONTINUED

psi	Pounds Per Square Inch
psi/hr	Pounds Per Square Inch Per Hour
psia	Pounds Per Square Inch Absolute
psig	Pounds Per Square Inch Gage
PSSA	Payload Support Strut Assembly
QD	Quick Disconnect
R	Rankin
RCN	Requirements Change Notice
RCS	Reaction Control System
RFCA	Radiator Flow Control Assembly
RGA	Rate Gyro Assembly
RH	Right-Hand
RI	Rockwell International
RM	Redundancy Management
RMS	Remote Manipulator System
RPC	Remote Power Controller
RSRM	Redesigned Solid Rocket Motor
RSS	Range Safety System
RTLS	Return-to-Launch Site
RTV	Room-Temperature Vulcanizate
S/L	Spacelab
S/N	Serial Number
S&A	Safe and Arm
SAREX	Shuttle Amateur Radio Experiment
sccs	Standard Cubic Centimeters Per Second
scim	Standard Cubic Inch Per Minute
sec	Second
SEM	Scanning Electron Microscope
SII	SRM Ignition Initiator
SLA	Super-Light Ablator
SODB	Shuttle Operational Data Book
SOFI	Spray-On Foam Insulation
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
SSV	Space Shuttle Vehicle

APPENDIX A

LIST OF ACRONYMS - CONTINUED

TAL	Transatlantic Abort Landing
TAPS	Two-Axis Pointing System
TC	Thiokol Corporation
TEM	Test Evaluation Motor
TPMS	ire Pressure Monitoring System
TPS	Thermal Protection System
TSM	Tail Service Mast
TVC	Thrust Vector Control
UHF	Ultrahigh Frequency
UIT	Ultraviolet Imaging Telescope
USBI	United Space Boosters, Inc.
VAB	Vehicle Assembly Building
WSB	Water Spray Boiler
WUPPE	Wisconsin Ultraviolet Photopolarimetry Experiment