MISSION SAFETY EVALUATION REPORT FOR STS-40

Postflight Edition

Safety Division

Office of Safety and Mission Quality

National Aeronautics and Space Administration

Washington, DC 20546

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MISSION SAFETY EVALUATION

REPORT FOR STS-40

Postflight Edition: October 1, 1991

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EXECUTIVE SUMMARY

Following a brief delay due to mid-level clouds, *Columbia* lifted off Pad 39B on June 5, 1991 at 9:24:51 a.m. Eastern Daylight Savings Time (EDT) to begin the nine-day STS-40 Spacelab Life Sciences (SLS-1) mission. All systems performed nominally during launch and ascent. This was the second launch attempt of STS-40.

Significant STS-40 anomalies included:

- Inertial Measurement Unit #2 calibration failures, leading to the scrub of the first launch attempt.
- The loosening of 1307 bulkhead thermal blankets and separation and protrusion of the port-side 1307 bulkhead environmental seal.
- Significant heat effects of the right-hand External Tank (ET) umbilical door centerline latch.

Columbia was cleared for an ontime entry and landing at Edwards Air Force Base on June 14, 1991. STS-40 touched down at 11:40 a.m. EDT on runway 22. The Flight Director reported no anomalies during entry, and all systems looked good. Crew egress was assisted for the first time by a "people mover", a converted airport mobile lounge. The "people mover" was outfitted with a full suite of medical equipment and staff to perform a quick medical examination of the astronauts before their bodies had a chance to adapt to gravity. Four of the 7 crew members remained at the Ames-Dryden Flight Research Facility for an additional week of biomedical testing.

FOREWORD

The Mission Safety Evaluation (MSE) is a National Aeronautics and Space Administration (NASA) Headquarters Safety Division, Code QS produced document that is prepared for use by the NASA Associate Administrator, Office of Safety and Mission Quality (OSMQ), and the Space Shuttle Program Director prior to each Space Shuttle flight. The intent of the MSE is to document safety risk factors that represent a change, or potential change, to the risk baselined by the Program Requirements Control Board (PRCB) in the Space Shuttle Hazard Reports (HRs). Unresolved safety risk factors impacting the STS-40 flight were also documented prior to the STS-40 Flight Readiness Review (FRR) (FRR Edition) and prior to the STS-40 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-40 safety risk factors that represent a change from previous flights, factors from previous flights that had an impact on this flight, and factors that are unique to this flight.

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

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

1.3 Organization

The MSE is presented in eight sections as follows:

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

Section 2 - Provides a summary description of the STS-40 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-40 launch, that were impacted or repeated by anomalies reported for the STS-40 flight.

Section 4 - Contains a list of safety risk factors that were considered resolved for STS-40.

Section 5 - Contains a list of Inflight Anomalies (IFAs) that developed during the STS-39 mission, the previous Space Shuttle flight.

Section 6 - Contains a list of IFAs that developed during the STS-35 mission, the previous flight of the Orbiter vehicle (OV-102).

Section 7 - Contains a list of IFAs that developed during the STS-40 mission. Those IFAs considered to represent a safety risk will be addressed in the MSE for the next Space Shuttle flight.

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

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

SECTION 2

STS-40 MISSION SUMMARY

2.1 Summary Description of the STS-40 Mission

Following a brief delay due to mid-level clouds, *Columbia* lifted off Pad 39B on June 5, 1991 at 9:24:51 a.m. Eastern Daylight Savings Time (EDT) to begin the 9-day STS-40 Spacelab Life Sciences (SLS-1) mission. All systems performed nominally during launch and ascent. This was the second launch attempt of STS-40.

The first launch attempt on June 1, 1991, was scrubbed when problems were found with Inertial Measurement Unit #2 (IMU #2) during prelaunch calibration runs. Data from the first calibration run indicated that the Y-axis accelerometer experienced a 5.54-sigma excursion before stabilizing. Repeated attempts at calibrating IMU #2 resulted in exceeding the Operational Maintenance Requirements and Specifications Document (OMRSD) limit of 5 sigma and required the replacement of IMU #2 prior to flight. IMU #2 was removed and replaced prior to the successful launch of STS-40/OV-102.

Soon after the Payload Bay Doors (PLBDs) were opened, video downlink of the aft 1307 bulkhead revealed that several thermal blankets were dislodged and damaged. In addition to the loose thermal blankets, a section of the port-side 1307 bulkhead PLBD environmental seal was separated from its retainer and protruded into the Payload Bay. These anomalies were indicative of air intrusion into the Payload Bay during ascent. Further inspection determined that the protruding seal had separated at a repair splice at the Y_0 = - 33-inch position on the bulkhead. There were 2 protruding pieces; the inboard piece was approximately 22 inches long, the outboard piece was approximately 8 inches long. There was concern that the protruding seal could prevent the port PLBD from fully closing for entry. Because analysis findings indicated that the seal could interfere with the PLBD locking mechanism, an Extravehicular Activity (EVA) demonstration was performed at the Kennedy Space Center (KSC). An astronaut, wearing EVA gloves, performed several tests which included replacing the seal in its retainer and cutting the protruding portions of the seal. These tests proved successful. The decision was made to close the PLBDs on the nominal timeline for entry. The PLBDs were successfully closed and locked for entry with no indication that the protruding seal interfered with closure. Postflight inspection found that the seal had separated at the inboard side of the repair splice. There was no indication of damage from entry heating in the area of the separation. See Section 7, Orbiter 2 for further details of this anomaly.

For STS-40 launch, the modified debris containment device attach link system was used for the first time. The results were moderately successful; there were no stud hang-ups, thread impressions, or broaching. Review of launch films indicated that a NASA Standard Initiator (NSI) cartridge escaped from Holddown Post (HDP) #5 and a frangible nut half escaped from HDP #3. Post-recovery inspection found HDP #3 and HDP #5 properly seated, most likely jarred into proper position at water impact. The plunger on HDP #7 was obstructed by a nut half. While these results were somewhat discouraging, there were no plans to alter the modified attach link design prior to the next launch.

Several other anomalies occurred during the mission; however, none were a threat to the safe completion of the STS-40 mission. These included a Reaction Control System (RCS) vernier thruster failing off on its initial firing, a Lithium Hydroxide (LiOH) canister access door was opened with the assistance of tools, a thermal blanket came loose from the tunnel adapter top hatch, S-Band communications problems, and repeated failures of the SLS-1 refrigerator/freezers.

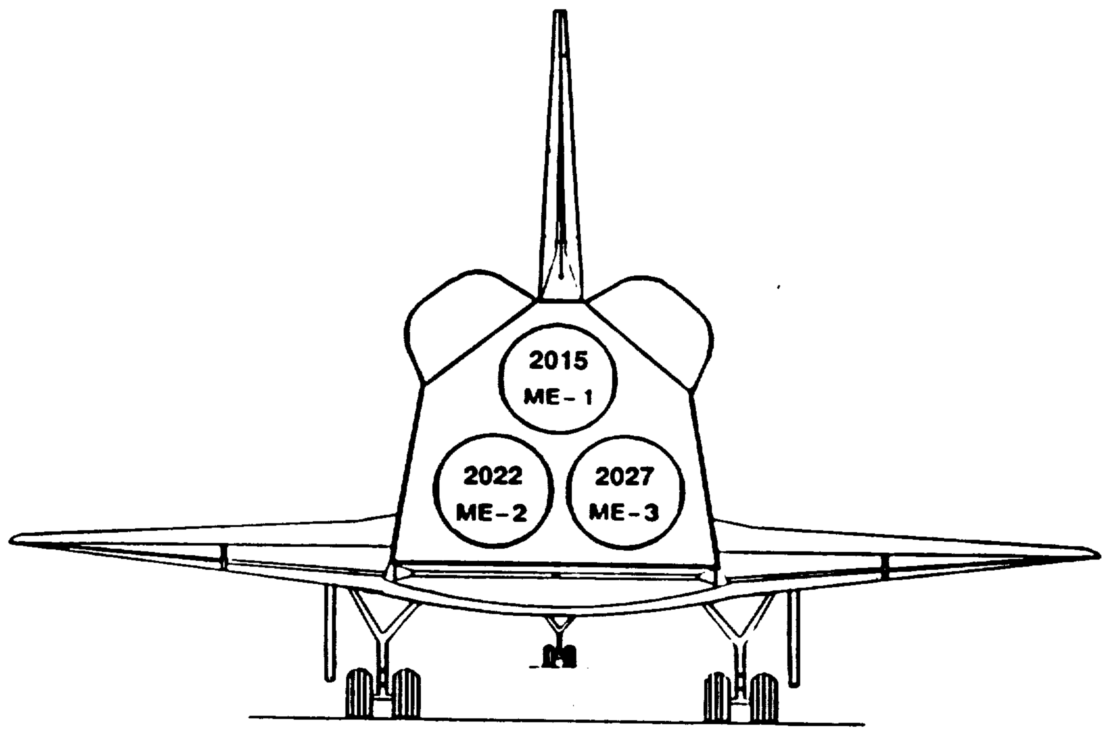
Columbia was cleared for an ontime entry and landing at Edwards Air Force Base on June 14, 1991. STS-40 touched down at 11:40 a.m. EDT on runway 22. The Flight Director reported no anomalies during entry, and all systems looked good. Crew egress was assisted for the first time by a "people mover", a converted airport mobile lounge. The "people mover" was outfitted with a full suite of medical equipment and staff to perform a quick medical examination of the astronauts before their bodies had a chance to adapt to gravity. Four of the 7 crew members remained at the Ames-Dryden Flight Research Facility for an additional week of biomedical testing.

Post-landing inspection of STS-40/OV-102 identified significant heat effects (melting) of the right-hand External Tank (ET) umbilical door centerline latch. Thermal Protection System (TPS) tile near the latch also showed signs of erosion around the edges. The umbilical door pressure seal, however, was not breached. The most probable cause of this anomaly was improper step-and-gap in the area. The step-and-gap around the right-hand ET door was recorded to be correct. See Section 7, Orbiter 4 for further details.

Columbia is the only Orbiter equipped with cameras in the ET umbilical cavity. Postflight review of ET separation film identified a cylindrical object floating away from the left-hand umbilical cavity. Film analysis determined that the object was an umbilical guide-pin bushing. See Section 7, Orbiter 6 for further details.

2.2 Flight/Vehicle Data

- Launch Date: June 5, 1991
- Launch Time: 9:24:51 a.m. EDT
- Launch Site: KSC Pad 39B
- RTLS: Kennedy Space Center, Shuttle Landing Facility
- TAL Site: Ben Guerir, Morocco
- Alternate TAL Site: Moron, Spain
- Landing Site: Edwards AFB, CA, Runway 22
- Landing Date: June 14, 1991
- Landing Time: 8:39 a.m. PDT
- Landing Weight: 226,000 Pounds
- Mission Duration: 9 Days
- Crew Size: 7
- Inclination: 39°
- Altitude: 160 x 150 Nautical Miles/Direct Insertion
- Orbiter: OV-102 (11) *Columbia*
- ET-41
- SRBs: BI-044
- RSRM Flight Set #16
- MLP #3



ENGINE	#2015	#2022	#2027
POWERHEAD	#2107	#2022	#4004
MCC*	#2029	#4006	#2024
NOZZLE	#4001	#2023	#2027
CONTROLLER	F15	F7	F23
FASCOS*	#25	#22	#20
HPFTP*	#4013R1	#6007R1	#4109R1
LPFTP*	#2216	#82207R1	#4005
HPOTP*	#4010R3	#4502R2	#6008R1
LPOTP*	#4401	#2104R1	#4302

* Acronyms can be found in Appendix A.

2.3 Miscellaneous Items of Interest for the STS-40 Mission.

- This was the last mission with the old General Purpose Computers (GPCs), AP-101B, in all 6 GPC positions (5 active, 1 spare). The new GPCs, AP-101S, will be installed on OV-102 during the major modification effort following STS-40.

- The nominal External Tank (ET) impact area was latitude 1.90° North, longitude 140.10° West; approximately 1400 miles southeast of Hawaii. The tumble valve was deactivated for this mission.

- The Hydrogen Dispersal System was installed on Mobile Launch Platform (MLP) #3. This completes the installation of the Hydrogen Dispersal System on all MLPs.

- The 31-psig Gaseous Oxygen (GOX) Flow Control Valves (FCVs) were shimmed to the step III (78%) high-flow position and electrically disconnected so that the FCVs could not transition to the low-flow position. This emulates the final fixed-orifice configuration for the FCV.

- This was the first flight of Space Shuttle Main Engine (SSME) #2015, located in the engine #1 position. SSME #2015 employs the first Multiple Inlet, Single Outlet (MISO) heat exchanger bypass orifice. This change reduces the potential of blockage that was associated with the single inlet, single outlet orifice used on other engines. Blockage of the heat exchanger bypass orifice could lead to a Criticality 1 event. The MISO was certified in 92 hot-fire tests and nearly 40,000 seconds (sec) of operation.

- The modified debris containment device attach link system was installed for the first time on the Holddown Posts supporting the STS-40 stack. This modification is in response to the problems experienced with stud hangup and debris release. The attach link diameter and length have been optimized for desired performance. The silicone rubber isolator geometry has been modified to eliminate premature fracture of the attach link and to eliminate the potential for isolator debris. The plunger geometry has also been modified to provide clearance for the new attach link and rubber isolator.

- The center-forward segment of the Left-Hand (LH) Solid Rocket Motor (SRM) was inadvertently moved in a rail car without the Transportation Loads Monitoring System installed prior to shipment to the Kennedy Space Center. A report indicated that the rail car containing the segment was mistaken for another rail car requiring routine maintenance and was believed empty when it was removed from the Thiokol Corporation (TC) grounds. The TC Senior Material Review Board assessed the potential loads induced during transportation to Salt Lake City, UT for maintenance and approved the segment for flight (Nonconformance Report GFP 405065-01).

- STS-40 RSRM Flight Set #16 was manufactured using Ammonium Perchlorate (AP) produced at the new Western Electro Chemical Corporation (WECCO) AP plant located in southern Utah. WECCO AP was qualified for flight in part through its use in Flight Support Motor-1 test firing.

- Relative to the High-Pressure Oxidizer Turbopump (HPOTP) first stage turbine disc interstage pilot rib cracking, the HPOTPs installed on OV-102/STS-40 were well within established Deviation Approval Request (DAR) limitation. All had less than 5 starts greater than 20 sec after this flight versus the DAR limit of no more than 14 starts greater than 20 sec.

2.4 Payload Data

Spacelab Life Sciences-1 (SLS-1) was the first in a series of dedicated life sciences missions planned to determine the effects of microgravity on humans and animals. The SLS-1 payload was a joint Johnson Space Center (JSC) and Ames Research Center (ARC) effort. The mission focused on the investigation of known problems with manned space flight employing human and animal subjects and the conduct of experiments that addressed fundamental biological problems. The crew performed experiments that explored how the heart, blood vessels, lungs, kidneys, and hormone-secreting glands respond to microgravity; the causes of space sickness; changes in muscles, bones, and cells during the microgravity environment of space flight; and the readjustment to gravity upon return to Earth.

There were some animal experiments on SLS-1 that required the injection of minute levels of radioactive material. The presence of these small radioactive/nuclear sources was reported by NASA Headquarters to the Assistant to the President for Space and Technology in accordance with their directives.

Changes were made to STS-40 Launch Commit Criteria (LCC) to include the Minimum Equipment List for SLS-1. One of the many changes made included the provision for allowing launch with 1 of 2 Auxiliary Electrical Buses. The Auxiliary Electrical Bus powers Fire Detection and Suppression, the SLS-1 vent valve, and experiment ventilation. If one Auxiliary Electrical Bus fails prior to launch, a subsequent loss of the remaining bus would result in the inability to control/extinguish a fire in SLS-1 by venting the module to space. This inability causes a mission scrub and a next Primary Landing Site return. The System Safety Review Panel reviewed this LCC and concurred with its implementation.

Spacelab Program Safety Review Process. The Spacelab safety review process is similar to that of other payloads flown on the Space Shuttle. Hazard Reports (HRs) were developed for the various Spacelab Program elements: the pressurized module, tunnel adapters, pallets, and Get-Away Special (GAS). The Spacelab Program HRs were rebaselined in 1987 as part of the total Space Shuttle Program safety rebaselining effort in preparation for reflight. As Spacelab Program elements are flown (i.e., Astro-1 pallet, SLS-1), design and configuration changes are reviewed to identify potential impact on the Spacelab risk baseline. This review is performed by the Payload Safety Review Panel through the Flight Safety Reviews (FSRs). For the STS-40 SLS-1 mission, the Phase III FSR was held in January 1990. Concerns raised at the Phase III FSR led to 2 Delta Phase III FSRs held in June 1990 and March 1991 to resolve remaining issues. The Spacelab Flight Readiness Review was held on April 26, 1991 and cleared SLS-1 for flight. There were no open safety issues relating to the SLS-1 mission.

Payload Bay

- SLS-1 mission included 18 primary experiments: 10 using human subjects and 8 using animals.

  Experiments Exploring Human Body Capabilities in Space

  - Influence of Weightlessness Upon Human Autonomic Cardiovascular Controls — investigated the theory that lightheadedness and a reduction in blood pressures in astronauts upon standing after landing may arise because the normal reflex system regulating blood pressure behaves differently after having adapted to a microgravity environment. Some crewmembers wore neck chambers to detect blood pressure in the neck.

  - Inflight Study of Cardiovascular Deconditioning — calculated how much blood is delivered by the heart to the body during space flight. This experiment used a non-invasive technique of prolonged expiration and rebreathing — inhaling in previously exhaled gases — to measure the cardiovascular and respiratory changes.

  - Vestibular Experiments in Spacelab — investigated Space Motion Sickness (SMS), any associated changes in inner ear vestibular function during weightlessness, and the impact of those changes postflight.

  - Protein Metabolism During Space Flight — looked at the mechanisms involved in protein metabolism including changes in protein synthesis rates, muscle breakdown rates, and use of dietary nitrogen in a weightless environment.

- Fluid-Electrolyte Regulation During Spaceflight — investigated the known changes of fluid, electrolyte, renal, and circulatory processes in humans as adaptation to the weightless environment. Detailed measurements were made before, during, and after flight to determine immediate- and long-term changes in kidney function; changes in water, salt, and mineral balance; shifts in body fluids from cells and tissues; and immediate- and long-term changes in levels of hormones which affect kidney function and circulation.

- Pulmonary Function During Weightlessness — studied the properties of the human lung without the influence of gravity. The tests were designed to examine the distribution and movement of blood and gas within the pulmonary system and how these measurements compare to normal respiration.

- Lymphocyte Proliferation in Weightlessness — investigated the effect of weightlessness on the activation of lymphocyte reproduction and tested whether there is a possible alteration of the cells responsible for part of the immune defense system during space flight.

- Influence of Space Flight on Erythrokinetics in Man — studied the mechanisms which may be responsible for the decrease in circulating red blood cells or erythrocytes and subsequent reduction in the oxygen carrying capacity of the blood, including the effect of space flight on red blood cell production rate and the role of changes in body weight and plasma volume on red blood cell production.

- Cardiovascular Adaptation to Microgravity — focused on the acute microgravity-induced changes in the cardiovascular structure and function responsible for a common problem during return to normal gravity of orthostatic hypotension or the inability to maintain normal blood pressure and flow while in an upright position.

- Pathophysiology of Mineral Loss During Space Flight — measured the changes which occur during space flight in circulating levels of calcium metabolizing hormones and to directly measure the uptake and release of calcium in the body. Changes in calcium balance during space flight appear to be similar to those observed in humans with osteoporosis.

Experiments on the Functioning of Basic Life Processes Using Animals

- Regulation of Erythropoiesis During Space Flight and
- Regulation of Blood Volume During Space Flight — this combined investigation determined whether changes seen in red blood cell mass and blood volume in crews on previous space flights occur in the rat and if the rat is a satisfactory model for studying microgravity-induced changes in human blood.

- Bone, Calcium and Space Flight — allowed for more precise calculation of the length of flight time required to significantly inhibit bone formation in rats and documented alterations in bone growth patterns and bone-breaking strength in rodents exposed to weightlessness. It will determine whether bone formation returns to normal levels after space flight.

- A Study of the Effects of Space Travel on Mammalian Gravity Receptors — investigated the structural changes that may occur within the inner ear in response to the microgravity of space. The vestibular symptoms from previous space flights included nausea, vomiting, dizziness, and instability when standing.

- Effects of Microgravity-Induced Weightlessness on Aurelia Ephyra Differentiation and Statolith Synthesis — studied the gravity receptors of ephyrae (a type of jellyfish) to determine how microgravity influences their development and function, as well as the animals' swimming behavior.

- Skeletal Myosin Isoenzymes in Rats Exposed to Microgravity — examined how microgravity affects the speed of muscle contractions and should provide additional data to help explain how microgravity affects the speed of muscle contractions and the growth and proliferation of slow-twitch and fast-twitch muscle fibers.

- Effects of Microgravity on Biochemical and Metabolic Properties of Skeletal Muscle in Rats — evaluated energy metabolism in the hind leg muscles of the rats exposed to microgravity using radioactive carbon. In addition, skeletal muscle cells of flight and ground-control animals were compared to assess any changes in the concentration of enzymes that break down glycogen.

- Effects of Microgravity on the Electron Microscopy, Histochemistry and Protease Activities of Rat Hindlimb Muscles — examined muscle tissues of flight and ground-control rodents to look for the shrinkage or death of muscle cells, breakdown of muscle fibers, or degeneration of motor nerves.

- **Get-Away Special (GAS) Experiments** — included 12 experiments

  - Solid State Microaccelerometer Experiment
  - Experiment in Crystal Growth
  - Orbital Ball Bearing Experiment (OBBFX)
  - In-Space Commercial Processing
  - Foamed Ultralight Metals
  - Chemical Precipitate Formation
  - Five Microgravity Experiments
  - Flower and Vegetable Seeds Exposure to Space
  - Semiconductor Crystal Growth Experiment
  - Orbiter Stability Experiment
  - Effect of Cosmic Radiation on Floppy Disks & Plant Seeds Exposure to Microgravity
  - Six Active Soldering Experiments

Middeck

- Physiological Monitoring System (PMS)

- Urine Monitoring System (UMS)

- Animal Enclosure Modules (AEMs)

- Middeck 0-Gravity Dynamics Experiment-0 (MODE-0) — tested basic techniques to aid in the development of the full MODE payload, scheduled for a later flight.

SECTION 3

SAFETY RISK FACTORS/ISSUES IMPACTED BY STS-40 ANOMALIES

This section lists safety risk factors/issues, considered resolved (or not a safety concern) for STS-40 prior to launch (see Sections 4, 5, and 6), that were repeated or related to anomalies that occurred during the STS-40 flight (see Section 7). The list indicates the section of this Mission Safety Evaluation (MSE) Report in which the item is addressed, the item designation (Element/Number) within that section, a description of the item, and brief comments concerning the anomalous condition that was reported.

ITEM		COMMENT
Section 5: STS-39 Inflight Anomalies		
Orbiter 3	Reaction Control System (RCS) vernier thruster F5R fuel injector temperature biased low. IFA No. STS-39-V-03	During firing of STS-39/OV-103 RCS vernier thruster F5R, the fuel injector temperature read 30°F to 40°F lower than the oxidizer injector temperature. This temperature variance should be no more than 10°F. Postflight trouble-shooting determined that this anomaly was caused by poor contact between the temperature sensor and the thruster. On STS-40, vernier jet L5L failed off due to low chamber pressure. L5L was hot-fired on orbit, reselected, and used for the remainder of the STS-40 mission with erratic Chamber Pressure (P_c). See Section 7, Orbiter 6 for further details.
Orbiter 4	Operations (OPS) recorder #2 uncommanded configuration before launch [potential Multiplexer-Demultiplexer (MDM) hybrid circuit failure]. IFA No. STS-39-V-04	During the second STS-39 launch attempt, OPS recorder #2 experienced uncommanded operation; the recorder was discovered "on" and had changed tracks. Testing of OPS recorder #2 was performed prior to launch, with no further anomalies identified. On Flight Day (FD) 7, OPS recorder #2 repeated the prelaunch anomaly. OPS recorder #2 was subsequently reconfigured from the ground and operated nominally for the remainder of the STS-39/OV-103 mission. During STS-40/OV-102, FD 4, OPS recorder #2 experienced an uncommanded mode and track change similar to the STS-39 anomalies. Investigation found that this occurrence was due to an erroneous uplink causing the mode/track change.

	ITEM	COMMENT
<u>Section 6: STS-35 Inflight Anomalies</u>		
Orbiter 4	Payload Bay Door (PLBD) environmental seal debond. IFA No. STS-35-V-16	Postflight inspection of STS-35/OV-102 found a 24-inch piece of the environmental seal teflon material loose between panels #1 and #2 on the right PLBD, at the top. The investigation of the STS-35 anomaly determined that the failure was the result of interference with a ground strap, causing the seal to debond. The anomaly that received the most attention on STS-40 was the loosening of the 1307 bulkhead thermal blankets and environmental seal. This anomaly was caused by air intrusion past the PLBD and 1307 bulkhead during ascent. The environmental seal was separated at a repair splice and was unrelated to the STS-35 anomaly. See Section 7, Orbiter 2 for further details of the STS-40 anomaly.
Orbiter 6	Window W-1 has a 0.15-inch diameter chip. IFA No. STS-35-V-18	Postflight inspection of STS-35/OV-102 windows revealed a chip in window W-1. A "spider web" type crack formation was radiating from the impact point. The crew indicated that they first noticed the chip on FD 6, and it was believed that impact occurred during ascent. During postflight inspection of STS-35/OV-102, window W-5 was found with a ding measuring 0.0162 inch deep x 0.0663 inch long x 0.0668 inch wide. No other window problems were noted. There was no indication by the crew that they noticed this ding on orbit. Plans are to replace W-5 before the next OV-102 mission.

ITEM		COMMENT
Section 6: STS-35 Inflight Anomalies		
Orbiter 8	RCS vernier thruster R5D failed "off". IFA No. STS-35-V-20	During orbital maneuvering, RCS vernier thruster R5D exhibited low 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. RCS vernier thruster L5L failed "off" due to low P_c during the first commanded firing on STS-40/OV-102. L5L was hot-fired on orbit, reselected, and operated for the remainder of the STS-40 mission with erratic P_c. See Section 7, Orbiter 6 for further details.

ITEM		COMMENT
Section 6: STS-35 Inflight Anomalies		
Orbiter 9	Orbiter/External Tank (ET) Liquid Oxygen (LO_2) aft attach/separation hole plugger did not fully extend. IFA No. STS-35-V-21	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. No debris was found on the runway after the ET doors were opened. Similar hole plugger failures occurred on STS-29 and STS-34. There were no attach/separation hole plugger anomalies on STS-40. There was, however, an umbilical separation guide fitting that detached at ET separation, witnessed by OV-102 LH_2 ET/Orbiter umbilical camera. See Section 7, Orbiter 5 for further details.

SECTION 4

RESOLVED STS-40 SAFETY RISK FACTORS

This section contains a summary of the safety risk factors that were considered resolved for STS-40. These items were reviewed by the NASA Safety Community. A description and information regarding problem resolution are provided for each safety risk factor. The safety position with respect to rationale for flight is based on findings resulting from System Safety Review Panel (SSRP), Prelaunch Assessment Review (PAR), and Program Requirements Control Board (PRCB) evaluations (or other special panel findings, etc.). It represents the safety assessment arrived at in accordance with actions taken, efforts conducted, and tests/retests and inspections performed to resolve each specific problem.

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

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

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

SECTION 4 INDEX

RESOLVED STS-40 SAFETY RISK FACTORS

SECTION 4 INDEX - CONTINUED

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	New Criticality (Crit) 1 and 1R2 failure modes have been identified for the Rate Gyro Assemblies (RGAs) on the Orbiter and Solid Rocket Boosters (SRBs). HR No. ORBI-061A {AR} INTG-144C {C} INTG-165A {C} B-50-18 Rev. C-DCN2 {C} ***No Orbiter or SRB RGA anomalies were reported on STS-40.***	Review of recent RGA test data indicated a large output transient, up to 45% of full scale, that lasts approximately 10 seconds (sec) when power to an RGA is lost. It was previously believed that the RGA output would immediately go to zero when power was removed. Redundancy Management (RM) software normally selects the second highest output value from 1 of 4 SRB RGAs for further processing. Post-SRB separation during ascent, and during descent, RM selects the second highest output from 1 of 4 Orbiter RGAs. However, because it is now known that RGA output can stay high for as long as 10 sec after power is removed, the potential exists for RM to select, as the second highest value, erroneous output data from an RGA that lost power. Selection of erroneous data could lead to loss of vehicle control and subsequent loss of the crew and vehicle. Reevaluation of the RGA Failure Modes and Effects Analysis (FMEA) for Orbiter and SRB RGA power circuits, and for the effects of simultaneous loss of power to 2 RGAs, identified Crit 1R2 failure modes for both the Orbiter and SRB RGAs. The first failure is a latent, redundant power feed circuit component (i.e., a remote power controller, a diode failing open, etc.). The second potential failure is loss of a second string redundant path and power feed to another RGA with a non-redundant power source. These 2 failures would result in simultaneous loss of power to 2 RGAs. A Crit 1/1 failure mode associated with the SRB RGAs was identified. No Crit 1/1 failure modes were identified for the Orbiter RGAs. In the case of the SRB RGAs, demate of a single connector (55W1P113/J3) on the Orbiter Master Event Controller (MEC) #2 or in the Orbiter Avionics Bay #5 feedthrough (50W92P299/J99) would result in simultaneous loss of power to 2 SRB RGAs. Additionally, opening of all 3 poles of the 3-pole MEC #2 power toggle switch

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)	New Crit 1 and 1R2 failure modes have been identified for the RGAs on the Orbiter and SRBs.	would also cause simultaneous loss of power to 2 SRB RGAs. Power distribution to the RGAs within the SRBs is isolated and redundant. However, analysis determined that a power bus transient greater then 3 milliseconds (ms) can cause the SRB Multiplexer-Demultiplexer (MDM) to experience a Power-On Reset (POR) and clear all previous commands. This will result in power-down of 2 of the 4 SRB RGAs commanded by the SRB MDM that experienced the POR. Coupled with the RGA power-down output transient, this could provide sufficient data for RM selection and result in a Crit 1/1 condition. No power transients have been experienced to date. A decision was made to cross-strap power to the forward SRB MDMs to alleviate this problem. This fix has some residual risk. If there are failures in 2 MDM current limiters of the same hybrid design, followed by a short circuit at the same regulator output, excessive current would be drawn from both MDM power supplies. In turn, this could cause a momentary power drop on both SRB power buses A and B. This would cause a POR of the aft MDM, resulting in catastrophic shutdown of both SRB Hydraulic Power Units (HPUs) and loss of Thrust Vector Control (TVC). The cross-strapped SRB power configuration was flown on STS-28. It is considered a lower risk than not cross-strapping the SRB MDM power and flying with several potential single-point failures in the Orbiter-side bus B circuitry that can lead to MDM POR and SRB RGA shutdown. There have been no SRB current limiter failures to date. A waiver to fly with the potential current limiter failure mode was approved for STS-39. Two Critical Items List (CIL) waivers, CR S50260D and CR S50260S, were submitted to address the new Crit 1/1 and 1R2 conditions. These waivers were approved for STS-37/OV-104 and STS-39/OV-103. CR S50260D addressed

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)	New Crit 1 and 1R2 failure modes have been identified for the RGAs on the Orbiter and SRBs.	component and power bus failures in the Orbiter that create the Crit 1R2 condition for Orbiter and SRB RGAs. CR S50260S addressed the 2 Orbiter connectors demates/failures that create the Crit 1/1 condition for SRB RGAs. The existing Crit 1/1 CIL for the 3-pole MEC #2 power toggle switch was unchanged by these findings. An update to the CIL was approved prior to STS-40. Flight Rule (FR) 8-47 was prepared to reduce the risk of simultaneous RGA power loss and output of erroneous data. This FR directs the crew to deselect 1 Orbiter RGA when the first failure is detected. Ground test procedures were incorporated to verify the integrity of the SRB RGA backup power logic source during essential power bus tests at T–1 hour. Photographic documentation of the Crit 1/1 connectors was also mandated to ensure proper connector seating. Review of the applicable photographs and video tape determined that these connectors were properly installed on STS-40/OV-102. Additionally, a switch guard was installed over the MEC #2 power toggle switch to preclude inadvertent action by the crew. An effort is underway to provide a design solution to eliminate these failure modes from the system. To date, there have been no failures in the Orbiter and SRB RGA power circuits. There is also a low failure rate for critical power circuit components (i.e., remote power controllers and diodes).

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1 (Continued)	New Crit 1 and 1R2 failure modes have been identified for the RGAs on the Orbiter and SRBs.	Rationale for STS-40 flight was: • Redundant power circuits were tested during normal flow processing. Additional prelaunch tests were identified to verify the SRB RGA backup power logic source. Power to the forward SRB MDMs was cross-strapped to preclude POR conditions in the event of Orbiter power transients. A waiver was approved to fly in this configuration. • RGA power circuit components and connectors have a high reliability . • Photographic documentation of critical connectors demonstrated proper connector installation. • Failure of either connector at the pad is instantly detectable (loss of SRB bus B). • The integrity of the 2 connectors in question is verified on every flow. • A switch guard was installed over the MEC #2 power toggle switch to preclude inadvertent actuation. ***This risk factor was acceptable for STS-40.***

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
2	Upgrade of Engine Interface Unit (EIU) loss of output CIL to Crit 1/1. HR No. INTG-019 {AR} INTG-165A {C} ***No EIU anomalies were reported on STS-40.***	During the process of upgrading an EIU loss of output CIL for the POR condition, analysis revealed that an EIU power supply anomaly could delay Space Shuttle Main Engine (SSME) shutdown for a period up to 80 ms. The occurrence of this failure, coupled with a low-level sensor initiated Main Engine Cutoff (MECO), could cause Crit 1/1 effects. This power supply anomaly could cause the loss of 2 of 3 Main Engine (ME) command channels for 1 command cycle, consequently delaying engine shutdown for up to 80 ms. An 80-ms delay could result in violation of the Interface Control Document (ICD) requirement for 80-pounds (lb) minimum of Liquid Oxygen (LO_2) during SSME shutdown, thereby possibly causing High-Pressure Oxidizer Turbopump (HPOTP) cavitation. Ascent performance data verified that a Low-Level Cutoff (LLCO) of the engines can take place with no failure occurrences and all ascent systems operating within accepted performance dispersions. Since no failures are required to get to the LLCO situation, a single EIU power supply failure, coupled with a LLCO, could result in catastrophic effects (Crit 1/1). EIU failure data indicate that EIUs have experienced 20 power supply POR anomalies; however, no PORs have occurred during flight. An EIU design change has been identified to minimize POR occurrences. A 2-flight waiver for STS-37 and STS-39 was approved. The update to the CIL was approved prior to STS-40.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
2 (Continued)	Upgrade of EIU loss of output CIL to Crit 1/1.	Rationale for STS-40 flight was: • There have been no LLCOs experienced in the Shuttle program. • No EIU power supply anomalies have been experienced during flight. • Acceptable ascent performance dispersions are identified prior to each mission. • The exposure time frame for the EIU failure is small (80 ms). • The worst-case scenario of Crit 1 effects requires a slow shutdown sequence during LLCO along with the 80-ms delay caused by the EIU failure. • The probability of a catastrophic situation is extremely low ($2x10^{-8}$). ***This risk factor was acceptable for STS-40.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
3	Auxiliary Power Unit (APU)/Hydraulic Power Unit (HPU) Quick Disconnect (QD) O-ring material issue. HR No. INTG-016 {AR} ORBI-250 {AR} A-20-24 Rev. B-DCN 5 {AR} SAR S70-0778P2 {AR} *No indications of hydrazine leakage from the APU/HPU QDs were reported on STS-40.*	Hydrazine leakage emanating from a fuel filter bowl QD was noted following an HPU calibration test green run. Investigation found that the O-ring had deteriorated into small particles. Analysis determined that the O-ring material was not Ethylene Propylene Rubber (EPR) as required by the manufacturing drawings. The actual O-ring material has yet to be identified; however, it appears to be Viton. It was estimated that the O-ring in the QD was exposed to hydrazine for only 21 hours (hr) at the time the leak was discovered. The failed O-ring was from a batch of 500 O-rings procured in 1985 by Symetrics from Valley Seal, a distributer. The O-rings were manufactured by Parker Seal. A preliminary review of the manufacturing batch certification documentation revealed that the O-rings were made from the proper material, EPR. At least 77 QDs with O-rings from the suspect batch were shipped from Symetrics to Sundstrand. The following is a breakdown of the location of the QDs and O-rings: • 19 are assigned to HPUs: 3 HPUs on STS-40 have suspect O-rings; Serial Number (S/N) 178, 183, and 192. • 3 are assigned to Orbiter units: APU #3, S/N 306, on STS-40/OV-102 and APU #1, S/N 203, on OV-103 have suspect O-rings. • 46 QDs are at Sundstrand: 12 have never been exposed to hydrazine, and 34 were exposed to hydrazine. • 7 QDs were scrapped; all exposed to hydrazine. • 2 have yet to be located.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
3 (Continued)	APU/HPU QD O-ring material issue.	The STS-40 APU and HPU QDs with suspect O-rings have been exposed to hydrazine for a minimum of 250 hr and 5 hot-fires, with no evidence of failure. A cap with a seal is installed over the QD for flight. The QD flight cap provides redundancy to the QD O-ring against hydrazine leaks. When the flight cap is removed, sniff checks are performed to verify no hydrazine leakage through the QD. QD flight caps are torqued to 20-25 inch-pound (in-lb) when installed. Rationale for STS-40 flight was: • Only 1 of 63 suspect QDs exposed to hydrazine has failed. • There have been no reported failures of Orbiter APU QDs. • QD flight caps provide redundancy against a hydrazine leak. • OV-102 APU #3 passed Sundstrand green-run, acceptance testing, and leak checks. It has 12 months of wet time, 1 flight, and 1 hot-fire with no indication of QD failure. • HPU QDs showed no signs of leakage with long-term exposure to hydrazine. Two of the 3 HPU suspect QDs have 1 previous flight and 1 hot-fire with no evidence of QD failure. ***This risk factor was acceptable for STS-40.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1	Cabin pressure bleed valve anomalies on OV-105. HR No. ORBI-074 {C} ***No cabin pressure bleed valve anomalies were reported on STS-40.***	Three cabin pressure bleed valves installed on OV-105, S/N 5, 6, and 8, recently failed leak tests. A fourth, S/N 7, inspected at the manufacturer, Charlton Technologies, Inc. (CTI), was found to be leaking; however, the leak was within specification. All 4 valves exhibited signs of seal debonding. Valves S/N 1 through S/N 4 had no history of leakage or seal debond. The cause of the debonding was not determined. The bond could be affected by either a lack of primer on the valve or incorrect application of the primer to the valve. Poor adhesion of the S/N 5 through S/N 8 molded seals, and seal leakage, was verified at CTI. Testing to date has been inconclusive in determining proper primer application. Records showed that the seals were molded in accordance with the applicable procedure; however, there was no specific inspection criteria for primer application. A historical comparison of seal material batches to seal leakage indicated no definitive correlation. All seal materials are batch tested for correct material properties. For flight vehicles, cabin pressure bleed valve leak tests at 15 pounds per square inch differential (psid) are performed every 5 flights. Maximum allowed leakage is 25 standard cubic centimeters per minute (sccm). During countdown, cabin pressure drop is monitored, with a 2-psid gross leak limit. Each valve is checked individually after venting the cabin; there is no reverification of leakage. OV-103 valves were last leak tested before STS-41, with no problems noted. OV-104 valves were checked during the STS-37 flow, with no anomalies. Cabin pressure bleed valves on OV-102 were tested during the STS-40 flow.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1 (Continued)	Cabin pressure bleed valve anomalies on OV-105.	Because of the problems found with the OV-105 cabin pressure bleed valve seal anomalies, seals on other pressure valves were inspected. The OV-102 inspection identified the potential for debonding of the butterfly valve (isolation valve) seals on positive-pressure relief valves, S/N 1 and S/N 9. Three of 5 technicians conducting the inspection stated that the seals appeared to be slightly debonded. The seal material in the positive-pressure relief valve is identical to that used in the cabin bleed valve. The positive-pressure relief valve is also manufactured by CTI. The positive-pressure relief valve has 2 sections that work in series and has a criticality of 1R2. The relief section vents at cabin pressure between 15.5 psid and 16.0 psid to prevent overpressurization of the cabin. The inboard section has a motor-driven butterfly valve that provides isolation of the relief section in the event of a relief valve section leak. No failures have been reported to date in the butterfly valve section. Eight incidences of the relief section failing to properly reseat occurred during testing; none occurred in flight. Relief valve operation, reseat pressure, and leakage are verified every 5 flights. A review determined that the OV-102 valves were last verified in May 1989, prior to STS-28. There has been no indication that the relief valves have leaked in the 3 flights since this verification. The butterfly valve operation is verified prior to every flight and was successfully performed on OV-102. Prelaunch, 2-psid cabin integrity tests will detect a failed-open relief section. During flight, onboard delta-pressure/delta-temperature sensors would detect cabin leakage through the butterfly valve, and an abort decision, based on existing Flight Rules, would be made.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	Cabin pressure bleed valve anomalies on OV-105.	Rationale for STS-40 flight was: • OV-102 cabin pressure bleed valves passed all leak tests. • Launch Commit Criteria (LCC) and Flight Rules were in place to account for valve leakage after turnaround testing. ***This risk factor was resolved for STS-40.***
2	STS-37/OV-104 short touchdown at Edwards Air Force Base (EAFB) lakebed runway. HR No. ORBI-325B {AR} ***STS-40/OV-102 experienced no similar problems during landing at EAFB.***	During the April 17, 1991 System Safety Review Panel (SSRP) telecon, STS-37/OV-104 (*Atlantis*) was reported to have touched down more than 600-ft short of the threshold on the EAFB lakebed runway #33. Surface winds at touchdown were 8° at 17 knots (kt), gusting to 21 kt. The headwind component was 10 kt gusting to 13 kt, and a right crosswind component was 14 kt gusting to 18 kt. Touchdown velocity was 166-kt equivalent airspeed at 623 ft with a 5-ft/sec descent rate. During descent, the Orbiter encountered wind shear from 13,000 ft down to 9,000 ft. Wind shear was again encountered at approximately 1,000 ft. The second shear resulted in an 850-ft decrease in range and an airspeed loss of approximately 20 kt. It is believed that both wind shears contributed to the short landing. Energy at the Terminal Area Energy Management (TAEM) interface was nominal. A high tailwind was experienced when entering the Heading Alignment Cone (HAC); however, this tailwind was well within acceptable criteria and vehicle performance parameters. The HAC displayed to the crew was automatically recomputed at 23,000 ft by onboard guidance, causing the HAC to shrink. The

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

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2 (Continued)	STS-37 short touchdown at EAFB lakebed runway.	A review of wind shear effects during other flight regimes, including heavyweight landings and abort site, was performed. All primary abort sites have adequate runway margins (aprons), landing aids, and weather forecasting equipment to ensure safe landings. Current onboard guidance provides adequate landing margin for heavyweight vehicles, and flight rules are in place to ensure optimum margin on short runways. Underrun aprons on all primary landing and abort sites are a minimum of 1,000 ft in length. Transatlantic Abort Landing (TAL) site underrun aprons are walked down for debris inspection prior to potential use. STS-40 was a heavyweight landing vehicle; 20,000-lb heavier than STS-37/OV-104. Simulation, using the actual parameters from STS-37, indicated that STS-40 could make the runway; however, the nose gear slapdown loads would be in the 3-sigma range. The STS-40 crew was briefed on the circumstances surrounding the STS-37 incident and performed simulated landings under the STS-37 conditions prior to launch. Additionally, calls were made to the crew concerning projected g-forces in the HAC and wind shear conditions.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2 (Continued)	STS-37 short touchdown at EAFB lakebed runway.	Rationale for STS-40 flight was: • Flight rules and current landing criteria were found to be adequate, and no changes were made for STS-40. • Communications procedures were updated prior to launch. • Crew training emphasized the importance of following command steering cues from the auto-guidance computer under marginal weather conditions. • For additional awareness, the STS-40 crew was briefed concerning the circumstances surrounding the STS-37 incident, and the corrective action to take should these circumstances recur. ***This risk factor was acceptable for STS-40.***
3	External Tank (ET) door lug clevis cracks on OV-102. HR No. ORBI-302A {AR} ***No anomalies attributed to ET door lug clevis cracks were reported on STS-40. The ET doors closed properly when commanded.***	During borescope inspection of STS-39/OV-103 Orbiter/ET umbilical door cavity thermal curtain installation, a crack was found on the Right-Hand (RH) door (LO_2-side), aft bellcrank lug clevis. This finding led to inspection of all Orbiter ET door lug clevises, including OV-102. Borescope and dye-penetrant inspection found starter cracks on 3 of 4 OV-102 lug clevises. The 4 ET door housings were removed from OV-102 and sent to Rockwell International (RI)/Downey for rework. The modified OV-102 ET door housings were installed on OV-103 to support the STS-39 mission.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)	ET door lug clevis cracks on OV-102.	The ET door housings on STS-40/OV-102 were those removed from OV-105; the housings were originally sent to Kennedy Space Center (KSC), without modification, for installation on STS-39/OV-103. Receipt inspection at KSC found a starter crack, 0.06 inch deep and 0.43 inch long, on the Left-Hand (LH) forward lug clevis. Because of this finding, all 4 OV-105 housings were returned to RI/Downey for modification. At RI/Downey, the lugs were machined off, and all base plates were inspected. The crack was stop-drilled to prevent the starter crack on the LH forward lug base plate from propagating. Doublers were added beneath each housing base plate. A vertical "J" brace was bolted into the 4 housings with Hi-Lok fasteners, replacing the lugs. Bellcranks were modified to provide an additional 0.02-inch clearance at the ET door seal area. After assembly, all drawing tolerances and requirements were verified, including the alignment index marks on the spline shafts and push-rod length measurements. The former OV-105 ET door housings were returned to KSC for installation on STS-40/OV-102. Operational Maintenance Requirements and Specifications Document (OMRSD) functional testing of the RH and LH ET doors was successfully completed.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3 (Continued)	ET door lug clevis cracks on OV-102.	Rationale for STS-40 flight was: • ET door housings installed on OV-102 were modified, and all drawing tolerances and requirements were verified. • Functionality of the OV-102 ET doors was successfully verified in accordance with OMRSD requirements. ***This risk factor was resolved for STS-40.***
4	Wing struts below minimum wall thickness on OV-102. HR No. ORBI-277 {C} ***No anomalies attributed to wing struts were reported on STS-40.***	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. Because of this finding, wing strut tubes on other Orbiters with a margin of safety of less than 35% were inspected on an availability basis. During the STS-40 flow, 20 OV-102 strut tubes were ultrasonically inspected. From this inspection, 10 tubes were found acceptable, and 10 tubes (8 in the LH wing and 2 in the RH wing) were found with below-minimum wall thickness. Five additional strut tubes (not included in the original 20) were found either dinged or dented during access (4 in the LH wing and 1 in the RH wing). Material Review Board (MRB) action was required to disposition the 15 tubes with either below-minimum wall thickness or dents. Four strut tubes were dispositioned to fly "as-is" on STS-40; 3 will be replaced during the OV-102 major modification period after STS-40. The remaining 11 were dispositioned for repair before STS-40.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4 (Continued)	Wing struts below minimum wall thickness on OV-102.	Repair of strut tubes is made by bonding a 0.020-inch thick clamshell over the length of the tube to provide additional strength. Hose clamps and safety wire are installed over the clamshell to further secure it. After repair, all strut tubes are proofloaded according to the flight loads applied at the location of the strut. Prooflоads range from 1,700 lb to 3,800 lb. The 11 OV-102 strut tubes dispositioned for repair had the clamshell installed and have passed proof tests. These 11 were reinstalled on OV-102 for STS-40 and will be replaced during the major modification period. Rationale for STS-40 flight was: • All OV-102 wing strut tubes with a margin of safety less than 35% were inspected. • Those strut tubes found with below minimum wall thickness or damaged have been dispositioned to fly "as-is", or have been repaired. ***This risk factor was resolved for STS-40.***
5	OV-102 Freon Coolant Loop (FCL) #1 flow rate degradation prior to STS-35. HR No. ORBI-275 {C} ***No FCL flow rate degradation anomalies were reported on STS-40.***	In 1983, flow rate in both OV-102 FCLs decayed below the OMRSD limits. At that time, the FCL #1 and #2 pump inlet filters were replaced, with minor contamination found. In July 1989, prior to STS-28, FCL #2 payload flow rate decreased below OMRSD limits. The FCL #2 Radiator Flow Control Assembly (RFCA), Flow Proportional Module (FPM), flow meters, and pump inlet filters were replaced. Prior to STS-35/OV-102, in May 1990, FCL #1 interchanger flow rate dropped below the OMRSD limits with the FPM in the payload position

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	OV-102 FCL #1 flow rate degradation prior to STS-35.	(see STS-35 MSE Postflight Edition, March 15, 1991, Section 4, Orbiter 10 for further details). This flow-rate problem led to the replacement of the FPM and the FCL #1 filter package; FCL #1 flow rate improved to 3050 pounds per hour (lb/hr), within specification limits. However, the FCL #1 flow rate steadily decreased after STS-35/OV-102 power-up at the pad. At liftoff, the FCL #1 flow rate was 2060 lb/hr, well below the 2150 lb/hr Launch Commit Criteria (LCC) limit. A waiver was previously approved to reduce the FCL #1 flow LCC to 1800 lb/hr. Operational workarounds were available in the event FCL #1 flow rate degraded further after launch; FCL #1 operated nominally on orbit. The inlet filter removed in May 1990 was corroded and broken; however, the FCL was not backflushed to remove all remaining contamination. This resulted in clogging of the bypass valve inlet filter on the RFCA. Troubleshooting determined that this clogging was the cause of the FCL #1 flow rate degradation prior to the STS-35 launch. During the STS-40/OV-102 flow, the RFCA was removed and replaced, and the system was backflushed to remove contamination. The FCL #1 flow rate is now 3150 lb/hr. The removed RFCA was sent to the vendor for failure analysis and repair. Initial flow testing indicated a 10.3 psid across the RFCA bypass valve; the acceptance test limit is 3.0 psid. The finding confirmed that the bypass valve inlet filter was clogged.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	OV-102 FCL #1 flow rate degradation prior to STS-35.	Rationale for STS-40 flight was: • FCL #1 flow rate returned to normal after the RFCA replacement. • FCL flow rates were constantly monitored for the remainder of the flow process; any degradation will be detected and assessed. • LCC limits protect against launching with a degraded system. • Workarounds and Flight Rules were in place if FCL #1 degraded after launch. ***This risk factor was resolved for STS-40.***
6	A Main Propulsion System (MPS) 3-way helium solenoid valve failed leak tests on OV-102 during the STS-40 flow. HR No. ORBI-108E {AR} ORBI-129A {C} ***No MPS helium leaks were experienced on STS-40.***	MPS leak tests on OV-102, after STS-35, isolated a leak to a 3-way helium solenoid valve (LV68) used in the 17-inch disconnect latch unlock. Additional tests confirmed the leak and further isolated it to a crack in the valve bellows. Similar 3-way helium solenoid valves are used in 46 locations in the MPS. The valves control helium pressure to open or close pneumatically-operated MPS valves. The concern was that worst-case failure of a helium valve could lead to helium leakage or valve rupture and result in a Crit 1 failure. This failure scenario was believed to potentially deplete the onboard helium supply, resulting in inability to close prevalves at Main Engine Cutoff (MECO) and the potential for main engine turbopump overspeed and explosion. Valve LV-68 was removed for evaluation and replaced. Retests indicated that the replacement valve was not leaking.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	A MPS 3-way helium solenoid valve failed leak tests on OV-102 during the STS-40 flow.	Helium solenoid valve bellows are 2-ply (nickel and copper plies) and are fabricated by an electroforming/electroplating process. Convoluted plies are soldered to end fittings to complete the bellows assembly. In normal operation, the valve bellows assembly is pressurized by solenoid inlet pressure. Internal bellows pressure and spring rate provide the forces necessary to maintain the valve in the closed position when the solenoid is deenergized. Bellows assemblies are proofed at 1550 pounds per square inch (psi), more than twice the operating pressure. Bellows assemblies are reproofed after solder rework; therefore, a valve could be subjected to multiple proof-pressure tests. Initial teardown and inspection of the failed LV-68 valve identified a deformation, or squirm, in the bellows that was caused by buckling instability. Further examination found a circumferential crack on a convolute crown in the bellows. The crack was approximately 0.150 inch long and was on the tension side of the squirm. Leak checks measured the leak rate to be 22 standard cubic feet per minute (scfm) at 300 psi. Extrapolation to the 750-psi operating pressure predicted the leak rate to be 54 scfm. Maximum allowable leakage is 10 standard cubic centimeters per second (sccs) (47 scfm) from the valve vent port. Metallurgical analysis determined that the crack was 80% through the inner bellows ply and was due to fatigue. Final separation in each ply was determined to be due to overload. The crack coincided with a small void in the copper strike. Other cracks were discovered in adjacent convolutes. These cracks propagated from the inner diameter toward similar voids in the copper strike. Stress analysis performed by the vendor indicated that the bellows design was marginal for the 1550-psi proof test. Predictions were that the internal pressure required to initiate squirm

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	A MPS 3-way helium solenoid valve failed leak tests on OV-102 during the STS-40 flow.	deformation is 1551 psi. A more conservative analysis indicated that the pressure needed to initiate squirm was much lower than 1500 psi. The investigation into this failure considered the potential for a lot-related failure mode. The failed bellows was manufactured in 1987 in a lot of 7 bellows; a start-up lot after a 5-year layoff. Records indicated that the failed bellows, and all other bellows in the lot, were subjected to 3 proof tests because of required solder rework. A second bellows from the lot of 7 was installed in the LO_2-side outboard fill and drain valve on OV-103. The outboard fill and drain valve on OV-103 was replaced prior to STS-39 flight. Of the 5 remaining bellows from the lot, 4 were scrapped, and the last was rejected at KSC after the valve failed helium signature tests. This valve too showed signs of squirming. The investigation also determined that solenoid valve certification tests were inadequate for verifying bellows life. Certification testing included 12,000 valve operation cycles; however, only 50 pressure cycles were performed. Pressure cycling was believed to be the primary contributor to bellows squirming and fatigue. There were no other solenoid valve bellows failures recorded during flight or ground checkout. This history included a significant number of valve cycles and operations. Bellows problems were, however, encountered during the production of the OV-105 bellows assemblies in 1989. In this case, initial production runs were scrapped. OV-105 bellows failures were not the result of cracked bellows; however, the bellows were found squirmed and dimensionally unstable after repeated proof-pressure tests. All OV-105 bellows were reworked because of solder problems induced by numerous process and personnel deficiencies at the vendor.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)	A MPS 3-way helium solenoid valve failed leak tests on OV-102 during the STS-40 flow.	Prelaunch and operational procedures are in place to control potential helium leaks through the valve bellows. Prior to launch, excessive helium loss would be detected by the Hazardous Gas Detection System (HGDS). If aft compartment helium concentrations exceeded the 10,000-parts per million (ppm) LCC limit, the launch would be scrubbed. After launch, helium tank pressures are monitored by the Caution and Warning System (CWS). An alarm would sound if helium tank pressures drop below 3,800 psi. If this was to occur during ascent, the crew would be required to manually close isolation valves LV7 and LV8 to conserve helium. LV7 and LV8 would be reopened along with LV10 (engine crossover) at MECO minus 30 sec. These actions are intended to conserve sufficient helium stores to effect engine prevalve closure at MECO. JSC analysis has determined that the worst-case leakage in flight due to a ruptured bellows is manageable. The leak rate, restricted by a 0.0930-inch diameter passage in the valve, was calculated to be 260 scfm. This leak rate would not deplete the helium supply during ascent; therefore, sufficient helium would remain in the system to shut engine prevalves at MECO. Additionally, a 260-scfm leak would not provide sufficient helium to overpressurize the aft compartment.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)	A MPS 3-way helium solenoid valve failed leak tests on OV-102 during the STS-40 flow.	Rationale for STS-40 flight was: • The 3-way helium solenoid valves had a highly-reliable history prior to the discovery of cracked bellows on OV-102. This history included 46 valves installed and operated on 38 Shuttle missions. • No bellows from the suspect lot were installed on OV-102 at the time of STS-40 launch. • Processing flow leak checks and prelaunch HGDS monitoring have the capability to identify leaks. If a leak were to occur after launch, CWS monitoring would alert the crew to take action to conserve the helium supply. ***This risk factor was acceptable for STS-40.***
7	Gaseous Hydrogen (GH_2) Flow Control Valve (FCV) weld crack found on OV-103. HR No. ORBI-306 {AR} ***No GH_2 FCV anomalies were reported on STS-40.***	During STS-39/OV-103 preparations, a small leak was detected at the engine #1 GH_2 FCV housing during mass spectrometer leak tests of the OV-103 GH_2 pressurization system. These tests were performed as part of the investigation into the high Hydrogen (H_2) concentration measured in the OV-103 aft compartment during STS-41 ascent. The leak was measured at 2.3×10^{-6} sccs; in excess of the 1×10^{-6} sccs specification limit. Initial calculations indicated that this leak rate was sufficient to account for the H_2 concentrations measured during STS-41; however, a more formal calculation of potential leak rates is in work. Analysis indicated

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)	GH_2 FCV weld crack found on OV-103.	that a worst-case leak through a circumferential crack would not provide sufficient H_2 to reach aft compartment flammability concentrations. Initial examination of the FCV found the leak source to be a 3/8-inch crack in the FCV housing outlet tube weld. This weld is a sealing weld only and provides no structural integrity. Examination of the cracked FCV housing outlet tube weld at Rockwell International (RI) indicated the weld to be of good quality (good penetration, no evidence of material defect, and good weld blend). Structural analysis determined the failure mechanism to be High-Cycle Fatigue (HCF). The source of the fatigue was not determined; however, loads induced in the high-vibration environment was the leading candidate. Investigation into potential vibration sources is underway. Evaluation of GH_2 and Gaseous Oxygen (GO_2) qualification FCV housings for similar fatigue conditions is in work. Leak tests and Nondestructive Evaluation (NDE) methods will be employed on the qualification housings. Visual inspection of OV-102 FCV outlet tube welds by MPS engineers found no signs of cracks. Mass spectrometer leak checks were performed with no problems or out-of-specification leaks found.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)	GH_2 FCV weld crack found on OV-103.	Rationale for STS-40 flight was: • Mass spectrometer and visual inspections found no indication of leaks in OV-102 FCVs. • A worst-case circumferential crack would not provide sufficient H_2 to reach aft compartment flammability concentrations. ***This risk factor was resolved for STS-40.***
8	OV-102 left Orbital Maneuvering System (OMS) Reaction Control System (RCS) helium isolation valves found with an internal leak. HR No. ORBI-111 {C} ***No degradation in OMS/RCS performance was experienced during the mission that could be attributed to helium isolation valve internal leakage.***	During OMS/RCS functional testing, an internal leak in the left RCS Oxidizer System helium isolation valves, LV 202 and LV 204, was discovered. The measured leak was 110,000 standard cubic centimeters per hour (scch) at 1750 psid, exceeding the 200-scch OMRSD limit. The helium isolation valves are used to isolate a leaking or failed-open helium regulator. The system is designed to operate from a single, good helium regulator leg. During ascent and entry, the helium isolation valves are commanded open to maintain pressure in the RCS propellant tanks. On orbit, 1 helium supply leg is closed to acquired helium regulator performance data. The RCS propellant management system design also includes relief valves to protect against propellant tank overpressurization. The RCS propellant management system can tolerate an external helium pressurization system leak of 1350 scch with no impact to the mission providing there are thruster firings from the affected RCS propellant tank for 5 days and no other actions are taken to reduce propellant tank pressurization.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8 (Continued)	OV-102 left OMS RCS helium isolation valves found with an internal leak.	KSC performed helium regulator flow tests to clean the isolation valve seats in the event contamination was blocking the valves from closing. The isolation valves were also commanded closed under flow conditions in an attempt to hard-seat the valves. The primary and secondary helium regulators and relief valves were tested and found to meet OMRSD requirements. To reduce the potential for RCS propellant tank overpressurization prior to launch, small amounts of propellant can be transferred from the RCS to the OMS propellant tank. On orbit, attitude control using the RCS in the crossfeed configuration can also reduce propellant tank pressure. A waiver was approved to allow STS-40/OV-102 to fly with this internal helium leak through the isolation valves. Leaking helium isolation valves have been waived for previous missions provided both the primary and secondary regulators are functioning within OMRSD parameters; STS-40/OV-102 regulators were successfully tested. There have been no recorded failures of helium regulators on the oxidizer side of the RCS propellant system.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8 (Continued)	OV-102 left OMS RCS helium isolation valves found with an internal leak.	Rationale for STS-40 flight was: • There are procedures in place in all flight regimes to reduce RCS propellant tank pressure in the event a pressure increase resulting from the leaking helium isolation valves is experienced. • Relief valves on the RCS propellant tanks provide a safeguard against uncontrolled overpressurization. The left RCS propellant tank relief valves were tested to verify proper function. • Primary and secondary helium regulators were tested to verify proper function. There are no recorded failures of helium regulators on the oxidizer side of the RCS propellant management system. ***This risk factor was acceptable for STS-40.***
9	OV-102 flight-critical data bus #3 wiring found damaged during midbody closeout. ***No anomalies were attributed to the prelaunch implemented design, jumpering around the damaged critical data bus #3 wiring.***	The OV-102 flight-critical data bus #3 wire harness was found damaged during closeout of the midbody. Damage was limited to the braided shield. The damaged wiring was repaired with electrical tape. The repaired wire harness failed several high-pot tests (1,500 volts) performed after midbody closeout and installation of the SpaceLab tunnel. This rendered the damaged area of the flight-critical data bus #3 wire harness inaccessible.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9 (Continued)	OV-102 flight-critical data bus #3 wiring found damaged during midbody closeout.	A workaround was developed that proposed to use a spare Orbiter Experiments (OEX) data bus. This proposal was reviewed by the Orbiter Project and approved. To accomplish this, KSC jumpered around the damaged flight-criticaldata bus. Z-splices were made in the forward Environmental Control and Life Support System (ECLSS) bay and in the aft compartment. The damaged data bus wire harness was capped and stowed. This approach required connector demates at the data bus couplers only. Electrical impedance and Electromagnetic Interference (EMI) characteristics were checked and verified not to be compromised. All other flight-critical data bus #3 OMRSD checkouts were successfully completed. Repairs to the flight-critical data bus #3 wire harness will be made during the OV-102 major modification period following STS-40. Rationale for STS-40 flight was: • KSC successfully implemented a design that jumpered around the damaged wire harness. • All OMRSD tests were successfully completed. There was no degradation in data transfer performance. ***This risk factor was resolved for STS-40.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
10	STS-40/OV-102 Left-Hand (LH) Main Landing Gear (MLG) valve did not transition to "off" during gear retraction. HR No. ORBI-182 {AR} ORBI-188 {C} *No MLG anomalies were experienced after disconnecting the Ground Support Equipment (GSE) prior to MLG retraction for flight closeout.*	During landing gear retraction-extension functional testing with the C70-0894 GSE unit connected, the LH MLG valve indication did not transition to "off" when retracted. Normal operation for the valve is to stay open ("on") during gear extension and closed ("off") during retraction. Both the RH MLG and Nose Landing Gear (NLG) indications transitioned nominally to "off". LCC requires 3 "on" indications; valve indications are monitored from the start of tanking to T–31 seconds (sec). Investigation into this problem determined that the C70-0894 GSE electrical ground configuration was incorrect. The GSE was disconnected, and gear retraction was performed; all valve indications were nominal. Data retrieval showed the landing gear dump control valve indication going "on" when it should have been "off" during gear retraction. This type of improper indication has been historically correlated to an electrical ground configuration problem. Using Breakout Boxes (BOBs) at the valves and the C70-0894 GSE unit, valve operation was performed while probing connector pins to determine/detect voltage output to the Multiplexer-Demultiplexer (MDM). The electrical ground configuration was varied in an attempt to simulate the configuration during the original test. Technicians were unable to recreate the encountered problem. The most probable cause of the problem was improper grounding of the C70-0894 GSE setup.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
10 (Continued)	STS-40/OV-102 LH MLG valve did not transition to off during gear retraction.	Rationale for STS-40 flight was: • Landing gear valve indications performed nominally after the C70-0894 GSE unit was removed. Valve indications were "on". • Experience indicates that the problem was most likely in the electrical ground configuration. ***This risk factor was resolved for STS-40.***
11	OV-105 T-handle access door for the emergency egress window jettison opened during ferry flight. HR No. ORBI-142A {AR} ***No T-handle anomalies were reported on STS-40.***	During the initial leg of the OV-105 ferry flight from Palmdale to KSC, the T-handle access door was found unlatched. The door was re-latched prior to the continuation of the ferry flight. The initial concern was that the OV-105 T-handle access door was rigged, latched, and closed-out prior to the ferry flight in accordance with Operations and Maintenance Instructions (OMIs) used to secure T-handle doors on all Orbiter vehicles; however, it was later reported that a RI/Palmdale procedure was used to close out the OV-105 T-handle access door. The door was re-secured at El Paso, Texas, after the first leg of the ferry flight and prior to continuing on to KSC. It was later learned that, following proper closeout of the T-handle access door, a request was made to identify and record the part number of a safety pin that was installed in the T-handle for the ferry flight. To access the part number, the access

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)	OV-105 T-handle access door for the emergency egress window jettison opened during ferry flight.	door had to be opened. The technician who opened the access door to record the safety pin part number admitted that he had not followed procedures in securing the door. Photographs taken after the first T-handle access door closeout showed the door closed. A subsequent photograph taken of another operation on May 1, 1991, showed that the door was partially open. This second photograph was taken prior to the ferry flight. The T-handle access door is used only in the contingency that the Orbiter has crashed or landed and the crew requires assistance to egress. Emergency ground crews can assist in Orbiter crew egress by opening the access door and pulling the T-handle to release the pyrotechnic device used to blow-open the LH overhead emergency egress window. If the T-handle access door were to open in flight, the result could be the exposure of the Orbiter structure to excessive, local aerodynamic heating during ascent or reentry. There is also the potential that the resulting high temperature could activate the pyrotechnic device and jettison the egress window in flight. These events have the potential for loss of vehicle and crew. The design of the T-handle access door linkage and locking mechanism is such that the door cannot be latched by simply pushing it into the closed position. The positive action of pushing the release plunger while pushing the door to the closed position allows the dog to move into position around the latch. Pushing the door alone will not move the dog around the latch. It is believed that when the last technician closed the access door, he simply pushed the door flush. The inherent interference fit between the door installation and the Felt Reusable Surface Insulation (FRSI) blanket gives the impression that the door is closed.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)	OV-105 T-handle access door for the emergency egress window jettison opened during ferry flight.	Visual inspection of proper T-handle access door position is performed prior to every launch. Functional open/close, latch/unlatch, and push force verification tests are performed every 5 flights. Door seals that create the preload on the door latch mechanism are also inspected every 5 flights. There is currently no inflight capability to determined if the access door is open. Stress analysis indicated that the load on the T-handle access door latch plunger imposed by launch vibration is 7.2 pounds-force; (lbf) the force required to overcome the seal preload and unlatch the door is 11.9 lbf. All T-handle access door components have been designed with a minimum factor of safety of 1.4. Because STS-39/OV-103 was on flight when the OV-105 T-handle access door was found open during ferry, a review was made to determine when the OV-103 T-handle access door components were inspected. It was determined that the OV-103 fifth-flight OMI inspection was performed prior to STS-31, 2 flights prior to STS-39. The OV-103 T-handle access door was found to be latched in the proper position during post-STS-39 inspection. A review also determined that the last time the STS-40/OV-102 T-handle access door components and function were verified was prior to STS-28. No anomalies were reported with the T-handle access door on any of the last 3 OV-102 flights. An inspection of the STS-40/OV-102 T-handle access door since the OV-105 incident determined that the door is closed and the plunger is in the correct configuration.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)	OV-105 T-handle access door for the emergency egress window jettison opened during ferry flight.	Rationale for STS-40 flight was: • The OV-105 incident was caused by improper securing and closeout of the T-handle access door. • OV-102 has flown 3 flights since the last T-handle access door inspection and functional test, with no anomalies reported; inspection and test is performed every 5 flights. • Recent inspection found the STS-40/OV-102 T-handle access door in the proper closed configuration. ***This risk factor was resolved for STS-40.***
12	Flight Critical (FC) Multiplexer-Demultiplexer (MDM), Flight Aft-2 (FA-2), indicated anomalous operation. HR No. ORBI-038A {AR} ***No MDM anomalies were reported during the STS-40 mission.***	During final preparation for STS-40/OV-102 launch, MDM FA-2 identified 2 consecutive Input/Output (I/O) errors. The I/O errors resulted in a Programmable Read-Only Memory (PROM) sequence bypass. Additional I/O errors were experienced during subsequent FA-2 monitoring while troubleshooting the original anomaly. The Built-In Test Equipment (BITE) status register indicated bits 3 and 4 were set after the first 2 consecutive I/O errors; bit 3 indicated "operation requested on nonexistent channel," and bit 4 indicated that the MDM was "unable to transfer to/from IOM" (Input/Output Module). The concern with these indications was that bit 3 was set. In FA-2, there are no non-existent channels; all 32 available channels are used.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
12 (Continued)	FC MDM, FA-2, indicated anomalous operation.	FA-2 is 1 of 8 FC MDMs and is assigned to FC data buses #2 and #6: 2 of 8 FC data buses. FA-2 is 1 of 4 redundant aft FC MDMs that function to multiplex and demultiplex critical commands between the General Purpose Computers (GPCs) and the Engine Interface Units (EIUs), Main Event Controllers (MECs), and Aerosurface Amplifiers (ASAs). FC commands passed through FA-2 include Thrust Vector Control (TVC), body flap movement, elevon movement, and rudder/speed brake control. Additionally, FA-2 provides the single, non-redundant link between the Orbiter Rate Gyro Assembly (RGA) #2, and the left and right RGA #2 on the Solid Rocket Boosters (SRBs). While the 3 links provided by FA-2 between the GPCs and the Orbiter RGA #2 and both SRB RGA #2 are not redundant, functional rate-data redundancy is provided through the 3 additional Flight Aft (FA) MDMs and the associated 3 Orbiter RGAs and 4 SRB RGAs (2 per side). A single erroneous output of a FA MDM command can enable 1 or more Reaction Control System (RCS) thruster firings during on-orbit proximity operations that can lead to loss of crew and vehicle. Multiple FC MDM failures on the same channel can also to lead to loss of crew and vehicle. Internal functional redundancy is also available in the MDMs. There are 2 Multiplexer Interface Adapters (MIAs) on the serial data bus side, 2 Sequence Control Units (SCUs), and 2 Analog-to-Digital (A/D) converters within each MDM. An A/D converter, a SCU, and a MIA are paired to provide functional redundancy through 1 of 2 MDM ports. The 16 IOMs are not, however, redundant in a MDM. If a failure is found in a paired A/D converter, SCU, and MIA string, the MDM can be "port-moded" to the remaining functional string.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
12 (Continued)	FC MDM, FA-2, indicated anomalous operation.	In the case of the FA-2 MDM anomaly on STS-40/OV-102, consideration was given to the potential for port-moding FA-2 and flying with reduced redundancy. Isolation of the I/O error cause to a single string would have offered the alternative of port-moding FA-2 to removal and replacement. Because there was not sufficient time to collect the data required to isolate the cause of the anomaly, the decision was made to remove and replace MDM FA-2 prior to flight. Rationale for STS-40 flight was: • MDM FA-2 was removed and replaced. Testing to verify functionality of the replacement FA-2 was successfully completed. ***This risk factor was resolved for STS-40.***
13	General Purpose Computer (GPC) #4 failure on STS-40/OV-102. HR No. ORBI-066 {AR} ***No further GPC problems were reported on STS-40. This was the last flight of the old GPCs, AP-101B.***	During final STS-40 launch preparations on May 21, 1991, GPC #4 crashed. An I/O error message, "Shuttle Software Interface Processing (SSIP) Inter-Computer Channel (ICC) Timeout", was posted on the GPC #4 Fault Summary Page just prior to the crash. A memory dump identified that the Central Processing Unit (CPU) stopped processing, indicating a hard failure. Isolation to either the CPU or the Input/Output Processor (IOP) could not be readily determined from the error message or the memory dump; therefore, the CPU box, the IOP box, and the interconnecting wire harness were removed and replaced. GPC #4 is 1 of 5 AP-101B GPCs installed on OV-102. STS-40 was the last flight of the old AP-101B GPC configuration; AP-101S GPCs will be installed on OV-102 during the major modification period after STS-40.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
13 (Continued)	GPC #4 failure on STS-40/OV-102.	During removal of the GPC #4 CPU, pin #24, on CPU connector J2, was found bent. Pin #24 was bent to a point that it touched pin #14 in the same connector. Preliminary investigation by IBM determined that the bent pin could not have caused the GPC #4 hard failure. GPC #4 had not been replaced since reflight (pre-STS-28/OV-102), and there is no record of actions to demate the J2 connector. There also have been no problems experienced with GPC #4 since reflight. IBM believes that the bending of pin #24 occurred during the recent removal of GPC #4. Rationale for STS-40 flight was: • GPC #4 CPU, IOP, and interconnecting wire harness were replaced. Retest of the newly installed GPC #4 components was successfully completed. ***This risk factor was resolved for STS-40.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14	Main Propulsion System (MPS) cryogenic temperature transducer failure. HR No. INTG-023 {AR} ME-C2 (All Phases) {C} ME-D2 (All Phases) {C} ***There were no problems reported on STS-40 associated with the temperature transducers that remained in the MPS or with the plugs installed on the LH_2 side of the MPS.***	During the STS-35/OV-102 Hydrogen (H_2) leak investigation in September 1990, an MPS cryogenic H_2 temperature transducer was found to exhibit leakage during helium mass spectrometer leak checks. The temperature transducer, Part Number (P/N) ME449-0013-0021, S/N 105, was located in the engine #3 Liquid Hydrogen (LH_2) feedline. There are 9 similar temperature transducers of varying P/N-dash numbers in the MPS feedlines leading to the Space Shuttle Main Engines (SSMEs) on each Orbiter; 4 on the LH_2 side, 5 on the Liquid Oxygen (LO_2) side of the MPS. S/N 105 -0021 transducer was removed and sent to the vendor, RDF Corporation of New Hampshire, for failure analysis. Due in part to logistical problems, the results of the RDF failure analysis were not available until May 20, 1991. The results indicated that there were full, 360° circumferential cracks in the transducer sheath-to-mandrel weld joint and in the necked-down area of the mandrel. The concern was that circumferential cracking to the extent seen on the S/N 105 -0021 transducer could lead to losing the 2-inch mandrel/sheath transducer tip into the oxidizer or fuel feedline during SSME operation. The original MPS configuration located 1 temperature transducer in each of the 3 LH_2 and 3 LO_2 12-inch feedlines to the SSMEs just prior to the inlet of the Low-Pressure Fuel Turbopumps (LPFTPs) and the Low-Pressure Oxidizer Turbopumps (LPOTPs). There are no filters or screens to prevent a piece of temperature transducer from entering the LPFTP or LPOTP. If a portion of the temperature transducer broke off and entered the LPFTP or LPOTP, the results would be catastrophic. The cracks found on the S/N 105 -0021 transducer in May 1991 led to the requirement to remove and x-ray the OV-102 and OV-103 MPS temperature transducers. All spare temperature transducers available at KSC were also acquired

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)	MPS cryogenic temperature transducer failure.	for x-ray inspection. X-ray inspection performed at the Shuttle Logistics Depot found that the only other -0021 temperature transducer, S/N 127, removed from STS-40/OV-102 showed indications of cracking in the sheath-to-mandrel weld. Four -0021 temperature transducers removed from OV-103 for x-ray inspection were either found with similar circumferential cracks in the sheath-to-mandrel weld or with indications of cracking. To date, 6 of 17 -0021 MPS temperature transducers have been found with weld cracks. No other ME449-0013 dash-number temperature transducers x-rayed to date have been found with cracks. Except for S/N 105, there have been no other -0021 transducers found with cracks in the necked-down area of the mandrel. The -0021 temperature transducers are used exclusively on the LH_2 side of the MPS system. The design of cryogenic temperature transducers now used in the MPS was based on cryogenic temperature transducers first used on the Apollo/Saturn Program. The -0021 transducer configuration replaced the -0018 temperature transducers that were used in the early stages of the Space Shuttle Program. Early problems with the electronics assembly in the -0018 transducers led to a design change which resulted in the -0021 configuration. In addition to the -0018 and -0021 transducers used on the LH_2 side of the MPS, -0017, -0020, and -0022 temperature transducers are used on the LO_2 side of the MPS. All 5 configurations have similar sheath-to-mandrel designs in the tip of the transducer. However, there are significant differences in the sheath-to-mandrel weld penetration across the transducer configurations. All P/N ME449-0013 transducer configurations were manufactured to the same automated weld schedule. For the sheath-to-mandrel interface, the weld schedule requires a minimum weld penetration of 0.005 inch. RDF Corporation shop practices resulted in a weld penetration no greater than 0.010 inch for all transducer configurations. Variances in weld penetration

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)	MPS cryogenic temperature transducer failure.	percentage among the transducer configurations resulted from different sheath thickness requirements at the weld. The -0018 had a sheath thickness of 0.010 inch, resulting in weld penetrations of 50% to 100% (0.005 inch to 0.010 inch). When the same weld schedule was applied to the -0021 configuration, with a sheath thickness of 0.040 inch, the resulting weld penetration ranged from 12% to 25%. The added sheath thickness reduced the structural integrity of the sheath-to-mandrel weld between the -0018 and -0021 configurations. The LO_2-side transducers (-0017, -0020, and -0022) have the same sheath thickness requirement, 0.020 inch. This results in a weld penetration of 25% to 50%. Since finding circumferential cracks on the -0021 S/N 105 transducer, a stress/fracture sensitivity analysis of all transducer configurations was performed. The analysis considered temperature change experienced by the transducer during tanking and SSME operation (ΔT = -493°F for LH_2 transducers, ΔT = -363°F for LO_2 transducers) and material thermal expansion/contraction property differences. The analysis compared thermally-induced stress to critical fracture stress to determine the design margin of the weld in all configurations. The results indicated that thermally-induced stress can exceed the stress required to induce total fracture of the 10% weld in the -0021 configuration. Thermally-induced stresses on the -0017, -0020, and -0022 transducers have been determined through analysis to be insufficient to lead to weld fracture, given the requirement for minimum weld penetration of 25%. The analysis results support the findings of the x-ray inspections performed on OV-102, OV-103, and spare temperature sensors. The thermal analysis did not, however, identify sufficient forces to result in the circumferential cracking found in the -0021 S/N 105 necked-down area of the mandrel. Scanning Electron Microscope (SEM) inspection of the cracks found in

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)	MPS cryogenic temperature transducer failure.	both the sheath-to-mandrel weld and the necked-down area of the mandrel was performed. The SEM inspection revealed uniform, brittle fractures around the circumference of both cracks. No ductility was observed in the sheath-to-mandrel weld crack. The sheath-to-mandrel crack was indicative of a failure at cryogenic temperatures. Indications of ductility in the form of shear dimples were, however, found in the mandrel crack. This finding demonstrated that the crack was initiated by a force other than thermally-induced stress at cryogenic temperatures. An impact test using a punch and hammer and a coupon of identical mandrel material was performed at room temperature to determine if failure could be manually induced. The results of the impact test duplicated the crack found in the necked-down area of the mandrel. This test demonstrated that the -0021 S/N 105 mandrel crack most likely was caused by forces outside of the operational environment. The investigation into this issue also led to Program action to determine the rationale for reinstalling temperature transducers near the inlet of the LPFTP and LPOTP. The Propulsion Systems Integration Group (PSIG) was convened to resolve this action. The PSIG determined that only a minor revision to the SSME throttle-down Flight Rule 5-50 was required to allow removal of the LH_2 temperature transducers at the inlet of the LPFTP. LPFTP LH_2 inlet temperatures can be replaced with a constant temperature value determined from flight experience and updated in real-time based on the LH_2 manifold temperature transducer; the fourth LH_2-side transducer. Initial review found it would not be infeasible to remove the LO_2-side temperature transducers. LO_2 SSME inlet temperature transducers are used as the sole source for protecting engine start box requirements for LO_2 temperatures. Further analysis is required to identify alternative approaches to protect the engine start box in the event that residual risk associated with installing temperature transducers near the LPOTP inlet is not

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
14 (Continued)	MPS cryogenic temperature transducer failure.	acceptable. For STS-40/OV-102, the 3 LH_2 temperature transducers installed at the LPFTP inlet were replaced with flight-certified plugs. The remaining 6 temperature transducers were installed following x-ray inspection to verify that no sheath-to-mandrel weld cracks were evident. A -0018 LH_2 temperature transducer was used in place of the -0021 LH_2 manifold transducer to further reduce the risk. A change to the LCC relating to MPS transducers was approved to reduce the risk of launching with a structurally-failed transducer. The previous LCC requires 2 of 3 engine inlet transducers to be operational prior to launch. The revised LCC will screen for structural failures by monitoring for offscale high, low, or erratic indications from the start of stable replenish to T−31 sec for all MPS transducers. Any anomalous indications will result in a launch scrub and troubleshooting. Rationale for STS-40 flight was: • All MPS temperature transducers installed on OV-102 successfully passed x-ray inspection. • Flight-certified plugs were installed in place of the LH_2 temperature transducers near the LPFTP inlet. • Leak check on all newly installed plugs and transducers was successfully performed. • The revised MPS transducer LCC will not allow a launch with anomalous transducer indications. ***This risk factor was acceptable for STS-40.***

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	STS-37 Right-Hand (RH) cowl bond line blowpaths. HR No. BN-06 Rev. C {C} ***No cowl bond line blowpaths were reported on ST-40.***	The STS-37 Redesigned Solid Rocket Motor (RSRM) underwent a special detailed postflight disassembly inspection as a result of Test and Evaluation Motor (TEM)-7 Inner Boot Ring (IBR) and fixed-housing ablative liner debond issues. STS-37 nozzle examination revealed 2 blowpaths on nozzle joint 2, cowl-to-insulation axial bond line, at the 286° and 295° locations. The 2 blowpaths occurred on the RH RSRM nozzle. Examination revealed no heat effects on the phenolic, adhesive, metal, or paint. The inspection also revealed no structural effects since the cowl assembly is structurally maintained by the cowl bond line and pinned in place with 36 0.375-inch diameter steel pins. In the event of a cowl bond line failure, the cowl will remain attached to the cowl housing. The Factor of Safety (FOS) without the bond is 2.15, well above the 2.0 program requirement. Cowl bond lines were examined on both RSRM nozzles, and typical soot patterns were noted on the nozzle joint 2 interface. Seventeen of 34 post-fired RSRM inspected nozzles displayed Room-Temperature Vulcanizing (RTV) blowpaths in nozzle joint 2. Soot to the primary seal is typical, and seal damage was never observed. Cowl housing thermal damage was also never observed. Rationale for STS-40 flight was: • The observed STS-37 cowl bond line condition resulted in no adverse thermal, structural, or sealing effects. • The FOS without the bond is 2.15, exceeding the 2.0 program requirement. ***This risk factor was resolved for STS-40.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
2	STS-39 Left-Hand (LH) nozzle throat assembly bond line void. HR No. BN-07 Rev. C {C} ***No nozzle throat assembly bond line voids were found on disassembly of STS-40 SRMs.***	Detailed nozzle bond line inspections have been required as a result of the TEM-7 IBR and fixed-housing ablative liner debonds. Increased attention given to the STS-39 SRMs led to the discovery of a bond line void in the LH nozzle throat assembly. The bond line void ran approximately 3.5 inch axially by 50 inch circumferentially near the aft end of the throat assembly. There was no sign of hot-gas penetration into the voids. Inspection of the RH nozzle was completed with no large voids found; however, a series of unconnected finger voids were also found. No significant bond line voids were found either on STS-37 nozzle throat assemblies. This was the first time that such a detailed inspection had been performed in this area on a flight or test nozzle. The throat phenolic ring is assembled prior to the throat inlet phenolic ring. Both phenolic rings use bond line shims to help center the ring and to control bond line thickness. The potential exists for a thin bond line void at the aft end of the throat phenolic ring, away from the shim locations. Bond line integrity at the ends of the throat and throat inlet are assured by inspection and are repaired when necessary. Bond line voids, extending the full length of the bond, would be obvious and would not escape post-assembly inspection. A review of throat assemblies from 4 flight sets, STS-26 through STS-30, and 6 static test motors found no similar voids. A limited assessment, performed using photographs of char and erosion samples for 10 flight sets, STS-26 through STS-35, also showed no indication of extensive voids. It appears from this limited review that the STS-39 LH throat assembly void was a first-time occurrence. There is no record or indication of an assembly process change in this area of the nozzle.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2 (Continued)	STS-39 LH nozzle throat assembly bond line void.	The resulting thermal and structural integrity of the nozzle throat assembly with a bond line void during motor firing is the concern with this issue. Thiokol performed a structural analysis of the STS-39 LH nozzle assembly with the void and determined the resulting FOS to be 12.5; well above the 2.0 structural FOS requirement. While this issue was not a concern, a complete review of the STS-39 and STS-40 build and inspection documentation was performed; no anomalies were reported. A thermal and structural assessment was undertaken to determine the result of hot-gas flow into the void area. The assessment found a large structural margin of safety. Worst-case thermal analysis resulted in acceptable steel temperature while maintaining O-ring sealing capability. Worst-case predicted O-ring erosion was 61 mils for an unbonded void of 8.5 inch axial x 50 inch circumferential. Rationale for STS-40 flight was: • The throat assembly bond line void was a first-time occurrence. • Preliminary structural analysis indicated a resulting FOS of 12.5 with a bond line void similar to that seen on STS-35.

RESOLVED STS-40 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2 (Continued)	STS-39 LH nozzle throat assembly bond line void.	• Bond line voids, extending the full length of the bond, would be obvious and would not escape post-assembly inspection. • Worst-case thermal analysis, assuming hot gases reaching into the void area, predicted acceptable steel temperatures and positive O-ring sealing capability. ***This risk factor was resolved for STS-40.***

SECTION 5

STS-39 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-39 mission, the previous Space Shuttle flight. The IFA list included in this section is not all inclusive. Only those STS-39/OV-103 IFAs which represented a potential safety concern are addressed. Each anomaly is briefly described, and risk acceptance information and rationale are provided. STS-39 IFAs still considered as unresolved safety risk factors for STS-40 (if any) are addressed in Section 3.

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-39 INFLIGHT ANOMALIES

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Abort Region Determinator (ARD) indicated different abort boundary times than what was expected. IFA No. STS-39-I-01 ***No ARD problems were reported on STS-40.***	Postflight data evaluation determined that STS-39 achieved Main Engine Cutoff (MECO) 1.1 seconds (sec) later than predicted. ARD calls for uphill aborts were also later than predicted, and the 3-g throttle started 5 sec later than predicted. The real-time estimate was that the Space Shuttle Vehicle (SSV) was approximately 4,000 pounds (lb) heavier than calculated prior to flight. This early estimate was based on typical propellant flow (approximately 3,600 lb/sec x 1.1 sec = 3,960 lb) and a 1-to-1 relationship between excess propellant and inert weight. Reconstruction of the trajectory determined that there was a delta Space Shuttle Main Engine (SSME) specific impulse (isp) of +0.86 sec, a delta SSME thrust of -2,050 lb, and a delta inert weight of -500 lb. Analysis results demonstrated that the combination of the higher-than-predicted isp and lower-than-predicted thrust will force the ARD to be overly conservative. This was considered preferable to having the ARD be non-conservative. The investigation into this anomaly found that the reconstructed trajectory matched the ARD predictions for STS-39. The higher-than-predicted isp and lower-than-predicted thrust were the result of conservative ARD predictions based on STS-39 engine test performance data. The low isp experienced on STS-39 was within the program experience base; since STS-2, the isp range has been ±0.9 sec. ***This risk factor was acceptable for STS-40.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Flash Evaporator System (FES) feedline "A", system #2, heater failure. IFA No. STS-39-V-01 HR No. INTG-164 {C} ORBI-276B {C} ***No FES problems were reported on STS-40.***	Prior to entering the tanking phase of the second launch countdown, FES feedline "A", system #2, heater failed. Review of data traces indicated that the cause of the heater failure was a short-to-ground, evidenced by a 20-24 ampere (amp) current spike for 5-10 milliseconds (ms) on main bus "B" prior to removal of heater power by a 10-amp line fuse. Although the FES feedline heaters are Criticality (Crit) 2R3, there was concern that the short could be in a cable bundle that also included Crit 1/1 functions. There was also concern for arc-tracking of the Kapton insulation, leading to a potential fire. It was believed that a technician inadvertently stepped on the feedline heater wire harness during repair of the secondary seal cavity pressure sensor, and this caused the first launch attempt to be scrubbed. The prelaunch investigation into this problem determined that the signature of the current spike, being short in duration, did not show signs of arc-tracking. The investigation prior to launch also included identifying all functions routed through the cable bundle in question and testing the functionality of all critical command paths. Nearly all critical command paths were verified; however, the commands for Solid Rocket Booster (SRB) holddown post release systems, that were routed through the suspect cable bundle, could not be verified until the actual command was generated at T–0. Redundant command paths for the SRB holddown post release were available. The decision was made to accept the risk of this condition and proceed with the launch of STS-39. Initial troubleshooting identified the potential for this anomaly to have originated in the Aft Load Controller Assembly (ALCA) #2. Removal and replacement of ALCA #2 did not totally alleviate the problem. Megger checks isolated a short

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1 (Continued)	FES feedline "A", system #2, heater failure.	circuit in the wire harness between ALCA #2 and FES #2. This wire harness was removed and replaced, and retest of FES #2 heaters was successfully completed prior to the next OV-103 mission. Rationale for STS-40 flight was: • Functional testing of the FES feedline heater circuits was performed prior to launch. • Functional testing of all Crit 1 command paths was performed prior to launch. ***This anomaly/risk factor was resolved for STS-40.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2	Auxiliary Power Unit (APU) #2 Fuel Pump(FP)/Gas Generator Valve Module (GGVM) coolant system "A" valve did not open. IFA No. STS-39-V-02 HR No. ORBI-265A {AR} ***No APU FP/GGVM coolant system problems were reported on STS-40.***	After on-orbit APU shutdown, FP/GGVM coolant systems are automatically activated to cool the FP/GGVM. On STS-39/OV-103, APU #2 FP/GGVM cooling system "A" failed to initiate cooling. Cooling system "B" was successfully activated to perform this cooling function. It was determined that the cooling system "A" spray valve LV25 had failed closed. FP/GGVM cooling is needed in the contingency that an abort from orbit is required soon after APU shutdown. Without additional cooling, the FP/GGVM takes approximately 6 hours (hr) to sufficiently cool beyond the point of potential hydrazine detonation, and subsequent fire or explosion in the APU, possibly causing loss of the vehicle and crew. APU FP/GGVM cooling spray valves are life-limited to 9 months due to susceptibility to nickel-hydroxide contamination. Valve LV25 on cooling system "A" had exceeded this life-limit by 30 days at the time of the STS-39 launch. A time/life exception (EV 2123R1) was approved prior to launch to allow LV25 to fly on STS-39. This valve was removed and replaced prior to the STS-48 flight, the next OV-103 mission.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2 (Continued)	APU #2 FP/GGVM coolant system "A" valve did not open.	Rationale for STS-40 flight was: • LV25 had exceeded the life-limit criteria for APU FP/GGVM cooling spray valves prior to launch. All cooling spray valves installed on STS-40/OV-102 were within the 9-month life limit restriction. ***This anomaly/risk factor was resolved for STS-40.***
3	Reaction Control System (RCS) vernier thruster F5R fuel injector temperature biased low. IFA No. STS-39-V-03 HR No. ORBI-056 {C} ***Vernier jet L5L failed off due to low chamber pressure on first use on STS-40. L5L was hot-fired on orbit, reselected, and used for the remainder of the STS-40 mission with erratic chamber pressure.***	During firing of RCS vernier thruster F5R, the fuel injector temperature read 30°F to 40°F lower than the oxidizer injector temperature. Both the oxidizer and fuel injector temperatures should not vary more than 10°F. Because there were no apparent thruster heater failures or leaks detected, it is believed that this anomaly was caused by a sensor or instrumentation error. Loss of a vernier thruster is a Crit 2/2 failure mode, loss of mission capability. There have been 2 previous vernier thruster oxidizer/fuel injector temperature anomalies similar to this occurrence; 1 on STS-3 and 1 on STS-4. Both anomalies were determined to be the result of poor sensor thermal conductivity that occurs in the vacuum of space. The corrective action taken to overcome thermal conductivity problems was to add thermal grease to the exterior of the sensor probe and the sensor injector well. It was subsequently determined that the sensor in F5R was the old configuration and did not have the added thermal grease.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)	RCS vernier thruster F5R fuel injector temperature biased low.	Postflight troubleshooting determined that this anomaly, as in the 2 previous cases, was caused by poor contact between the sensor and the thruster. There was not a true temperature degradation in F5B on STS-39. Rationale for STS-40 flight was: • Loss of a vernier thruster is a Crit 2/2 failure. ***This anomaly/risk factor was acceptable for STS-40.***

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4	Operations (OPS) recorder #2 uncommanded configuration before launch [potential Multiplexer-Demultiplexer (MDM) hybrid circuit failure]. IFA No. STS-39-V-04 HR No. ORBI-038A {AR} ***On Flight Day (FD) 4, OPS recorder #2 experienced what was thought to be an uncommanded mode and track change similar to this anomaly. Troubleshooting found that the occurrence on STS-40 was due to an erroneous uplink that caused the mode and track change.***	During the second STS-39 launch attempt, OPS recorder #2 experienced uncommanded operation; the recorder was discovered "on" and had changed tracks. Data review indicated that OPS recorder #2 operated in the following sequence prior to being found "on" and commanded "off": approximately 4 seconds (sec) "on", 1 sec "off", and 3 minutes (min) 20 sec "on". Preliminary engineering analysis prior to launch indicated that the anomaly was in the recorder. Testing of OPS recorder #2 was prescribed and performed prior to launch, with no further anomalies identified. OPS recorder #2 was cleared for launch and operated nominally through most of the mission. On FD 7, OPS recorder #2 was witnessed to repeat the prelaunch anomaly. While "on", OPS recorder #2 changed tracks, speed, and mode without the required commands. OPS recorder #2 was subsequently reconfigured from the ground, and it operated nominally for the remainder of the mission. Investigation into the prelaunch OPS recorder #2 anomaly continued throughout the mission. Several different scenarios were identified that could recreate the prelaunch anomaly. Consideration of these scenarios led to the preliminary determination that MDM Payload Forward 2 (PF2), Serial Number (S/N) 76, the MDM between the General Purpose Computers (GPCs) and OPS recorder #2, could be the cause of the anomaly. It was believed that PF2 generated erroneous output to OPS recorder #2 causing the track, speed, and mode change experienced prior to and during the STS-39 mission. The potential for erroneous MDM output, if generic, was a concern in the cases where MDMs are used in Crit 1 proximity/rendezvous operations. MDM PF2 performs only Crit 3/3 functions.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4 (Continued)	OPS recorder #2 uncommanded configuration before launch (potential MDM hybrid circuit failure).	Review of MDM failure history indicated 2 failures in MDM hybrid circuits. One of these 2 failures was in 1985 and was with MDM PF2, S/N 72, the unit on STS-39/OV-103. The 1985 anomaly had characteristics similar to the STS-39 OPS recorder #2 incident. At that time, the anomaly was attributed to the MDM control hybrid circuit on card 10. The control hybrid circuit was replaced, and MDM S/N 72 was installed on OV-103 in the PF2 position. There were no further anomalies with this MDM until STS-39. Analysis indicated that the shift hybrid circuit may have caused both the 1985 anomaly and the STS-39 anomaly. MDM PF2, S/N 72, was removed from OV-103 at Kennedy Space Center (KSC) and sent to the vendor for further failure analysis. There was no indication of a generic MDM problem. Rationale for STS-40 flight was: • The cause of this anomaly was MDM/PF2, S/N 72. This MDM was removed from OV-103. • There was no generic MDM problem. ***This anomaly/risk factor was resolved for STS-40.***

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Supply water dump nozzle temperature drop. IFA No. STS-39-V-08 ***There were no water dump nozzle problems reported on STS-40.***	Approximately 20 min into supply water dump #5, the water dump nozzle temperature rapidly decreased 30°F, from 163°F to 133°F, over a 14-min period. Nozzle temperatures normally remain around 170°F. After this period, the nozzle temperature recovered to normal. With the supply water dump valve closed prior to a subsequent dump, a rapid 5°F drop in nozzle temperature was observed. Nozzle heaters were "on" when this event occurred. Data review from the last 2 OV-103 flights indicated that the nozzle temperatures rose while heaters were "on" and the supply water dump valve was closed. At no time was the supply water dump function inhibited by the fluctuation in nozzle temperature during STS-39. Failure to melt ice in the nozzle would prevent the use of the primary supply water dump method, a Crit 1R3 condition. There are alternate supply water dump methods available, including routing supply water through the wastewater dump lines. Troubleshooting at KSC was unsuccessful in reproducing this anomaly. This anomaly was considered to be caused by a transient effect.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	Supply water dump nozzle temperature drop.	Rationale for STS-40 flight was: • Supply water dumping was not inhibited by the nozzle temperature anomalies on STS-39. • Supply water dumping is a Criticality 1R3 function. ***This anomaly/risk factor was acceptable for STS-40.***
6	Auxiliary Power Unit (APU) #2 lube oil outlet pressure was low. IFA No. STS-39-V-11 (STS-39-K-01) ***There were no APU lube oil pressure or cooling problems reported on STS-40.***	During entry, APU #2, S/N 301, lube oil outlet pressure was low, reading 25 pounds per square inch absolute (psia); nominal outlet pressure is 40 to 50 psia. Additionally, the minimum delta pressure (ΔP) across the pump was 20 psia during entry; nominal is 25-30 psia. The pump was certified to a ΔP of 25-30 psia and qualified to a minimum ΔP of 23 psia. During the pressure anomalies, APU #2 gearbox bearing temperatures #1 and #2 were within limits at 308°F maximum. Flight Rules require APU shutdown if the lube oil outlet temperature or gearbox bearing temperatures exceed 425°F. Loss of lube oil flow can result in high gearbox temperatures, leading to APU shutdown and loss of critical APU function. This was the first APU lube oil outlet pressure-low anomaly in the program. Previous lube oil anomalies were related to high lube oil outlet pressures. Troubleshooting at KSC determined that APU #2 was not properly serviced prior to STS-39 launch, resulting in a low quantity of lube oil in APU #2. For this

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	APU #2 lube oil outlet pressure was low.	reason, APU #2, S/N 301, was not removed from OV-103 prior to STS-48. This Orbiter IFA was subsequently transferred to a KSC IFA (STS-39-K-01) for procedural corrective action. Rationale for STS-40 flight was: • APU #2 gearbox bearing temperatures were well within Flight Rule limits. • Major lube-oil related problems are Launch Commit Criteria (LCC) screenable prior to launch. • OV-102 APU #1 and APU #3 performed nominally during STS-35; APU #2, recently installed on OV-102, was successfully hot-fired with no anomalies. ***This anomaly/risk factor was resolved for STS-40.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	Right-Hand (RH) outboard Main Landing Gear (MLG) tire excessive wear. IFA No. STS-39-V-12 HR No. ORBI-021 {AR} ***No unusual or excessive tire wear problems were reported on STS-40. STS-40 landed at Edwards Air Force Base, Runway 22.***	Postflight inspection of the MLG tires found that the RH outboard tire showed signs of significant wear. The outer 3 plies were worn excessively; MLG tires have 16 plies. The RH outboard tire did not lose tire pressure as a result of the excessive wear. The data indicated touchdown occurred at 210 knots; nominal touchdown speed is between 185 knots and 210 knots. Orbiter sink rate was nominal at 2 ft/sec. The RH MLG tires contacted the runway approximately 216 ft earlier than the Left-Hand (LH) MLG tires. At initial touchdown, the vehicle centerline was 10 ft to the left of the runway centerline, drifting left at a rate of 3 ft/sec. The Commander initiated a right roll command and applied right rudder to correct this drift prior to nose landing gear touchdown. It is believed that this action resulted in shifting the vehicle weight to the RH outboard tire, contributing to the excessive wear. There was a 12-knot headwind and a 1-knot crosswind at the time of the landing. The roughness of the KSC Shuttle Landing Facility (SLF) runway may also have contributed to the excessive tire wear. Previous experience with tire wear has been limited to localized spin-up spots in the MLG tires; there has been no similar uniform wear to this extent in the history of the Space Shuttle Program. The worst-case spin-up spot tire wear led to the failure of a MLG tire and cessation of the use of KSC as a planned end-of-mission landing site. The SLF is available for all missions as a primary, backup, or contingency landing site. Investigation into this anomaly concluded that the excessive tire wear was the result of environmental and crew performance dispersions. All MLG tires are replaced between flights.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
7 (Continued)	RH outboard MLG tire excessive wear.	Rationale for STS-40 flight was: • The excessive tire wear was limited to 1 of 4 MLG tires, the RH outboard tire. The RH outboard tire did not fail. The wear conditions were well within the tire capability. • The STS-39 landing conditions and SLF roughness were the most likely cause of the tire wear. • The STS-40 end-of-mission landing site was Edwards Air Force Base; the SLF was the backup landing site. • Existing Flight Rules and restrictions are adequate. ***This anomaly/risk factor was acceptable for STS-40.***
8	Loss of communications during entry. IFA No. STS-39-V-13 ***There was no unexpected loss of communications during STS-40 entry.***	Communications were lost during entry; this occurred ahead of schedule and for a longer period of time than predicted and normal. Postflight assessment of this anomaly indicated that the communications dropout was the result of the Orbiter's attitude during the high-inclination entry and the relative position of the Tracking and Data Relay Satellite (TDRS) during that portion of the reentry; there were no hardware or software problems found. Because of these conditions, the S-band

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
8 (Continued)	Loss of communications during entry.	antenna used for 2-way data and voice communications between the Orbiter and the ground was not in the required line-of-sight with the TDRS. Onboard navigation control would have provided sufficient data to the Commander and Pilot to achieve a safe landing if a hardware or software problem had caused the communications loss. Data dropouts were typical during Space Shuttle missions prior to the Tracking and Data Relay Satellite System (TDRSS) availability and use. Mission planning can minimize or preclude data and voice dropouts through attitude control. When attitude control is not available, mission planning can accurately predict the periods of communications dropouts. Rationale for STS-40 flight was: • There were unanticipated losses of communications during STS-40 mission based on the planned flight profile. • Onboard navigation is sufficient to effect a safe landing. ***This anomaly/risk factor was resolved for STS-40.***

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
9	The Pilot's Rotation Hand Controller (RHC) bottomed-out during entry. IFA No. STS-39-V-14 HR No. ORBI-152 {C} ***There were no RHC problems reported on STS-40.***	During entry, the Pilot was in control of the vehicle until approximately 3 min prior to touchdown. At that time, the Pilot turned control of the vehicle to the Commander, who completed the landing. When the Pilot took his hand off the RHC, it dropped into the slot and bottomed-out. The Pilot indicated that he tried to pull it back up and attempted to lock it in place, but was unable to lock it because the adjustment knob was jammed. Postlanding inspection found the adjustment knob in the full counterclockwise, or loose position. The Commander and Pilot RHCs are adjustable in the up and down direction, as well as in the fore and aft direction. The RHC is normally locked into the desired position with 2 adjustment knobs. The lower knob, used for up and down adjustment, is a standard friction-type knob; turning it counterclockwise to loosen, clockwise to tighten. The Commander and Pilot nominally adjust their respective RHC prior to entry, with the 2 knobs, to best fit their relative hand position. The STS-39 Pilot would have been able to use his RHC, if required, even though it would not have been adjusted to the optimum height. Rationale for STS-40 flight was: • The positional adjustment anomaly would not impede the proper and accurate use of the RHC. ***This anomaly/risk factor was acceptable for STS-40.***

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	High-Pressure Oxidizer Turbopump (HPOTP) secondary seal cavity pressure sensor failure on STS-39, engine #2029. IFA No. STS-39-E-01 HR No. ME-C1 (All Phases) {AR} ***There were no SSME anomalies reported on STS-40.***	During the initial stages of STS-39 tanking, engine #2029 posted a Fault Identification (FID) during Purge Sequence Number (PSN)-3. The FID indicated that the HPOTP secondary seal cavity pressure, Channel "A", violated LCC sensor qualification limits [4 pounds per square inch absolute (psia) minimum, 20 psia maximum during PSN-1 through PSN-4] and was disqualified. Channel "A" indicated that the secondary seal cavity pressure had risen to 330 psia, while Channel "B" on the same sensor indicated the pressure to be 16 psia. The secondary seal cavity pressure sensor channels on the other Space Shuttle Main Engine (SSME) HPOTPs also read 16 psia. The disqualification of Channel "A" led to the scrub of the first STS-39 launch attempt. The LCC requires both secondary seal cavity pressure sensor channels to be within qualification limits prior to launch to protect redundancy should 1 channel fail during ascent. If disqualification of Channel "A" had been waived for launch and Channel "B" had failed high during ascent and was disqualified, the result would have been the loss of redline protection for that engine. In the event of loss of redline protection and exceedance of the redline limit, the result would be the catastrophic loss of the engine and the potential loss of the vehicle and crew. The calculated probability of the loss of HPOTP secondary seal cavity pressure redline protection, coupled with exceedance of the redline limit, is 1 in 213,000 launches for any 1 of 3 SSMEs.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1 (Continued)	HPOTP secondary seal cavity pressure sensor failure on STS-39, engine #2029.	The failed pressure sensor and associated wire harness were removed from STS-39/OV-103 and sent to the Marshall Space Flight Center (MSFC) Huntsville Simulation Laboratory (HSL) for failure analysis. The sensor was installed in the HSL in the flight configuration and checked out in ambient conditions; no problems were found. The sensor was chilled using liquid nitrogen, and Channel "A" failed high. Harness connectors on Channel "A" and "B" were reversed at the sensor, and the sensor was again chilled. The same failure signature was identified on Channel "B", the expected result. A new sensor was installed, connected to the harness from STS-39, and chilled with liquid nitrogen. No failure was found, thus isolating the failure mode to the sensor. Following failure analysis at the HSL, the sensor was returned to the vendor for teardown inspection. Resistance and calibration checks performed at the vendor indicated that the fault was downstream of the bridge circuit at ambient temperature. Teardown inspection isolated the fault to the impedance board within the sensor. A fracture in the impedance board was visually evident after removing the Room-Temperature Vulcanizing (RTV). The strain-gage grid network was found to be lifted in the vicinity of the fracture. The straight surface feature of the fracture was indicative of pre-existent damage. Surface "rounding" suggested the presence of the fracture before strain-gage grid etching. The epoxy overcoat was missing in the local area of the fracture.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1 (Continued)	HPOTP secondary seal cavity pressure sensor failure on STS-39, engine #2029.	This was the first impedance board failure in the SSME Program; there was no evidence of a generic design problem. Hot-fire experience of 19 units with impedance boards from the same lot is in excess of 580 starts and 229,313 sec. Nine of these units have experienced a total of 43 flights. Impedance boards in the SSME Program have witnessed over 3,250,000 sec of hot-fire exposure; exposure in flight units is in excess of 750,000 sec. Rationale for STS-40 flight was: • The sensor failure on STS-39 was caused by a pre-existing fracture in the impedance board. Damage was localized to a small region of the failed board. • There have been no other failures in the history of the SSME Program. • Impedance boards from the same lot have experienced 580 engine starts and 229,313 sec of hot-fire operation. Nine boards from these lots have flown a total of 43 flights. • LCC limits are in place to protect against launching with reduced redline protection. ***This anomaly/risk factor was acceptable for STS-40.***

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Right-Hand (RH) Solid Rocket Motor (SRM) nozzle cowl/Outer Boot Ring (OBR) erosion/washing. IFA No. STS-39-M-01 HR No. BN-04 Rev. D {AR} ***Disassembly of the STS-40 nozzle cowl/OBR did not reveal similar erosion or washing.***	Postflight inspection of the STS-39 RH nozzle cowl/OBR Carbon Cloth Phenolic (CCP) identified 11 circumferential areas of erosion/washing and wedgeouts. Erosion/washing and wedgeouts are typical in this region; however, the STS-39 RH cowl showed unusual erratic erosion, ply lifting, and atypical short-ply wedgeouts. The investigation determined that these conditions occurred during motor operation. Based on preliminary visual inspection, the condition of the RH nozzle cowl/OBR was thought to be outside the SRM experience base. However, closer inspection at Thiokol determined that the total heat-affected depth and associated margins of safety comply with the Contractor End-Item (CEI) Specification char and erosion criteria. Measurements indicated that the wedgeouts were of similar size as those seen on previous flight motors for erosion and char depth. Additionally, the calculated margin of safety for the worst-case region was positive (15%). Review of postflight inspection of nozzle cowls/OBRs determined that material-affected depth is not greatly influenced by wedgeout or deeper erosion areas. Postflight assessments did, however, demonstrate that higher erosion correlates with lower char depth. Conversely, areas with less erosion show more char. Statistical assessment of the Redesigned Solid Rocket Motor (RSRM) flight data base concluded that there was a high probability of meeting the 1.5 Factor of Safety (FOS) requirement. For the OBR, a 96.0% chance of meeting or exceeding the 1.5 FOS requirement was calculated. For the cowl, the probability of meeting the 1.5 FOS requirement is 99.2%. It was also determined statistically that the probability of violating a 1.0 FOS is less than 0.01% for both the OBR and cowl.

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

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1 (Continued)	RH SRM nozzle cowl/OBR erosion/washing.	Rationale for STS-40 flight was: • The erosion/washing, wedgeouts, and ply lifting seen on the STS-39 RH nozzle cowl/OBR were within CEI erosion and char criteria. • The calculated margin of safety in the worst STS-39 RH cowl/OBR region was positive. • No apparent trend toward increased wedgeouts, washing, or ply lifting was indicated by experience with flight and test motors. • The use of CCP in the STS-40 aft exit cones from the same lot as that used on STS-39 RH cowl/OBR was not a concern based on the aft exit cone flight and test performance history. ***This anomaly/risk factor was resolved for STS-40.***

SECTION 6

STS-35 INFLIGHT ANOMALIES

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

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
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} ***No BFS software anomalies were experienced on STS-40.***	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.

C-2

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)	BFS software patch for pad B definition of longitudinal location was incorrect.	In the future, to ensure that 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. Rationale for STS-40 flight was: • A similar I-load patch was required for STS-40 because it too was moved from pad A to pad B. Procedures were in place to verify the I-load patch. • Large errors are detectable during I-load software certification testing. *This anomaly/risk factor was resolved for STS-40.*

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-V-04 ***The left RCS drain panel heater "A" performed nominally on STS-40.***	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. Troubleshooting determined that the heater "A" thermostat had failed. There were no reported problems with the left RCS drain panel thermostat on previous OV-102 missions. RCS heaters are Crit 2R3 components. The left RCS drain panel thermostat was replaced.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	Left RCS drain panel heater "A" was not at normal temperature.	Rationale for STS-40 flight was: • The left RCS drain panel heater "A" thermostat was replaced. • RCS drain heaters are redundant. • Loss of both heaters requires a preferred attitude maneuver for temperature control (Flight Rule 6-10B). • If temperature control cannot be maintained with attitude maneuvering, worst- case effects would be mission termination (Flight Rule 6-10B). ***This anomaly/risk factor was resolved for STS-40.***
2	Degradation of wastewater dump function. IFA No. STS-35-V-05 HR No. ORBI-254 {AR} ***Wastewater dump functions were nominal on STS-40.***	Water from the wastewater storage tank is periodically dumped overboard into space during a nominal mission. A gradual degradation of the wastewater dump rate was noted during the first 3 wastewater 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 manifest additional CWCs on all flights. Similar problems on STS-32/OV-102 led to removal and cleaning of the last 22 inches of the dump line. The blockage experienced on STS-35 was upstream of this section.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	Degradation of wastewater dump function.	Troubleshooting of the STS-35 anomaly found that the wastewater 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 3 flights. The filter material, polyurethane, will deteriorate after approximately 8 years. A spare filter assembly obtained from the logistics stockroom was found with similar signs of deterioration. This spare filter, that had never been used (still in the shipping package), was manufactured in 1980. These findings led to the requirement to check the STS-39/OV-103 wastewater dump line filter assembly. Upon removal, deterioration was found. The 3-inch long #3 filter (coarse) had approximately 2/3 of the material missing. When the #2 filter (medium) was touched, it fell apart. The #1 filter (fine) was completely gone, and a small amount of gray powder residue was found in the liquid remaining in the filter assembly housing. The STS-39/OV-103 replacement filter was manufactured in 1988. The STS-37/OV-104 wastewater dump line filter assembly was also inspected. This investigation showed no degradation of any filter elements. The filter elements were from the lot manufactured in 1988. The STS-37/OV-104 wastewater dump line was flushed and tested. A wastewater dump line filter from the 1988 lot was installed on STS-40/OV-102. In addition, the dump valve, the dump nozzle, the cross-tie quick disconnect, and the section of dump line between the valve and nozzle were replaced. The wastewater system was flushed in the forward and reverse directions. Operational Maintenance Requirements and Specifications Document (OMRSD) flow checks were successfully passed.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	Degradation of wastewater dump function.	Rationale for STS-40 flight was: • The wastewater dump line filter was the cause of the STS-35/OV-102 line blockage; the STS-40/OV-102 filter was replaced with one from the 1988 lot. • The STS-40/OV-102 wastewater dump system successfully passed all OMRSD flow checks. • The capacity of the waste storage tank is adequate for a minimum-duration mission without water dump. • Three or more failures are required to cause crew illness. A second CWC was added to the STS-40 manifest. ***This anomaly/risk factor was resolved for STS-40.***

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3	-Z star tracker Serial Number (S/N) 006 failed 2 initial self-tests. IFA No. STS-35-V-10 ***No star tracker anomalies were reported on STS-40.***	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. 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 was determined that this slow response time is a function of warmup time and is a generic deficiency in the self-test circuit. 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. Rationale for STS-40 flight was: • This anomaly demonstrated a generic warmup-related deficiency in the star tracker self-test circuitry. • There are 3 redundant strings; -Z star tracker, -Y star tracker, and the COAS. ***This anomaly was not a safety concern for STS-40.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Payload Bay Door (PLBD) environmental seal debond. IFA No. STS-35-V-16 ***There were no PLBD environmental seal problems reported on STS-40. However, portions of the 1307 bulkhead environmental seal and several thermal blankets were loosened on ascent. See Section 7, Orbiter 2 for further details.***	Postflight inspection of OV-102 found a 24-inch piece of the environmental seal teflon material 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-inch splice segment of the PLBD-to-aft bulkhead environmental seal debonded on STS-41/OV-103. Evaluation of the STS-41 problem determined the cause to be insufficient application of the seal (bad etching and bonding). The investigation into the cause of the STS-35 anomaly determined that the failure was the result of interference with a ground strap, causing the seal to debond. Rationale for STS-40 flight was: • The seal was properly bonded to the PLBD. Interference with the ground strap was corrected. ***This anomaly/risk factor was resolved for STS-40.***

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-V-17 HR No. ORBI-036 {AR} ORBI-121 {AR} ***WSB operations were nominal on STS-40.***	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 overcooled the lube oil. A similar anomaly occurred with WSB #2A on STS-38/OV-104 Preliminary analysis of this anomaly indicated 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. The new closed-loop hot-oil flush was performed during the STS-40/OV-102 turnaround process. The closed-loop flush procedure, developed by the WSB vendor, requires hot-oil circulation through the APU system until the heat exchanger outlet temperatures reach 250°F for a minimum of 1 hr. This procedure assures complete melting and dilution of all residual contaminants in the APU lube oil. Post-hot-oil flush WSB #3 testing indicated proper functionality.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	WSB #3A operation was abnormal during ascent and entry.	Rationale for STS-40 flight was: • The newly developed closed-loop hot-oil flush was performed on APU #3/WSB #3. • OMRSD testing on OV-102/STS-40 indicated proper WSB #3 function after the hot-oil flush. • Redundant systems are available. ***This anomaly/risk factor was resolved STS-40.***
6	Window W-1 has a 0.15-inch diameter chip. IFA No. STS-35-V-18 HR No. ORBI-009 {AR} ***Window W-5 was found with a ding measuring 0.0162 inch deep x 0.0663 inch long x 0.0668 inch wide. No other window problems were noted. Plans are to replace W-5 before the next OV-102 mission.***	Postflight inspection of the OV-102 windows revealed a chip in window W-1, measuring 0.15 inch in diameter and approximately 0.0109 inch deep. A "spider web" type crack formation was radiating from the impact point. During the crew debriefing, 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 was removed and replaced. ***This anomaly/risk factor was acceptable for STS-40.***

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	WSB #2 was subjected to abnormally large quantities of wax. IFA No. STS-35-V-19 HR No. ORBI-121 {AR} ***There were no WSB anomalies reported on STS-40.***	During ascent and entry, 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 required a hot-oil flush during the STS-40/OV-102 turnaround processing. APU #2 was removed and replaced during this process due to life-limit criteria. Upon reinstallation of APU #2 on OV-102, the new closed-loop hot-oil flush was performed. Additionally, APU #2 was hot-fired at the pad. There were no apparent problems with wax in the APU #2 lube oil during the hot-fire. Rationale for STS-40 flight was: • The newly developed closed-loop hot-oil flush was performed on APU #2/ WSB #2. • OMRSD testing on STS-40/OV-102 indicated proper WSB #2 function after the hot oil flush. • APU #2 was hot-fired at the pad with no indications of wax. • Redundant systems are available. ***This anomaly/risk factor was resolved for STS-40.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8	RCS vernier thruster R5D failed "off". IFA No. STS-35-V-20 HR No. ORBI-056 {C} ***RCS vernier thruster L5L failed "off" due to low Chamber Pressure (P_c) during the first commanded firing. L5L was hot-fired on orbit, reselected, and operated for the remainder of the STS-40 mission with erratic P_c***	During orbital maneuvering, RCS vernier thruster R5D exhibited low 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. Rationale for STS-40 flight was: • There are redundant down-firing thrusters in each Orbital Maneuvering System (OMS) pod. • Small helium bubbles trapped in propellant lines can be removed by hot-firing the thrusters. • A deselected vernier thruster due to helium bubbles can be reselected after gas is flushed out of the system during a hot-fire. ***This anomaly/risk factor was acceptable for STS-40.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	Orbiter/External Tank (ET) Liquid Oxygen (LO_2) aft attach/separation hole plugger did not fully extend. IFA No. STS-35-V-21 HR No. ORBI-302A {AR} ***There were no attach/separation hole plugger anomalies on STS-40. There was, however, an umbilical separation guide fitting that detached at ET separation, witnessed by OV-102 LH_2 ET/Orbiter umbilical camera. See Section 7, Orbiter 5 for further details.***	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. 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. Rationale for STS-40 flight was: • The likelihood of debris jamming the ET umbilical door is remote. • Doors may be recycled in flight when closing or latching is obstructed. • The ET separation burn moves the Orbiter away from potentially escaping debris. ***This anomaly/risk factor was acceptable for STS-40.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
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-V-22 HR No. INTG-051B {C} ***There were no reports of bent stop bolts on STS-40.***	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 (not reported as an IFA on STS-38/OV-104). 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, demating operations were not required. Because of the recent reports of bent stop bolts on STS-38 and STS-35 and both flows requiring 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
<u>ORBITER</u>		
10 (Continued)	RH stop bolt was found bent on the STS-35 centering ring of the forward ET attach/separation assembly.	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 to be insufficient to cause the bolt bending experienced on STS-34. Rationale for STS-40 flight was: • The bent stop bolt on STS-35/OV-102 was repaired during the STS-40/OV-102 processing flow. • All stop bolts, even when bent, have performed the intended function. • Stop bolts do not carry flight loads and are nonfunctional after Orbiter/ET mate. • Analysis demonstrated that there are no known flight loads that could cause stop bolt bending. ***This anomaly/risk factor was resolved for STS-40.***

STS-35 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
11	Pilot seat down-limit switch failure. IFA No. STS-35-V-23 HR No. ORBI-340 {AR} ***There were no seat anomalies reported on STS-40.***	During the ingress 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 (IFA No. STS-32-27). The seat did, however, work properly on orbit. The down-limit switch was replaced during the STS-35 turnaround process. Troubleshooting at KSC determined that the limit switch had stuck. Closer inspection found that the switch housing was warped, causing the switch to stick. The switch housing was not replaced after the STS-32 anomaly, only the switch. The switch housing was replaced. Retest of the pilot seat positional switches was successful. Rationale for STS-40 flight was: • The warped switch housing, now believed to be the root cause of the STS-32 and STS-35 IFAs, was replaced. ***This anomaly/risk factor was resolved for STS-40.***

SECTION 7

STS-40 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-40/OV-102 mission. Each anomaly is briefly described.

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-40 INFLIGHT ANOMALIES

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Inertial Measurement Unit (IMU) #2, Serial Number (S/N) 023, failed preflight calibration. IFA No. STS-40-V-01 HR No. ORBI-051 {C}	During the first launch attempt of STS-40/OV-102, IMU #2, S/N 023, indicated shifts in accelerometer output data during repeated preflight calibration runs. The data from the first calibration run indicated that the Y-axis accelerometer experienced a 5.54-sigma excursion before it stabilized. The second calibration run data indicated excursions less than 1 sigma; within the Operational Maintenance Requirements and Specifications Document (OMRSD) limit. Data from a third calibration run indicated similar excursions as the first. OMRSD requirements state that a recalibration will be performed in the event of an accelerometer measurement between 2 and 5 sigma; over 5 sigma, the IMU is required to be replaced. Launch Commit Criteria (LCC) require all 3 IMUs to be calibrated and functioning properly prior to launch. Because the cause of the 5-sigma excursions could not be determined, IMU #2, S/N 023, was declared failed, and the launch attempt was scrubbed. IMU #2 was removed and replaced prior to the successful launch of STS-40/OV-102. IMU S/N 023 had a good history of operation, with no similar accelerometer problems. Initial failure analysis at the Johnson Space Center (JSC) Inertial Systems Laboratory (ISL) could not repeat the stability problem encountered during prelaunch calibration runs. Troubleshooting and testing continue. There have been 7 previous inflight IMU anomalies recorded to date. There was no indication that this is a generic IMU problem.

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	IMU #2, S/N 023, failed preflight calibration.	IMUs are Criticality 1R2 for erroneous output and Criticality 1R3 for loss of output. During flight, Redundancy Management (RM) monitors the 3 IMUs for erroneous accelerometer data and deselects the accelerometer output when limits are exceeded.
2	1307 bulkhead environmental seal and thermal blanket anomalies. IFA No. STS-40-V-02 HR No. ORBI-305A {C}	Upon opening the Payload Bay Doors (PLBDs) on orbit, a section of the port-side 1307 bulkhead PLBD environmental seal was discovered separated from its retainer and protruding into the Payload Bay (PLB). Three thermal blankets on the 1307 bulkhead were also found partially unfastened, but not completely loose. These anomalies were indicative of air intrusion into the PLB during ascent. Further inspection of the protruding seal determined that it had separated at a repair splice, located at the Y_o = - 33-inch position on the bulkhead. There were 2 protruding pieces; the inboard piece was approximately 22 inches long, the outboard piece was approximately 8 inches long. The separated environmental seal was constructed from an Inconel wire spring, with a teflon sheath, approximately 0.75 inch in diameter and 64 inches in length. The seal is attached at the centerline and at the Y_o= - 64-inch position. The concern was that the protruding environmental seal could prevent the port PLBD from fully closing. Extensive analysis was undertaken during the STS-40 mission to determine the potential effects of the protruding seal during PLBD closing. This analysis determined that the 8-inch outboard seal protrusion would not be a problem. The 22-inch inboard seal, however, was believed to be likely to interfere with the PLBD latch hook and roller assemblies. This would result in the

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2 (Continued)	1307 bulkhead environmental seal and thermal blanket anomalies.	latch not fully locking. In either case, analysis demonstrated that there was sufficient margin in the PLBD drive motor to overcome friction caused by the seal jamming in the interface between the port PLBD and the 1307 bulkhead. Because of the analysis findings, and to test techniques in the event that an Extravehicular Activity (EVA) contingency was required, a demonstration simulating the protruding environmental seal was performed at the Kennedy Space Center (KSC) on OV-103. An astronaut, wearing EVA gloves, performed several tests which included replacing the seal in its retainer and cutting the protruding portions of the seal. These tests were successful in proving that options existed in the event that a contingency EVA was required. Tests were also made to prove analysis results concerning the potential for the seal to impede PLBD closure and locking. These test cases demonstrated that the 22-inch inboard seal protrusion could migrate into the latch and roller assemblies during PLBD closure. However, in all cases the latch assembly was able to overcome the seal to attain the locked position. Analyses were also performed to determine the thermal and structural effects of the 1307 bulkhead seal not being in the proper configuration. Thermal analysis assumed that 63 inches of the 64-inch seal was missing during reentry. Results of this analysis predicted bulkhead temperatures resulting from air intrusion and aeroheating to be within allowable limits. Venting analysis, assuming that the entire 64-inch seal was missing, demonstrated positive margins for all associated structural elements.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	1307 bulkhead environmental seal and thermal blanket anomalies.	Based on the analyses and the demonstrations on OV-103, the decision was made to close the STS-40/OV-102 PLBDs on the nominal timeline for reentry. The PLBDs were successfully closed and locked for reentry, with no indication that the protruding seal interfered with closure. Postflight quick-look inspection indicated that the environmental seal had separated at the inboard side of the repair splice. There was no indication of heat damage in the area of the separation. Further inspections and removal of the seal will be performed upon removal of SpaceLab from the PLB.
3	Heat erosion of STS-40/OV-102 Right-Hand (RH) External Tank (ET) umbilical door centerline latch. IFA No. STS-40-V-11 HR No. ORBI-302A {AR}	Postflight inspection of STS-40/OV-102 identified significant heat effects (melting) of the RH ET umbilical door centerline latch (forward, outboard). The forward end was found severely discolored and eroded; the erosion had a measured maximum width of 2 inches that was tapered from forward to aft. The measured depth of the erosion across the latch was 0.1 inch. A 0.180-inch step was also observed at the mold line between the latch fitting and flow restrictor fingers; this gap (0.10 inch) should be a butt fit. The Thermal Protection System (TPS) tile near the forward and aft latch fitting was also found with signs of erosion around the edges. Further evaluation of these findings determined that the umbilical door pressure seal was not breached. However, the hot-gas (air) flow path, determined by heat effects on the structure, entered near the forward outboard corner of the RH ET umbilical door, traversed the full length of the door in the cavity between the thermal barrier and the pressure seal, and exited through a small opening at the aft outboard side of the RH ET umbilical door. The environmental pressure seal exhibited localized surface overtemperature due to hot-gas impingement (approximately ±3/4 inch from the initial impingement zone). A 0.025 to 0.030-inch gap was also found between the sill fitting and structure (into the aft

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)	Heat erosion of STS-40/OV-102 RH ET umbilical door centerline latch.	fuselage) on both the RH and Left-Hand (LH) door cavities; the gaps should have been sealed with Room-Temperature Vulcanizing (RTV) by design. The concern associated with this anomaly was the potential for structural damage resulting from excessive thermal effects during reentry and the potential for hot gas to get past the pressure seal into the aft fuselage. In this anomaly, however, the door latch fitting appeared to have worked as a heat sink, minimizing thermal effects and resulting in no evident structural damage or degradation. A similar anomaly occurred on STS-1/OV-102 on the same door latch; it was attributed to step-and-gap problems with the TPS. There are 2 potential causes for this anomaly: improper step-and-gap due to lack of RTV or damage to the tile/latch fitting as a result of a strike by the centerline latch. There was no clear evidence to isolate this anomaly to 1 of these potential causes. Review of the processing records indicated that the RTV was properly applied to the latch fitting. The step-and-gap in this area was also recorded as being correct. The melted latch fitting and surrounding tile will be replaced.

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4	Auxiliary Power Unit (APU) #1 fuel test line temperature above Fault Detection and Annunciation (FDA) alarm limits. IFA No. STS-40-V-12 HR No. ORBI-104A {C}	During reentry, APU #1 fuel test line temperature reached 99°F, above the 95°F FDA limit. Because of previous APU fuel line heater anomalies, the crew was instructed to turn off the APU #1 fuel line heater. A decrease in temperature was witnessed, however, prior to the heater being commanded off by the crew. The redundant temperature sensor indicated a rise in the fuel line temperature, but did not indicate temperatures above the FDA limit. A similar anomaly was experienced on STS-28, APU #1 fuel test line. This is a Criticality 2R3 measurement; however, problems with APU fuel line heaters have increased the sensitivity to exceeding the FDA limit. The FDA limit was increased in early 1990 from 90°F to the current 95°F. Analysis is in work to determine if a further increase is warranted.
5	Cylindrical object debris seen at External Tank (ET) separation. IFA No. STS-40-V-16 HR No. ORBI-302A {AR}	OV-102 is the only Orbiter equipped with cameras in the umbilical cavity. During review of ET separation film, a cylindrical object was seen floating away from the Orbiter LH umbilical cavity. The object was an umbilical guide-pin bushing. Two guide pins are used during the ET/Orbiter umbilical mate process. The Inconel 718 bushings are interference-fit into the 2219 aluminum body of the umbilical. The concern was that debris of this type would impede the ET umbilical door from fully closing. Previous analysis of the potential for escaping debris fragments preventing the ET umbilical door from closing determined that the probability is very small. At ET separation, the Orbiter performs a maneuver moving it away from the ET and escaping debris prior to ET umbilical door closure. This was clearly demonstrated in the STS-40 film of the escaping bushing.

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	Vernier thruster L5L failed off due to low Chamber Pressure (P_c). IFA No. STS-40-V-07 HR No. ORBI-056 {C}	Reaction Control System (RCS) vernier thruster L5L experienced erratic P_c and was declare "failed off" by Redundancy Management (RM). L5L was subsequently hot-fired and recovered to approximately 95% of normal P_c, but still demonstrated slower than normal P_c rise rates. The crew manually reselected L5L, allowing its use for the remainder of the mission. This failure mode is indicative of iron nitrate contamination. Upon return to KSC, L5L was removed from pod LP03 and returned to Marquardt for failure analysis.

SECTION 8

BACKGROUND INFORMATION

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

APPENDIX A

LIST OF ACRONYMS

ΔP	Delta Pressure
A/D	Analog-to-Digital
AEM	Animal Enclosure Module
AFB	Air Force Base
ALCA	Aft Load Controller Assembly
amp	Ampere
AP	Ammonium Perchlorate
APU	Auxiliary Power Unit
AR	Accepted Risk
ARC	Ames Research Center
ARD	Abort Region Determinator
ASA	Aerosurface Amplifier
BFS	Backup Flight System
BITE	Built-In Test Equipment
BOB	Breakout Box
C	Controlled
CA	California
CCP	Carbon Cloth Phenolic
CEI	Contractor End-Item
CIL	Critical Items List
COAS	Crew Optical Alignment Sight
CPU	Central Processing Unit
CR	Change Request
Crit	Criticality
CTI	Charlton Technologies, Inc.
CWC	Contingency Water Container
CWS	Caution and Warning System
DAR	Deviation Approval Request

APPENDIX A

LIST OF ACRONYMS - CONTINUED

EAFB	Edwards Air Force Base
ECLSS	Environmental Control and Life Support System
EDT	Eastern Daylight Time
EIU	Engine Interface Unit
EMI	Electromagnetic Interference
EPR	Ethylene Propylene Rubber
ET	External Tank
EVA	Extravehicular Activity
F	Fahrenheit
FA	Flight Aft
FA-2	Flight Aft-2
FASCOS	Flight Acceleration Safety Cutoff System
FC	Flight Critical
FCL	Freon Coolant Loop
FCV	Flow Control Valve
FD	Flight Day
FDA	Fault Detection and Annunciation
FES	Flash Evaporator System
FID	Fault Identification
FMEA	Failure Modes and Effects Analysis
FMEA/CIL	Failure Modes and Effects Analysis/Critical Items List
FOS	Factor of Safety
FP	Fuel Pump
FPM	Flow Proportional Module
FR	Flight Rule
FRR	Flight Readiness Review
FRSI	Felt Reusable Surface Insulation
FSR	Flight Safety Review
ft	Feet
GAS	Get-Away Special
GGVM	Gas Generator Valve Module
GH_2	Gaseous Hydrogen
GO_2	Gaseous Oxygen
GOX	Gaseous Oxygen
GPC	General Purpose Computer
GSE	Ground Support Equipment

APPENDIX A

LIST OF ACRONYMS - CONTINUED

H_2	Hydrogen
HAC	Heading Alignment Cone
HCF	High-Cycle Fatigue
HDP	Holddown Post
HGDS	Hazardous Gas Detection System
HPFTP	High-Pressure Fuel Turbopump
HPOTP	High-Pressure Oxidizer Turbopump
HPU	Hydraulic Power Unit
HR	Hazard Report
hr	Hour
HSL	Huntsville Simulation Laboratory
I/O	Input/Output
IBR	Inner Boot Ring
ICC	Inter-Computer Channel
ICD	Interface Control Document
IFA	Inflight Anomaly
IMU	Inertial Measurement Unit
in-lb	Inch-Pound
IOM	Input/Output Module
IOP	Input/Output Processor
ISL	Inertial Systems Laboratory
isp	Specific Impulse
JSC	Johnson Space Center
KSC	Kennedy Space Center
kt	Knot
L-2	Launch Minus 2 Day
lb	Pound
lb/hr	Pounds Per Hour
lbf	Pounds-Force
LCC	Launch Commit Criteria
LH	Left-Hand
LH_2	Liquid Hydrogen
LiOH	Lithium Hydroxide
LLCO	Low-Level Cutoff
LO_2	Liquid Oxygen
LPFTP	Low-Pressure Fuel Turbopump

APPENDIX A

LIST OF ACRONYMS - CONTINUED

LPOTP	Low-Pressure Oxidizer Turbopump
LSFR	Launch Site Flow Review
MCC	Main Combustion Chamber
MDM	Multiplexer-Demultiplexer
ME	Main Engine
MEC	Main Engine Controller
	Master Event Controller
MECO	Main Engine Cutoff
MIA	Multiplexer Interface Adapter
min	Minute
MISO	Multiple Inlet, Single Outlet
MLG	Main Landing Gear
MLP	Mobile Launch Platform
MODE-0	Middeck 0-Gravity Dynamics Experiment-0
MPS	Main Propulsion System
MRB	Material Review Board
ms	Millisecond
MSE	Mission Safety Evaluation
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NDE	Nondestructive Evaluation
NLG	Nose Landing Gear
NSI	NASA Standard Initator
NSRS	NASA Safety Reporting System
OBBFX	Orbital Ball Bearing Experiment
OBR	Outer Boot Ring
OEX	Orbiter Experiments
OMI	Operations and Maintenance Instruction
OMRSD	Operational Maintenance Requirements and Specifications Document
OMS	Orbital Maneuvering System
OPS	Operations
OSMQ	Office of Safety and Mission Quality
ORBI	Orbiter
OV	Orbiter Vehicle

APPENDIX A

LIST OF ACRONYMS - CONTINUED

P/N	Part Number
PAR	Prelaunch Assessment Review
PASS	Primary Avionics System Software
P_c	Chamber Pressure
PDT	Pacific Daylight Time
PF2	Payload Forward 2
PLB	Payload Bay
PLBD	Payload Bay Door
PMS	Physiological Monitoring System
POR	Power-On Reset
ppm	Parts Per Million
PRCB	Program Requirements Control Board
PROM	Programmable Read-Only Memory
psi	Pounds Per Square Inch
psia	Pounds Per Square Inch Absolute
psid	Pounds Per Square Inch Differential
PSIG	Propulsion Systems Integration Group
PSN	Purge Sequence Number
QD	Quick Disconnect
RCS	Reaction Control System
RFCA	Radiator Flow Control Assembly
RGA	Rate Gyro Assembly
RH	Right-Hand
RHC	Rotation Hand Controller
RI	Rockwell International
RM	Redundancy Management
RSRM	Redesigned Solid Rocket Motor
RTLS	Return to Launch Site
RTV	Room-Temperature Vulcanizing
S/N	Serial Number
scch	Standard Cubic Centimeters Per Hour
sccm	Standard Cubic Centimeters Per Minute
sccs	Standard Cubic Centimeters Per Second
scfm	Standard Cubic Feet Per Minute
SCU	Sequence Control Unit
sec	Second
SEM	Scanning Electron Microscope

LIST OF ACRONYMS - CONTINUED

SLF	Shuttle Landing Facility
SLS	Spacelab Life Sciences
SMS	Space Motion Sickness
SODB	Shuttle Operational Data Book
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSIP	Shuttle Software Interface Processing
SSME	Space Shuttle Main Engine
SSRP	System Safety Review Panel
SSV	Space Shuttle Vehicle
TAEM	Terminal Area Energy Management
TAL	Transatlantic Abort Landing
TC	Thiokol Corporation
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TEM	Test and Evaluation Motor
TPS	Thermal Protection System
TVC	Thrust Vector Control
UMS	Urine Monitoring System
WECCO	Western Electro Chemical Corporation
WSB	Water Spray Boiler