MISSION SAFETY EVALUATION REPORT FOR STS-48

Postflight Edition

Safety Division

Office of Safety and Mission Quality

National Aeronautics and Space Administration

Washington, DC 20546

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EXECUTIVE SUMMARY

Space Shuttle *Discovery* was launched into a 57-degree inclination orbit from the Kennedy Space Center (KSC) Launch Complex 39A at 7:11 p.m. Eastern Daylight Time (EDT) on September 12, 1991. STS-48 was the second mission since return-to-flight to have KSC as the planned end-of-mission landing site, and the first mission to have a planned night landing at KSC. However, due to weather conditions at KSC in Florida, *Discovery* flew one extra orbit and landed at Edwards Air Force Base, California, at 3:38 a.m. EDT on September 18, 1991.

Operation of all systems was generally satisfactory during the 5-day mission. On Flight Day 3, the Upper Atmosphere Research Satellite (UARS) was deployed from *Discovery's* payload bay 350 statute miles above Earth. This orbiting observatory will study mankind's effects on the planet's atmosphere and its shielding ozone layer. The UARS mission objectives are to provide an increased understanding of the energy input into the upper atmosphere, global photochemistry of the upper atmosphere, dynamics of the upper atmosphere, the coupling among these processes, and the coupling between the upper and lower atmosphere. This will provide important scientific data on the Earth's middle atmosphere – that slice of air between 10 and 60 miles above the Earth. The UARS will have two opportunities to study winters in the northern hemisphere and one opportunity to study the Antarctic ozone hole during the satellite's 20-month life. The UARS is the first major flight element of NASA's Mission to Planet Earth, a multi-year global research program.

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-48 flight were also documented prior to the STS-48 Flight Readiness Review (FRR) (FRR Edition) and the STS-48 Launch Minus Two-Day (L-2) Review (L-2 Edition). This final Postflight Edition evaluates performance against safety risk factors identified in the previous MSE editions for this mission.

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

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SECTION 1

INTRODUCTION

1.1 Purpose

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

1.2 Scope

This report addresses STS-48 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-48 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-48 launch, that were impacted or repeated by anomalies reported for the STS-48 flight.

Section 4 - Contains a list of safety risk factors that were considered resolved for STS-48.

Section 5 - Contains a list of Inflight Anomalies (IFAs) that developed during the STS-43 mission, the previous Space Shuttle flight.

Section 6 - Contains a list of IFAs that developed during the STS-39 mission, the previous flight of the Orbiter Vehicle (OV-104).

Section 7 - Contains a list of IFAs that developed during the STS-48 mission. Those IFAs that are considered to represent a safety risk will be addressed in the MSE for the next Space Shuttle flight.

Section 8 - Contains background and historical data on the issues, problems, concerns, and anomalies addressed in Sections 3 through 7. This section is not normally provided as part of the MSE, but is available upon request. It contains 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-48 MISSION SUMMARY

2.1 Summary Description of the STS-48 Mission

Space Shuttle *Discovery* was launched into a 57-degree inclination orbit from the Kennedy Space Center (KSC) Launch Complex 39A at 6:11 p.m. Eastern Daylight Time (EDT) on September 12, 1991. Launch was delayed for 14 minutes at the T–5 minute mark due to a noise problem in the air-to-ground link. The noise cleared itself, and the countdown proceeded normally to launch.

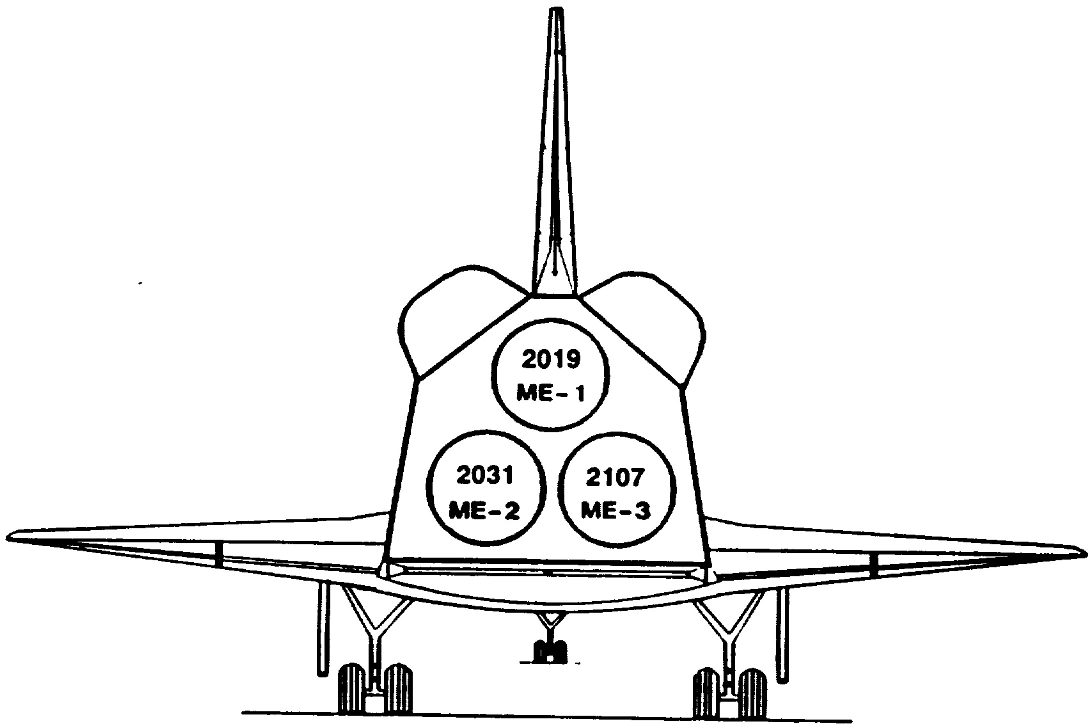
On Flight Day (FD) 3, the Upper Atmosphere Research Satellite (UARS) was deployed from *Discovery's* payload bay 350 statute miles above Earth to study mankind's effect on the planet's atmosphere and its shielding ozone layer. The UARS mission objectives are to provide an increased understanding of the energy input into the upper atmosphere, global photochemistry of the upper atmosphere, dynamics of the upper atmosphere, the coupling among these processes, and the coupling between the upper and lower atmosphere. This will provide data for a coordinated study of the structure, chemistry, energy balance, and physical action of the Earth's middle atmosphere – that slice of air between 10 and 60 miles above the Earth. The UARS will have two opportunities to study winters in the northern hemisphere and one opportunity to study the Antarctic ozone hole during the satellite's 20-month life. The UARS is the first major flight element of NASA's Mission to Planet Earth, a multi-year global research program that will use ground-based, airborne, and space-based instruments to study the Earth as a complete environmental system.

Operation of all systems was generally satisfactory during the 5-day mission. However, several problems were encountered during the STS-48 flight. During External Tank (ET) umbilical door closing, the forward ET door centerline latch motor failed to draw current on phase B. This did not affect ET door closure since each centerline latch has 2 redundant motors and the latch actuators will operate satisfactorily with only 1 good motor (3 good phases). During the second supply water dump on FD 2, the dump nozzle temperature dropped from 150°F to 98°F. The dump nozzle displayed the same type of erratic heater temperature characteristics as seen on STS-39. The dump line was subsequently purged to clear any water and prevent freezing, and was isolated by closing the dump line isolation valve. On FD 5, during the fifth supply water dump, an unusual heating cycle for the dump nozzles was noted. The nozzle heaters were left on following this dump in order to bake out the nozzles. There were indications that ice was possibly forming in the dump line. The most probable source for ice formation is leakage from the dump valve. If the dump nozzle were to be ruled failed, excess water could be dumped into the Flash Evaporator System first, with a second alternative of utilizing the wastewater dump line via the water contingency crosstie (Flight Rule 9-919). On FD4, a 7-second Reaction Control System (RCS) burn was performed to avoid a "spent" upper stage booster that was in the general area of the Orbiter flight path. This collision avoidance maneuver resulted in a small change in *Discovery's* orbit to 308 x 302 nautical mile. On FD 6, the hydraulic system #2 circulation pump was activated by a hydraulic line temperature sensor. The pump circulates hydraulic fluid through the heat exchanger to keep the fluid temperature above the low limit of 0°F. At the completion of the circulation pump cycle, the accumulator exhibited a pressure decay of 155 pounds per square inch per hour (psi/hour). This was attributed to a known contamination from the accumulator resulting in a small leak in the unloader valve. During Flight Control System checkout, the unloader valve seat was flushed of contamination via hydraulic pressure supplied by the Auxiliary Power Unit; the decay rate decreased to 24 psi/hour, which was within the allowable limit of 48 psi/hour.

STS-48 was the second mission since return-to-flight to have KSC as the planned End-Of-Mission (EOM) landing site, and the first mission to have a planned night landing at KSC. However, due to weather conditions at KSC in Florida, *Discovery* flew one extra orbit and landed at Edwards Air Force Base, California, at 3:38 a.m. EDT on September 18, 1991.

2.2 Flight/Vehicle Data

- Launch Date: September 12, 1991
- Launch Time: 6:11 p.m. EDT
- Launch Site: KSC Pad 39A
- RTLS: Kennedy Space Center, Shuttle Landing Facility
- TAL Site: Zaragosa, Spain
- Alternate TAL Site: Moron, Spain; Ben Guerier, Morocco
- Landing Site: Edwards AFB, CA, Runway 22
- Landing Date: September 18, 1991
- Landing Time: 3:38 a.m. EDT
- Mission Duration: 5 Days, 9 Hours, 27 Minutes
- Crew Size: 5
- Inclination: 57.0°
- Altitude: 293 Nautical Miles/Direct Insertion
- Orbiter: OV-103 (13) *Discovery*
- ET-42
- SRBs: BI-046
- RSRM Flight Set #18
- MLP #3



ENGINE	#2019	#2031	#2107
POWERHEAD	#2020	#2028	#2014
MCC*	#2023	#2019	#4002
NOZZLE	#2024	#4017	#4016
CONTROLLER	F4	F29	F25
FASCOS*	#23	#12	#29
HPFTP*	#5203R1	#2323R2	#6003R1
LPFTP*	#2022R1	#2120R1	#4007
HPOTP*	#2424R4	#2226R4	#4305R1
LPOTP*	#2025R1	#2120	#2216

* Acronyms can be found in Appendix A.

2.3 Miscellaneous Items of Interest for the STS-48 Mission.

Planned End-Of-Mission Return to Kennedy Space Center. STS-48/OV-103 was the second mission since return to flight to have KSC as the planned EOM landing site. It was the first mission, however, to have a planned night landing at KSC. STS-43/OV-104, the first mission to have a planned return to KSC, landed safely on August 11, 1991. The extensive deliberation which preceded the decision to land STS-43 at KSC resulted in the determination that for missions with EOM vehicle weights less than or equal to 205,000 pounds (lb), KSC would be designated at the primary EOM landing site; for EOM vehicle weights greater than 205,000 lb, Edwards Air Force Base (EAFB) would be the primary landing site. STS-48/OV-103 EOM vehicle weight was less than 205,000 lb (192,267 lb). In making the decision to return to KSC as a primary landing site, revised landing wind constraints were established:

- For vehicle weight less than or equal to 205,000 lb, crosswind limit is less than or equal to 12 knots.
- For vehicle weight greater than 205,000 lb (when EAFB is unavailable), crosswind limit is less than or equal to 10 knots.
- Peak winds, regardless of direction, must be less than or equal to 20 knots.

The lower landing weight and wind restrictions provide increased tire wear safety margin when landing at KSC.

New Remote Manipulator System (RMS) Failure Mode Identified. A new accepted risk hazard report, RMSX0067026, addressing the RMS was presented to the System Safety Review Panel (SSRP) on August 14, 1991. The hazardous condition was identified as an RMS failure condition that could cause a single-joint runaway of the RMS while parked with the brakes on. This condition would result in damage to the UARS payload, RMS, and/or Orbiter. A mishap could occur as a result of an unannunciated failure (short) of 1 transistor in the direct drive circuitry of the RMS, and a second unannunciated RMS failure (short) of 1 of the 2 transistors in the direct drive brake release/command circuitry. The subsequent failure would cause the brake to be removed and result in a failed joint running away at direct drive rates. Direct drive rates would cause damage to the STS-48 payload/UARS solar arrays. Annunciation of the second failure (audible alarm and caution/warning light) occurs when the brake slip routine detects joint movement greater than 0.05 degrees while the brakes are commanded on; however, the brake slip routine cannot stop the single-joint runaway, and collision of the payload with the RMS or Orbiter is possible. The potential resulting damage to the RMS could render it immobile and prevent stowage of the arm. Jettison of the RMS would become necessary if severely damaged. Damage to the UARS could result in loss of mission. UARS impact with the Orbiter structure may prevent payload bay door closure or cause other catastrophic structural damage.

The primary controls for uncommanded RMS motion are FMEA/CIL 2860, Flight Rule (FR) 12-2, and crew action to remove primary power to the RMS. Crew response is the only control to stop and prevent a collision if this failure were to occur. Nominally under FR 12-2, during crew sleep periods and during extended periods of non-RMS operation, the RMS is cradled, latched, and the system is left in the temperature monitoring mode. During non-sleep periods, the RMS may be left unattended if the conditions of the FR are met. For STS-48, mission-specific flight rules were approved that modified Flight Rule 12-2 and allowed for unattended overnight parking. This approved modification to FR 12-2 would be invoked as a contingency action in the event that UARS deployment was not completed prior to the end of the crew work day.

The crew established that they could respond to an alarm within 15 seconds during the crew awake periods and within 60 seconds for crew asleep periods, and prevent the payload/UARS from colliding with the Orbiter. Clearance analysis showed that if these failures occurred, UARS would not collide with the Orbiter within 60 seconds. The crew was informed of this failure case and the proper response of powering down the RMS.

Elimination of Cadmium Plating on the Solid Rocket Motor Safe and Arm Gasket Retainers. Solid Rocket Motor (SRM) Engineering Change Proposal (ECP) SRM-2418 was implemented on STS-48 for the first time. ECP SRM-2418 eliminates the cadmium plating on the safe and arm gasket retainer. This change was implemented to eliminate the potential for cadmium-induced embrittlement of surrounding hardware. A corrosion preventative grease, HD-2, was applied to the gasket retainer and replaced the cadmium plating. HD-2 is certified for use in other, non-cadmium-plated steel gaskets in the SRM. Eliminating the cadmium plating does not affect the fit or function of the gaskets.

New Manipulator Controller Interface Unit (MCIU) Configuration. The RMS was used on STS-48 to deploy the UARS. A new-configuration MCIU, MCIU-5, was used for the first time with the RMS and replaced the MCIU-3 configuration. Modifications made to the MCIU-3 configuration to achieve the MCIU-5 configuration include: eliminating 14 Criticality 1/1 failure modes, replacing obsolete parts, improving fault detection capabilities, and replacing autosafing with autobrakes (MCIU-3 had both autosafing and autobraking). Two MCIUs were manifested on STS-48: 1 operational and 1 spare. The suggestion was made to manifest an MCIU-3 unit as the spare for STS-48; however, additional crew training and equipment would have been required to support this action.

First Flight of Enhanced Multiplexer-Demultiplexer (EMDM). STS-48 was the first flight to employ an EMDM. Only 1 EMDM was installed in OV-103, in the Operational-Instrumentation Aft #1 (OA-1) position. Multiplexer-Demultiplexers (MDMs) and EMDMs are designed to be functionally equivalent and can be used in a mixed set. Benefits gained in the EMDM over the current MDM include: parts reduction by using gate arrays (MDMs use hybrid circuits for digital logic); incorporation of a power supply for each Input/Output Module (IOM) (MDMs have 1 power supply for all IOMs); enhanced power supply Built-In-Test Equipment (BITE); and a design modification to eliminate an IOM mismapping single-point failure [the MDM has a single-point failure that allows the Sequence Control Unit (SCU) to command multiple IOMs; the EMDM prevents wrong IOM selection by the SCU].

Potential for Tile Slumping Aft of the Nose Cap on STS-48/OV-103. Postflight inspection of OV-103 after STS-39 (the last mission) identified areas of slumped tile aft of the nose cap. This is the area where the other Orbiters have the Reinforced Carbon-Carbon Chin Panel installed in place of tiles. Similar damage was witnessed on STS-26/OV-103. Because STS-48 had a high-inclination mission profile (57°, same as STS-39), similar aeroheating was expected, and the potential for similar tile slumping existed. Plans are to replace tiles in this area with the chin panel during the OV-103 major modification period after STS-42.

New Nose Landing Gear (NLG) Tires. STS-48/OV-103 was the first flight with the new NLG tires, MC194-0007-0006. This new configuration replaces the -0002 NLG tires previously used. Changes incorporated in the -0006 tires include; a commercial tire inner liner compound to provide a lower leak rate; a new filling gum between ply layers to improve layer adhesion; new rubber, instead of scrap, in the sidewall compound for improved ozone resistance; an ozone protective coating replacing the final finish paint; and increased overlap of tread and sidewall rubber to avoid folds during production. Concern was raised over the introduction of these tires into the fleet prior to completion of the -0006 certification program. Certification testing was completed on August 8, 1991 at Wright Patterson Air Force Base, but test results were not yet reviewed by the Orbiter Project and Rockwell International. However, no certification test anomalies were reported. The test tire was pre-rolled prior to testing; NLG tires installed on STS-48/OV-103 were not pre-rolled prior to installation. Analysis determined that pre-roll and shearography are not considered mandatory for flight. Coordination is underway to complete the review of the certification test results and approve the -0006 tires for flight. A Quality Site Approval was effected to enable use of the -0006 NLG tires for STS-48 and STS-44, and to allow orderly completion of the tire certification process.

Potential for First Use of Day-Of-Launch I-Load Update (DOLILU). The DOLILU system is used on the day of launch to determine and verify pitch and yaw I-Loads. The I-Loads are used in the measured day-of-launch environment to enhance Space Shuttle safety, performance, and launch probability. Use of undetected incorrect I-Loads could lead to a catastrophic event. The Space Shuttle Program Requirements Control Board (PRCB) has authorized implementation of DOLILU capability for the purpose of compensating for off-nominal environmental effects. Assessments of I-Loads are independently performed at Johnson Space Center (JSC) and Rockwell/Downey.

A new Integration Hazard Report, INTG-180: Loss of Vehicle Due to Mis-Evaluation of I-Loads and Uplink of Incorrect I-Loads, was processed through the Space Shuttle SSRP on August 13, 1991, and was subsequently reviewed and approved by the Level II PRCB on August 15, 1991 (subject to completion of open work items). INTG-180 is a Controlled hazard, with flight effectivity of STS-48 and subs. Open work closure date was specified as the STS-48 Flight Readiness Review (FRR).

Another feature of the use of DOLILU is the capability for real-time presentation of the DOLILU trajectory on the Range Safety Display System. However, recent testing determined that the Range Safety Display System apparently has a software coding error which does not allow automated import of DOLILU. A corrective action plan was established which indicates this problem will not be corrected until December 1991. For STS-48, the Range Safety Officer was provided with facsimile copies of the DOLILU trajectories in a similar manner as near-real-time wind data is provided. The loss of this feature does not adversely impact the Range Safety Officer in the performance of his required duties.

The Safety and Mission Quality community expressed reservations concerning the use of DOLILU until additional analysis results were available. It was suggested that the standard I-Loads continue to be implemented on STS-48, with DOLILU initially being performed in parallel for comparison and analysis purposes only.

2.4 Payload Data

The highlight of Space Shuttle mission STS-48 was the deployment of the Upper Atmosphere Research Satellite. The UARS is the first major flight element of NASA's Mission to Planet Earth, a multi-year global research program that will use ground-based, airborne, and space-based instruments to study the Earth as a complete environmental system. Mission to Planet Earth is NASA's contribution to the U.S. Global Change Research Program, a multi-agency effort to better understand, analyze, and predict the effects of human activity on the Earth's environment.

Small nuclear sources for the UARS and the Shuttle Activation Monitor (SAM) were reviewed and approved by the Interagency Nuclear Safety Review Panel (INSRP) and appropriately reported to Executive Office of the President, Office of Science and Technology Policy. There was no significant threat to crew health by the presence of these sources.

Payload Bay:

- Upper Atmosphere Research Satellite (UARS) — the observatory consists of a standard design Multi-mission Modular Spacecraft (MMS), coupled to an Instrument Module (IM) that includes 10 instruments/sensors. The MMS Hydrazine Propulsion Module will power orbit adjustment maneuvers for the initial boost to orbit and maintain the required altitude. The system consists of four 5-pound thrusters and 12 small 0.2-pound attitude control thrusters. The Modular Attitude Control System (MACS) is a three-axis system of sensors contained within the MMS that tells the UARS where it is pointed and actuators that can point the spacecraft as required. The entire observatory was designed to permit its retrieval and return to Earth by a Space Shuttle crew if required.

- Ascent Particle Monitor (APM) — collects particulate materials from the Orbiter in the payload bay during the immediate prelaunch period and ascent, using an automated mechanical/electrical assembly.

Middeck:

- Air Force Maui Optical Site (AMOS) Calibration Test — is an electro-optical facility that tracks the Orbiter and records signatures from thruster firings, water dumps, or the phenomena of Shuttle glow (caused by the interaction of atomic oxygen with the spacecraft). The data is used to calibrate the infrared and optical sensors at the facility. No hardware onboard the Shuttle is needed for the system.

- Cosmic Radiation Effects and Activation Monitor (CREAM) — is designed to collect data on cosmic ray energy loss spectra, neutron fluxes, and induced radioactivity. The data will be collected by an active cosmic ray monitor and a passive sodium iodide detector, and up to five passive detector packages, that are placed at specific locations throughout the Orbiter cabin. CREAM data will be obtained from the same locations that will be used to gather data for the Shuttle Activation Monitor experiment in an attempt to correlate data between these two units.

- Investigation into Polymer Membranes Processing (IPMP) — will investigate the physical and chemical processes that occur during the formation of polymer membranes in microgravity to improve the knowledge base that can be applied to commercial membrane processing techniques. The STS-48 mission will provide additional data on the polymer precipitation process.

- Middeck Zero-Gravity Dynamics Experiment (MODE) — will study mechanical and fluid behavior of components for Space Station Freedom and other future spacecraft and is one of the more complex experiments ever to be tested in the Orbiter's middeck cabin area. On orbit, the astronauts command the computer via a keypad to execute test routines stored on the optical recorder before launch. Once a test routine begins, the computer and associated control circuits energize the containers or the truss with precisely controlled forces and then measure the response. The four fluid test articles are Lexan cylinders – two containing silicon oil which has dynamic properties that approximate those of typical spacecraft fluid propellants and two containing water. The structural test article is a truss model of part of a large space structure. Four different truss configurations are slated for testing: (1) the basic truss will be evaluated which is an in-line combination of truss sections, with an erectable module flanked by deployable modules mounted on either end; (2) a rotary joint, similar to the Space Station Freedom "alpha joint" that will govern the orientation of the Station's solar arrays, will replace the erectable section; (3) an L-shaped combination of a deployable truss, rotary joint and erectable module (all mounted in-line), and another deployable section mounted at a 90-degree angle to the end of the erectable truss; and (4) the final configuration that mounts a flexible appendage simulating a solar panel or a solar dynamic module to the elbow of the L-shaped third configuration.

- Physiological and Anatomical Rodent Experiment (PARE) — is the first in a series of experiments designed to determine whether exposure to microgravity results in physiological or anatomical changes in rodents. The PARE-01 experiment examines changes caused by exposure to microgravity in anti-gravity muscles (those used for movement) and in tissues not involved in movement. The objectives of this flight experiment are to determine whether microgravity affects insulin control of glucose transport in an anti-gravity muscle (the soleus); to confirm that in microgravity, non-load-bearing tissues (the heart, liver and adipose tissue) store additional amounts of glycogen as a result of altered regulation of glucose metabolism; and to provide the first data regarding changes in muscle mass and protein content in developing mammals exposed to microgravity.

- Protein Crystal Growth-II (PCG-II) — is a continuing series of experiments to demonstrate the techniques to produce large, high-quality crystals by a vapor-diffusion process at room temperature (22°C). Growth of relatively large and highly-ordered protein crystals reduces the time required to determine protein molecular structures by x-ray diffraction and computer modeling. Sixty different protein crystal growth experiments will be conducted simultaneously.

- Radiation Monitoring Equipment (RME-III) — measures ionizing radiation exposure to the crew within the Orbiter cabin. RME-III measures gamma ray, electron, neutron and proton radiation and calculates, in real time, exposure in RADS-tissue equivalent. RME-III will be activated by the crew as soon as possible after reaching orbit and will be operated throughout the mission.

- Shuttle Activation Monitor (SAM) — is designed to measure gamma ray data within the Orbiter as a function of time and location in the middeck.

- Electronic Still Camera (ESC) — is a hand-held, self-contained digital camera, and is the first model in a planned evolutionary development leading to a family of high-resolution digital-imaging devices. The digital image is stored on removable hard disks or small optical disks, and can be converted to a format suitable for downlink transmission or enhanced using image-processing software. During STS-48, the ESC was used to image areas of interest to commercial remote sensing users. Scenes of Earth, such as major cities and geological formations, will be used to compare the ESC to other Earth-looking sensors. Images of Shuttle crew member tasks in the middeck and payload bay were taken to test the camera's use for documentation and mission support.

2.5 Upper Atmosphere Research Satellite (UARS) Description

NASA's UARS is an Earth-orbiting observatory that will carry out the first systematic, comprehensive study of the stratosphere and furnish important new data on the mesosphere and thermosphere. The UARS mission is the most complex space investigation of the upper atmosphere ever attempted. The UARS will operate 600 km above the Earth in an orbit inclined 57 degrees to the Equator. This orbit will permit the UARS sensors to "see" up to 80 degrees in latitude – thus providing essentially global coverage of the stratosphere and mesosphere – and to make measurements over the full range of local times at all geographic locations every 36 days. UARS will have two opportunities to study winters in the northern hemisphere and one opportunity to study the Antarctic ozone hole during the satellite's planned 20-month life.

The UARS mission objectives are to provide an increased understanding of: energy input into the upper atmosphere, global photochemistry of the upper atmosphere, dynamics of the upper atmosphere, the coupling among these processes, and the coupling between the upper and lower atmosphere. An important product of these studies will be computer models that simulate processes in the upper atmosphere. These simulations will test our understanding of these processes and provide predictions of changes in atmospheric structure and behavior important to future policy formulation.

Three types of measurements will be carried out: (1) composition and temperature, (2) winds, and (3) energy inputs. The data sets from these measurements will yield the first simultaneous, comprehensive, global coverage of these closely-coupled atmospheric properties. A tenth instrument [Active Cavity Radiometer II (ACRIM II) is not technically part of the UARS mission], devoted to measurements of total solar irradiance (the "solar constant"), was also carried to extend a data set of importance to global climate studies.

Atmospheric Composition and Temperature

Four UARS sensors will make global measurements of the vertical distributions of ozone, methane, water vapor, and other minor species involved in the chemistry of the ozone layer. In addition, two of these sensors will derive atmospheric temperature profiles through observations of infrared radiation emitted by carbon dioxide, which is assumed to be well mixed throughout the atmosphere.

- The Cryogenic Limb Array Etalon Spectrometer (CLAES) will determine concentrations of members of the nitrogen and chlorine families, as well as ozone, water vapor, methane, and carbon dioxide, through observations of infrared thermal emissions at wavelengths from 3.5 to 12.7 microns. The CLAES utilizes a telescope, a spectrometer, and a linear array of 20 detectors to make simultaneous measurements at 20 altitudes ranging from 10 to 60 km.

- The Improved Stratospheric and Mesospheric Sounder (ISAMS) experiment will measure the concentrations of nitrogen chemical species, as well as ozone, water vapor, methane, and carbon monoxide, through observations in the infrared spectral region from 4.6 to 16.6 microns. The ISAMS, a filter radiometer, observes infrared molecular emissions by means of a movable off-axis reflecting telescope. The telescope can also be commanded to view regions to either side of the UARS observatory, thus providing increased geographic coverage.

- The Microwave Limb Sounder (MLS) will measure emissions of chlorine monoxide, hydrogen peroxide, water vapor, and ozone in the microwave spectral region at frequencies of 63, 183, and 205 GHz (i.e., wavelengths of 4.8, 1.64, and 1.46 mm). The observations of chlorine monoxide are of particular importance, since this gas is a key reactant in the chorine chemical cycle that destroys ozone; microwave measurements are essential for observations of this species.

- The Halogen Occultation Experiment (HALOE) will determine the vertical distributions of hydrofluoric and hydrochloric acids as well as those of methane, carbon dioxide, ozone, water vapor, and members of the nitrogen family. During every UARS orbit, at times of spacecraft sunrise and sunset, the HALOE will be pointed toward the Sun to measure the absorption of energy along this line of sight. There are 28 solar occultation opportunities per day, providing data for 14 different longitudes in each of the Northern and Southern Hemispheres.

Atmospheric Winds

The UARS mission will provide the first direct, global-scale measurements of the horizontal wind field in the upper atmosphere. The two other UARS instruments, a High Resolution Doppler Imager (HRDI) and a Wind Imaging Interferometer (WINDII), will provide direct observations of wind velocity through measurements of the doppler shifts of selected emission and absorption lines. These doppler shifts will be measured in two different directions, yielding two components of the wind velocity relative to the spacecraft; the true wind velocity can then be calculated.

- At altitudes below about 45 km, the HRDI will observe the doppler shifts of spectral lines within the atmospheric band system of molecular oxygen to determine the wind field. The oxygen bands contain many lines that appear as deep absorption features in the brilliant spectrum of scattered sunlight. The HRDI will exploit these daytime absorption features to provide wind data for the stratosphere and upper troposphere to an accuracy of 5 m/sec or better. At altitudes above about 60 km, the HRDI will observe emission lines of neutral and ionized atomic oxygen in the visible and near-infrared spectral regions. The emission lines are observable both day and night. These measurements will furnish the wind field in the mesosphere and thermosphere to an accuracy of 15 m/sec or better.

- The WINDII utilizes emission lines for the basic doppler-shift measurements. In addition to lines of neutral and ionized atomic oxygen, these include two lines of the OH molecule and a molecular-oxygen line. The WINDII will obtain measurements both day and night at altitudes above 80 km.

Energy Inputs

The upper atmosphere receives energy from the Sun via two sources: ultraviolet radiation and magnetospheric charged particles. The UARS observation program will provide the measurements necessary for determining the net effect of the solar energy inputs on the amount and distribution of ozone in the stratosphere.

- The Solar Ultraviolet Spectral Irradiance Monitor (SUSIM), mounted on the UARS solar/stellar positioning platform, will measure solar ultraviolet radiation in the wavelength range from 120 to 400 nanometers (nm) with a resolution down to 0.1 nm. The instrument is designed to provide its own long-term, absolute calibration light sources to track any change in instrument response during spaceflight.

- The Solar/Stellar Irradiance Comparison Experiment (SOLSTICE) will measure solar ultraviolet radiation in the wavelength range from 115 to 430 nm with a resolution of 0.12 nm. This instrument has the unique ability to compare the solar ultraviolet output with the ultraviolet radiation of stable bright blue stars, using the same optics. The SOLSTICE will be pointed toward the Sun during the daylight portion of each orbit, and toward one of the calibration stars during most of the nighttime portion of the orbit.

- The Particle Environment Monitor (PEM) instrument will determine the type, amount, energy, and distribution of charged particles injected into the Earth's thermosphere, mesosphere, and stratosphere. The PEM will utilize three separate boom-mounted sensors to measure electrons with energies from 1 eV to 5 MeV, protons with energies from 1 eV to 150 MeV, and the strength of the Earth's magnetic field – all in the vicinity of the spacecraft. The PEM will include a 16-element array of x-ray detectors to provide wide spatial coverage of the energy injected into the upper atmosphere by high-energy electrons.

SECTION 3

SAFETY RISK FACTORS/ISSUES IMPACTED BY STS-48 ANOMALIES

This section lists safety risk factors/issues, considered resolved (or not a safety concern) for STS-48 prior to launch (see Sections 4, 5, and 6), that were repeated or related to anomalies that occurred during the STS-48 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 6: STS-39 Inflight Anomalies		
Orbiter 5	Supply water dump nozzle temperature drop.	On the STS-39 flight, approximately 20 minutes (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 (IFA No. STS-39-V-08). 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. Review of data 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 fluctuations in nozzle temperature during STS-39. Postflight troubleshooting at Kennedy Space Center (KSC) was unsuccessful in reproducing this anomaly; it was considered to be caused by a transient effect. A bi-level discrete was added to the nozzle telemetry to monitor power to the nozzle heaters on subsequent flights. On STS-48, during supply water dump #2, the power discrete remained "on" while the nozzle temperatures were erratic, an indication of a problem in the nozzle heater circuitry. During the 2 post-bakeout periods following supply dumps #4 and #5, the supply water line and nozzle temperatures indicated water leakage past the dump valve (IFA No. STS-48-V-04). The crew performed a

ITEM		COMMENT
Section 6: STS-39 Inflight Anomalies		
Orbiter 5 (Continued)	Supply water dump nozzle temperature drop.	purge procedure on the line to clear any water and prevent freezing of the line, dump valve, and dump nozzle. After successful purge of any free water, the line was isolated by closing the dump line isolation valve. KSC performed a post-STS-48 walkdown inspection and leak check prior to removal and replacement of the dump valve, line, and nozzle. The removed units were sent to JSC for analysis. Failure to melt ice in the dump nozzle would prevent the use of the primary supply water dump method (a Crit 1R3 condition). However, excess supply water could be dumped into the Flash Evaporator System or the wastewater dump line could be utilized via the waste water contingency crosstie (Flight Rule 9-919).

SECTION 4

RESOLVED STS-48 SAFETY RISK FACTORS

This section contains a summary of the safety risk factors that were considered resolved for STS-48. 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 Reports (HRs) 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.

SECTION 4 INDEX

RESOLVED STS-48 SAFETY RISK FACTORS

SECTION 4 INDEX

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	External Tank (ET) 17-inch Liquid Hydrogen (LH_2) disconnect, Serial Number (S/N) 1217, failed LH_2 screen test. HR No. INTG-006A {AR} INTG-071 {AR} ORBI-306 {AR} ***No leakage problems were encountered with the ET 17-inch LH_2 disconnect on STS-48.***	The LH_2 leaks experienced on STS-35 and STS-38 in 1990 led to the establishment of a test program for all ET 17-inch LH_2 disconnects that had questionable Acceptance Test Program (ATP) results. Disconnects selected for the LH_2 screening program included: all 2000-series units that were ATP tested with a blanking plate, a sampling of 4 units (6000 series) that were successfully ATP tested with the simulator, and S/N 1217 that was found to have incomplete ATP data records. The remaining disconnects, primarily 6000-series units, were considered acceptable for flight based on original Gaseous Nitrogen (GN_2) ATP test results. S/N 1217 was tested in early 1991 with GN_2, and leakage was recorded in the range of 10-12 standard cubic inches per minute (scim), a low leak rate. During LH_2 screen testing in July 1991, S/N 1217 was found to have a 1,350-scim leak. Post-test troubleshooting determined the cause of the leak to be teflon flaking of shaft seals. Teflon flaking of disconnect shaft seals was determined to be a contributor to LH_2 leaks experienced on STS-35 and STS-38. The high leak rate experienced during the S/N 1217 LH_2 test, coupled with the low S/N 1217 GN_2 test results, has raised concerns about the validity of original GN_2

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1 (Continued)	ET 17-inch LH_2 disconnect, S/N 1217, failed LH_2 screen test.	ATP test data and the acceptability of units in the field that have not been tested with LH_2. The units in question are:

S/N	GN_2-ATP LEAKAGE	LOCATION
6811	11 scim	KSC/STS-48/ET-42
2030	0 scim	MAF*/STS-50/ET-50
6802	33 scim	KSC/STS-44/ET-52
6801	22 scim	KSC/STS-42/ET-53
6804	26 scim	MAF/STS-51/ET-54
6803	27 scim	MAF/STS-58/ET-TBD
6810	14 scim	MAF/STS-62/ET-TBD

*MAF - Michoud Assembly Facility

The Orbiter Project has established a plan for all field units, except S/N 6811, to be screen tested with LH_2 to minimize the potential risk associated with not identifying a leaking 17-inch disconnect until initiation of tanking on launch day. This approach represents the minimum impact on the current launch processing schedule. S/N 6811 was excluded from consideration because ET-42 had already been mated to STS-48/OV-103 when the concern was raised. The Space Shuttle Program, with the concurrence of the Safety, Reliability and Quality Assurance (SR&QA) community, has accepted the residual risk associated with proceeding with the ET-42/STS-48 tanking and the potential for identifying a Hydrogen (H_2)

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1 (Continued)	ET 17-inch LH_2 disconnect, S/N 1217, failed LH_2 screen test.	leak during tanking. Launch Commit Criteria (LCC) are in place to preclude a launch with an excessive H_2 leak at the 17-inch disconnect. The applicable LCC is as follows: • No presence of unusual vapors and liquid droplets. The term "unusual vapors and liquid droplets" is defined as: – An obvious blowing leak or vapor cloud which obscures the disconnect or feedline region for an extended period (> 5 minutes). – Consistent, frequent liquid drops falling or flowing with identifiable vapor trails. • No H_2 concentration greater than 40,000 parts per million (ppm) (4%) on both 17-inch disconnect sensors [Leak Detector (LD) 54 and LD 55]. If a sensor has been declared failed, the remaining sensor must not exceed 40,000 ppm (4%). • No H_2 concentration greater than 20,000 ppm (2%) on 1 of 2 17-inch disconnect sensors (LD 54 or LD 55) without evaluation of available data by the Mission Management Team (MMT) and MMT approval to continue the launch countdown. • If intermittent or erratic readings occur, the data would be evaluated over a 10-minute period to determine the actual H_2 concentration.

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)	ET 17-inch LH_2 disconnect, S/N 1217, failed LH_2 screen test.	Rationale for STS-48 flight was: • The risk of an H_2 leak in the S/N 6811 unit was low. • If a leak was experienced during tanking, LCC were in place to preclude a launch with an excessive H_2 leak. ***This risk factor was resolved for STS-48.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Main Propulsion System (MPS) cryogenic temperature transducer failure. HR No. INTG-023 {AR} ME-C2 (All Phases) {C} ME-D2 (All Phases) {C} ***No MPS cryogenic temperature transducer anomalies were reported on STS-48.***	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, Serial Number (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. The S/N 105 -0021 transducer was removed and sent to the vendor, RDF Corporation of New Hampshire, for failure analysis. The results of the RDF Corporation failure analysis indicated that there were 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 were no filters or screens to prevent a piece of the 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 could 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

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	MPS cryogenic temperature transducer failure.	transducers. All spare temperature transducers available at the Kennedy Space Center (KSC) were also acquired 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 found with either similar circumferential cracks in the sheath-to-mandrel weld or with indications of cracking. There has also been a -0021 transducer from OV-105, with no flight experience, found with a sheath-to-mandrel weld crack. Evaluation of this unit is underway. To date, 7 of 18 -0021 MPS temperature transducers have been found with weld cracks. In July 1991, a -0018 MPS temperature transducer removed from the Main Propulsion Test Article (MPTA) was found with similar sheath-to-mandrel circumferential cracks. This -0018 transducer was reported to have experienced 25 tankings and 12 engine firings prior to discovery of the crack. However, records on MPTA components are not as strictly maintained as flight components, and this -0018 transducer may have experienced significantly more operating time. No other ME449-0013 dash-number temperature transducers x-rayed to date were found with cracks. Except for S/N 105, there were 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 investigation into this issue led to Space Shuttle 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) would be required to allow for removal of the LH_2

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	MPS cryogenic temperature transducer failure.	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. However, a review determined that it would not be feasible to remove the LO_2-side temperature transducers. LO_2 SSME inlet temperature transducers are used as the sole source for protecting the engine start box requirements for LO_2 temperatures. Detailed analysis identified no alternative approaches to protect the engine start box. To date, there have been no LO_2 temperature transducers found with sheath-to-mandrel cracks. For STS-48/OV-103, 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. The -0018 transducer used for STS-48/OV-103 had not flown and had not experienced a tanking or engine firing. A change to the Launch Commit Criteria (LCC) relating to MPS transducers was also approved to reduce the risk of launching with a structurally-failed transducer. The previous LCC required 2 of 3 engine inlet transducers to be operational prior to launch. The revised MPS transducer LCC screens 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.

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1 (Continued)	MPS cryogenic temperature transducer failure.	Rationale for STS-48 flight was: • There were no LO_2 temperature transducers found with similar cracks. • All MPS temperature transducers installed on OV-103, including the -0018 transducer in the LH_2 manifold, successfully passed x-ray inspection for cracks and were cleared for flight. • Flight-certified plugs were installed in place of the LH_2 temperature transducers near the LPFTP inlet. • Leak checks on all newly installed plugs and transducers were successfully performed. • The revised MPS transducer LCC does not allow a launch with anomalous transducer indications. ***This risk factor was acceptable for STS-48.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2	Generic power supply problem in the new General Purpose Computers (GPCs), AP-101S. HR No. INTG-021B {AR} INTG-065C {AR} *No GPC power supply anomalies were reported on STS-48.*	GPC Serial Number (S/N) 516 failed during acceptance thermal testing at International Business Machines (IBM) on March 5, 1991. The GPC failed to power-up, and +5-volt Direct Current (DC) Central Processing Unit (CPU) power was not present during attempted start at 35°F after –5°F cold soak. The +5-volt DC power supply is used to supply power to logic devices within the GPC, including the main memory devices. Failure investigation found the failure was caused by shorting of diode-block mounting screws, on the top of the power supply frames, against the GPC top cover. The diode-block mounting screws are electrically "hot". A short to the top cover causes the +5-volt DC power supply to shut down due to an undervoltage detection. The short to the top cover was attributed to the allowable tolerance buildup of the screw head, washer, and bushing exceeding the allowable counterbored hole depth. In addition, it was revealed during the failure investigation that the diode block can be installed offset from center; this can lead to damage to the bushing, causing additional thickness and resulting in the screw head protruding above the power supply frame. The screw head depth was checked on 19 GPCs (10 screws per GPC), including 5 flight units AP-101S, 9 preproduction units, and 5 Spacelab units AP-101SL. The screw heads were found to have from 0.002- to 0.022-inch clearance below the power supply frame. The worst case of 0.002–inch clearance was found on the Spacelab AP-101SL qualification unit that was subjected to full qualification thermal and vibration testing. Thinner washers, tighter bushing tolerances, and a new

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2 (Continued)	Generic power supply problem in the new GPCs, AP-101S.	alignment tool are to be implemented on all GPC power supply units currently at the vendor, and on all field units on an attrition basis. All failures experienced to date were detected during factory testing. There were no incidents of this type experienced in the field on laboratory or flight units; accumulated field operating time was over 50,000 hours on flight units, over 400,000 hours on pre-production units, and over 1,400 hours on flight units recently modified for power-on-reset problems (including the GPCs for this flight). Thermal and vibration testing are valid screens for this failure mechanism. All AP-101S GPCs in the field were subjected to thermal and vibration testing. Two or more GPC failures during ascent, coupled with an Engine Interface Unit (EIU) or Space Shuttle Main Engine (SSME) controller failure that produces more than 1 command path failure to the main engine during the last 30 seconds prior to Main Engine Cutoff (MECO), results in a catastrophic main engine shutdown. GPC functions, except the Backup Flight Computer application, are redundant, and a spare GPC is available on the Orbiter for replacement of a failed unit on orbit. Worst-case loss of GPC output is Crit 1R2.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	Generic power supply problem in the new GPCs, AP-101S.	Rationale for STS-48 flight was: • Failures experienced to date were detected during factory testing. • Thermal and vibration testing are valid screens for this failure; all flight GPCs, including those on STS-48/OV-103, passed these tests. • Loss of GPC output is Crit 1R2 (worst case). • GPCs are redundant, and a spare GPC is available on the Orbiter for replacement of a failed unit on orbit. ***This risk factor was acceptable for STS-48.***

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Avionics software discrepancy provides the potential for an undetected failure of a Space Shuttle Main Engine (SSME) controller channel. HR No. INTG-165 {C} ORBI-066 {AR} ***No SSME controller channel anomalies were experienced on STS-48.***	Discrepancy Report (DR) 101776 was written in 1989 against the avionics software because the General Purpose Computer (GPC) was not reading all SSME controller channel failure flags. Troubleshooting determined that the GPC only looks for an SSME channel failure flag in the time frame between issuance of a command and receipt of the command "received acknowledgement" indication. At the time, DR 101776 was dispositioned as a Crit 3 problem based on the belief that the SSME controller would set a Major Component Failure (MCF) indication for all channel failures, thus providing sufficient indication of a channel failure. Recent SSME Block II controller software testing revealed that when a channel failure is caused by a recoverable power transient (less than 30 milliseconds) an MCF indication is not set. The software design in both the Block I controller currently in use, and in the Block II controller, is identical in this area. The controller software designers had originally made the assumption that the avionics software could determine a channel failure by catching a channel fail flag. DR 101776 was elevated to a Crit 1 software discrepancy prior to STS-43 due to the determination that the SSME controller software does not set an MCF during a channel failure recoverable power transient. The elevated concern with this discrepancy is that, because the GPC avionics software may miss an SSME controller channel failure flag, a subsequent channel failure in the same controller would cause the associated SSME to lock on the last command received and become uncommandable. There are periods between T–11 second (sec) to Solid Rocket Booster (SRB) ignition when occurrence of this discrepancy could result in

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)	Avionics software discrepancy provides the potential for an undetected failure of an SSME controller channel.	a catastrophic event after launch. This is due to the fact that only 3 commands are issued in this time period: "100% throttle" at T–11 sec, "start enable" at T–9.5 sec, and "start" at T–6.6 sec. At SRB ignition, continuous SSME commanding occurs. Because the avionics only recognizes a channel failure flag between command issuance and a receipt acknowledgement, any channel failure flag set after T–6.6 sec will not be detected by the avionics software until after the next SSME controller command is sent following SRB ignition. This results in a launch that is 1 failure away from a potential catastrophic event caused by the loss of avionics software SSME control resulting in the potential for catastrophic engine shutdown. The Launch Commit Criteria (LCC) requires 3 of 3 SSME controller channels in each SSME controller for launch to protect against loss of 2 channels during flight. There are several scenarios that would result in non-nominal conditions in the event of a second SSME controller channel failure after an undetected prelaunch channel failure in the same controller. All but one scenario results in contingency recovery of engine control through crew intervention. As in the case of dual Engine Interface Unit (EIU) Power-On Reset (POR), loss of a second SSME controller channel in the last 30 sec prior to Main Engine Cutoff (MECO) would not provide the crew with sufficient time to manually shut down the associated SSME prior to prevalve closure. In this case, the GPCs would close the prevalve on a running SSME, resulting in a catastrophic shutdown. The risk associated with this issue, while potentially catastrophic, is considered low. Two independent SSME channel failures are required. There has been no record of SSME controller channel failures due to recoverable power transients. The

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3 (Continued)	Avionics software discrepancy provides the potential for an undetected failure of an SSME controller channel.	window of exposure to the first, undetected SSME controller channel failure is small: 6.6 sec. The time of catastrophic exposure due to a second channel failure, as in the case of dual EIU PORs, is considered short: the last 30 sec prior to MECO. A waiver was written for STS-43 to identify the Crit 1 risks associated with DR 101776. For STS-48, the decision was made to extend this waiver and not attempt to incorporate a patch to fix this discrepancy. Rationale for STS-48 flight was: • Catastrophic SSME shutdown requires an undetected channel failure in the last 6.6 sec prior to launch due to a recoverable power transient coupled with a hard channel failure in the last 30 sec prior to MECO. • There have been no SSME controller channel failures caused by recoverable power transients experienced in the Space Shuttle Program. • The waiver written for STS-43 was extended for STS-48. ***This risk factor was acceptable for STS-48.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Contamination found in STS-48/OV-103 Main Propulsion System (MPS). HR No. INTG-023 {AR} ME-A1A Rev. F {AR} ***No engine anomalies attributed to MPS contamination were reported on STS-48.***	Decay checks of OV-103 MPS engine #1 determined that the Liquid Hydrogen (LH_2) feedline exceeded the 0.25 pounds per square inch (psi) per minute pressure rise requirement. Further testing identified that the engine #1 LH_2 relief valve had a leak rate of 3,898 standard cubic inches per minute (scim); the specification limit is 400 scim. The relief valve was removed and replaced, and the new valve operated properly. Engine #1 decay tests were successfully completed. Inspection found that the replaced relief valve would not properly seat due to the presence of contamination. The contamination was a plastic sliver, measuring 0.30 inch x 0.15 inch. Further inspection of the entire OV-103 MPS was directed, based on this finding. Inspection of engine #1 MPS components uncovered the single piece identified above. No contamination was found in the engine #1 prevalve screen; however, the screen is upside down when the vehicle is in the horizontal position, and any similar contamination would have fallen from the screen. A piece of a "clean part" label with black tape was found in the engine #2 prevalve screen along with 2 small pieces of plastic. The engine #3 LH_2 prevalve screen contained several pieces of contamination: 10-15 pieces of plastic film and 1 piece of black tape. Borescope inspection also found a large amount of plastic slivers in the engine #3 recirculation pump package. No similar contamination was found in the engine #1 and engine #2 recirculation pump packages. Inspection of the OV-103 MPS Liquid Oxygen (LO_2) plumbing did not yield similar contamination.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4 (Continued)	Contamination found in STS-48/OV-103 MPS.	The majority of the contamination was removed from the LH_2 system. The particles did, however, remain in the engine #1 prevalve behind the flow liner and in the 17-inch disconnect behind the flow liner. These particles were not of sufficient size or quantity to affect propellant flow to the engines. In addition, if undetected particles exist, the feedline screens will prevent particles ≥1,000 microns from being ingested by the engine. The engines are capable of ingesting particles <400 microns and a very small amount of particles up to 1,000 microns. Analysis of the contamination at a Kennedy Space Center (KSC) laboratory determined the contamination to be polyethylene. This contamination most likely originated from a "clean part" bag left in the MPS, Mobile Launch Platform, or the External Tank prior to STS-39/OV-103. The time period of the introduction of this contamination into the OV-103 MPS is uncertain. Engines #2019, #2031, and #2107 were on STS-37, the last flight of OV-104. These engines were inspected in the KSC engine shop for contamination prior to installation on STS-48/OV-103. There was no indication of anomalous engine operation during STS-37. Rationale for STS-48 flight was: • This was considered an isolated contamination incident relating only to the OV-103 MPS. • There were no engine anomalies reported on the last flight of OV-103 or OV-104.

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4 (Continued)	Contamination found in STS-48/OV-103 MPS.	• Most of the contamination was removed. The particles remaining were not of sufficient size or quantity to affect propellant flow. • The SSMEs are capable of ingesting particles <400 microns and a very small amount of particles <1,000 microns; particles ≥1,000 microns would be trapped in the feedline screens. ***This risk factor was acceptable for STS-48.***
5	Master Event Controller (MEC) preflight Built-In-Test Equipment (BITE) errors. HR No. ORBI-289A {AR} INTG-052C {C} INTG-135B {C} INTG-143A {AR} INTG-164A {C} ***No MEC prelaunch anomalies due to BITE errors were reported on STS-48.***	Two MECs recently failed preflight BITE tests. MEC Serial Number (S/N) 2 failed on STS-48/OV-103 during testing on July 21, 1991. BITE indicated a power supply failure. Troubleshooting determined that the −12 volt power supplied to the Multiplexer Interface Adapter (MIA) logic on channel 3 was out-of-tolerance. The second unit to fail BITE tests, S/N 8 on OV-102 on July 26, 1991, also indicated a power supply problem. In this case, the +5-volt power supplied to the MIA logic on channel 4 appeared to be out-of-tolerance. These recent failures raised the concern for a generic MEC power supply problem. In both cases, however, the MIA circuitry continued to communicate properly with the General Purpose Computers (GPCs), indicating the potential for a failure in the BITE circuitry. There has been 1 previous MEC power supply failure. This occurred during acceptance testing of MEC S/N 12 and was caused by a cracked solder joint. MECs S/N 2 and S/N 8 had no previous power supply problem history. MECs

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	MEC preflight BITE errors.	S/N 2 and S/N 8 were returned to the vendor for failure analysis; the cause of the BITE indications has not yet been determined. MEC S/N 1 was installed in STS-48/OV-103; it did not experience power supply problems. There are 2 MECs on each Orbiter to provide redundancy for time-critical Orbiter/External Tank/Solid Rocket Booster separation functions. Each MEC has 2 cores (A and B) to provide redundancy and verification of critical command voting. All MEC output command paths and functions are verified prior to each flight. End-to-end verification tests of each MEC core function are performed after 5 flights. BITE tests are performed prior to launch to ensure that MEC power supplies, commands, and internal electronics are functioning. Both MECs are required to be functioning properly prior to launch. Rationale for STS-48 flight was: • MECs are redundant; there are 2 cores per MEC. • A power supply or MIA failure would be detected prior to launch by BITE; the launch would be scrubbed if a BITE failure indication is present. ***This risk factor was acceptable for STS-48.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	Inter-Process Variable (IPV) anomaly found in Backup Flight Software (BFS). *No BFS anomalies were reported on STS-48.*	Independent Verification and Validation (IV&V) analysis of software release OI-20 uncovered an anomaly in the BFS. The anomaly, documented in Discrepancy Report (DR) 108655, can occur when alternative landing sites are selected by the crew during a Transoceanic Abort Landing (TAL) contingency, and results in the BFS General Purpose Computer (GPC) being left in an undefined microcode instruction state. In this state, the BFS GPC can no longer be used for flight-critical operations and may result in the loss of the crew and vehicle. This software problem is limited to the BFS; no similar problem exists in the Primary Avionics System Software (PASS). The anomaly was isolated to a Multifunctional CRT (Cathode Ray Tube) Display System (MCDS) service subroutine, named TWALT. TWALT can be called from both the MCDS Input Module (MIM) foreground task and the MCDS Update Module (MUM) background task. At the beginning of TWALT, the return address (from the register) of the calling module is temporarily stored in the local IPV, named HSDCRET. Upon exiting TWALT, the return address is fetched and used for the return branch to the calling module. As a background task, MUM can be interrupted, or suspended, while executing TWALT. While MUM execution of TWALT is suspended, MIM (as a foreground task) can call TWALT and overwrite the existing MUM return address in HSDCRET with the MIM return address. Later, when the background MUM resumes and executes the return branch from TWALT, it will actually return to the MIM because HSDCRET now has the MIM return address resident. This leads to the MUM attempting to execute incorrect MIM software, resulting in an undefined instruction state.

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	IPV anomaly found in BFS.	Occurrence of this anomaly is limited to situations where the foreground MIM call to TWALT interrupts a background MUM execution of TWALT. Foreground MIM calls to TWALT occur in response to a valid Item 40 entry, landing area selection, in Operational Sequence (OPS)-1 or in response to a valid Item 41 entry, landing area selection, in OPS-1 or OPS-6. Background MUM calls to TWALT occur once every major BFS cycle if the BFS Horizontal Situation Display (HSD) is active on a Display Electronics Unit (DEU). The BFS HSD is active during a TAL contingency and is used to select/reselect TAL landing sites. Testing has determined that this anomaly will not occur if the BFS HSD is not active. The probability of this anomaly occurring is considered extremely remote. A TAL contingency, resulting from a major system failure, is needed to require operational use of the BFS. Once in the BFS, the probability of occurrence has been calculated to be 1 in 40,000. This calculation is based on known BFS operation and software code lengths. A single halfword software patch has been identified to eliminate the potential for occurrence of this BFS anomaly. This patch would "no-op" the IPV/HSDCRET return address. This patch was not implemented for STS-48 because it is not desirable to patch flight software after testing and delivery to support a launch. For STS-48, an OPS note was developed that instructed the crew to disengage the BFS and engage the PASS HSD if a change in TAL landing sites was required. After the change is made, the crew is instructed to re-engage the BFS. Because the BFS monitors the status of the PASS, changes made in the PASS will be effected in the BFS. This procedural fix, however, is not available for a late TAL scenario because the BFS must be continuously engaged. Therefore, there is no time to disengage the BFS and make a landing site change in the PASS.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	IPV anomaly found in BFS.	Rationale for STS-48 flight was: • A major system failure is required to enter a TAL contingency. • The probability for this anomaly to occur in the BFS is 1 in 40,000. • A procedural OPS note was available to further minimize the potential for this anomaly to occur in the event a TAL contingency was required. ***This risk factor was acceptable for STS-48.***
7	STS-48/OV-103 Right-Hand (RH) Reaction Control System (RCS) Gaseous Helium (GHe) regulator leakage. ***With the measures taken to delay opening of the RH RCS GHe isolation valves until after the T-20 minute mark in the prelaunch countdown, no further problems with RH RCS fuel tank pressure rise were reported on STS-48.***	During pressurization of the RH RCS GHe tank, the RH RCS fuel tank pressure rose from 265 pounds per square inch gage (psig) to 296 psig. "A" and "B" leg regulators and isolation valves were cycled to locate the internal system leak. No leakage was found through the isolation valves. There was significant leakage, however, through both "A" and "B" leg regulators. Leakage through both regulators was in excess of the Operational Maintenance Requirements and Specifications Document (OMRSD) limit of 360 standard cubic centimeters per hour (scch). Regulator functional testing found that the leak on the "A" and "B" leg primary and secondary sides ranged from 2,300 scch to 13,115 scch. The "B" leg regulator demonstrated the worst leakage at an average rate of 12,784 scch on the secondary side. Leak rates stabilized after testing and were consistent through the time that the isolation valves were closed in preparation for launch. No further leakage or tank pressure increases were recorded.

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)	STS-48/OV-103 RH RCS GHe regulator leakage.	A change in the countdown procedure was implemented to delay opening the RH RCS GHe isolation valves until after the T–20 minute mark. Normally, RCS isolation valves are opened at crew ingress. Delaying the isolation valve opening minimized the potential for reaching critical fuel tank pressures during the mission. At the last recorded leak rate, it would take approximately 11 hours to reach critical pressures with the isolation valves open. Fuel consumption during flight will relieve increasing tank pressures. In the event that the RH RCS fuel tank pressure continues to increase, the fuel-side burst disc and relief valve will relieve GHe pressure. The burst disc will rupture at 324 to 340 psig, and the relief valve opens at 315 psig. Rationale for STS-48 flight was: • A change in the countdown procedure was implemented to minimize RH RCS fuel tank pressure prior to launch by delaying the opening of the GHe isolation valves. • After launch, fuel consumption will reduce the rate of pressure increase in the RH RCS fuel tank. • A burst disc and relief valve were in place to eliminate the potential for RCS fuel tank rupture. ***This risk factor was acceptable for STS-48.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1	Undetected cold wall coolant tube leaks. HR No. ME-B7 (All Phases) {C} ***No Space Shuttle Main Engine anomalies were reported on STS-48.***	Engine #2107, nozzle #4012, was leak checked over 240° of the circumference after STS-37 at Dryden; no leaks were noted during the standard soap leak check. However, when reentry insulation was removed at Kennedy Space Center (KSC) to install fatigue arrestors, backside heating was noted on the tenth bay panel; this is an indication of tube leaks during ascent. Additional leak checks were performed that revealed: 5 Class III leaks, 1 Class II leak, and 1 Class I leak. The projected total fuel leakage rate was 0.0094 pounds/second (hydrogen at 104% of rated power level). Nozzle #4012 was removed from engine #2107 and sent to the KSC engine facility for further examination. Nozzle #4016 was installed on engine #2107 for STS-48/OV-103. Leak checks were performed to verify integrity of nozzle #4016 tubes. Nozzle #4012 tubes were opened and examined. Corrosion was found on the cold wall outside diameter at the aft manifold lip. The corrosion appeared to be caused by etchant residue from a feedline replacement during fabrication; the etchant is used to prepare feedline stubouts and bracket welds. There were 3 previous cases of known tube corrosion: 2 due to copper plating repair in the forward manifold, and 1 due to a nickel-plated patch. Based on previous hot-fire history, leaks are normally located at aft manifold stubouts and are attributed to High-Cycle Fatigue (HCF). The nozzle #4012 leaks were adjacent to the primary component fuel drain line.

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1 (Continued)	Undetected cold wall coolant tube leaks.	Rationale for STS-48 flight was: • STS-48/OV-103 engine #2107, nozzle #4016, was leak checked 360°; all leak checks were satisfactory. (No tenth bay panel insulation was installed on this engine because it was in position #1.) • The leaks on engine #2107, nozzle #4012, were unique, resulting from corrosion caused by residual etchant from the feedline replacement during fabrication. • The leaks on engine #2107, nozzle #4012, did not result in damage to the insulation; there was no loss of protection for reentry heating. • Hot-fire data did not indicate any effect on engine performance from the tube leaks . ***This risk factor was resolved for STS-48.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
2	Low-Pressure Oxidizer Turbopump (LPOTP) #2222 turbine stator vane crack. HR No. ME-C1 (All Phases) {AR} ME-C2 (All Phases) {C} ***No Space Shuttle Main Engine anomalies were reported on STS-48.***	A crack in a turbine stator vane of LPOTP #2222 was discovered during a routine dye-penetrant inspection. The crack, 0.125 inch in length, was located on 1 turbine stator trailing edge in the third row of 5 turbine stators. LPOTP #2222 was a high-time development unit with 59 starts and 24,153 seconds (sec) of operation. The concern was the potential for a through crack in a turbine stator vane, resulting in catastrophic shutdown of the LPOTP, or the catastrophic shutdown of the High-Pressure Oxidizer Turbopump (HPOTP) downstream of the LPOTP due to loss and migration of a LPOTP turbine stator vane. Cracks in turbine stator vanes have been experienced in the past. However, the cracked vanes were limited to 2 turbine stators that were machined using Electrical Discharge Machining (EDM). Cracks were initiated at the EDM recast layer pockets on the stator vane trailing edge root radii. Failure analysis determined that these cracks resulted from High-Cycle Fatigue (HCF). The crack located in the #2222 turbine stator vane was the first on a conventionally-machined stator. Failure analysis and examination confirmed that this crack was also the result of HCF. Crack initiation was most likely due to vane resonance. Because the vane responds as a system, it was most likely excited during a major portion of the pump operating range. Metallurgical and fracture analysis are in work to predict the number of cycles required for crack initiation and to determine the rate of crack growth. Deviation Approval Request (DAR) 2545 set the life-limit for LPOTP turbines. This life-limit has been further revised since the problem was first identified during the STS-43 mission reviews. Based on metallurgical and fracture analysis, DAR 2545 set the LPOTP turbine life-limit at 11,724 sec of operation; the point where LPOTP #2222 turbine stators were last inspected. DAR 2545 also requires

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
2 (Continued)	LPOTP #2222 turbine stator vane crack.	periodic inspections prior to the life-limit. The first inspection point is at 6,038 sec, 25% of 24,253 sec, the point were turbine stator cracks were found on LPOTP #2222, and every 3,000 sec of operation thereafter until the life-limit.

STS-48 LPOTPs were within the DAR 2545 inspection and life-limits. The following were the times remaining on STS-48 LPOTP turbine stators prior to the initial inspection point of 6,038 sec:

LPOTP	Turbine S/N	Total Time	Time Remaining
#2025R1	8661639	5,351 sec	687 sec
#2120	8469685	4,215 sec	1,823 sec
#2216	8636149	3,407 sec	2,631 sec

For LPOTP #2025R1, turbine S/N 8661639, the time remaining to the inspection point was sufficient to support a nominal STS-48 mission profile (520 sec). However, in the event of an abort, 230 sec of addition run time takes S/N 8661639 above the inspection point by 63 sec. This was not considered a concern because the total run time after an abort profile (6,101 sec) was nearly 1/2 the time that the LPOTP #2222 unit was last inspected (11,724 sec). It was not believed that the unit failed directly after the inspection.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
2 (Continued)	LPOTP #2222 turbine stator vane crack.	Rationale for STS-48 flight was: • Flight units had a margin >4 (postflight) on seconds compared to the turbine life-limit of 11,724 seconds of operation set by DAR 2545 for LPOTP turbines. • Flight units had sufficient operating time remaining to complete a nominal STS-48 mission profile before the initial DAR inspection point. The exceedance in the event of an abort was not a concern. ***This risk factor was acceptable for STS-48.***
3	High-Pressure Fuel Turbopump (HPFTP) thrust ball cracks. HR No. ME-D1 Rev. F {AR} (All Phases) ***No Space Shuttle Main Engine anomalies were reported on STS-48.***	Four cracks on the HPFTP thrust ball #6008 were found upon disassembly of STS-38 engine #2019 at Canoga Park. The specification indicates no cracks are allowed. These cracks, approximately 0.10-0.15 inch long, ran perpendicular to the wear pattern. All were separate cracks (not run together). No spalling was observed. Two cracks were also noted on the thrust ball of a recent development unit, HPFTP #2814R1. The cracks on both balls were initially found by visual inspection; however, detection with unaided eyes is very difficult. No abnormal conditions were noted on the HPFTP #6008 thrust bearing (RS007605). No abnormal startup and shutdown transients were observed. The shaft insert contacts the ball during startup and shutdown only; approximately 2 seconds (sec) at start and 8-10 sec at shutdown. The thrust ball and insert are inspected after every hot-fire, including flight. Moly lube is added as required to

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
3 (Continued)	HPFTP thrust ball cracks.	minimize wear; this is allowed by the specification. Wear galling and cracks on the balls are not permitted. The Failure Modes and Effects Analysis (FMEA) classifies HPFTP thrust ball cracks as Crit 1 failures due to the possibility of the ball chipping and the pieces entering the fuel stream. However, there has been no loss of major pieces of the thrust ball in the Program history. Rationale for flight of STS-48 was: • There was no evidence that pieces would chip off the HPFTP thrust ball if similar cracks developed during engine start. • No evidence was found of loss of large pieces of HPFTP thrust balls in the Program history. • All 3 thrust balls in the STS-48/OV-103 HPFTPs were visually inspected; no cracks were noted. ***This risk factor was resolved for STS-48.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
4	Low-pressure oxidizer duct flex joint bellows life issue. HR No. ME-C2 (All Phases) {AR} ***No Space Shuttle Main Engine anomalies were reported on STS-48.***	Low-pressure oxidizer duct flex joints have a life-limit requirement to protect against tripod failures during Space Shuttle Main Engine (SSME) operation. The life-limit for flight units has been established at 18,284 seconds (sec) of equivalent full-power-level operation. Periodic visual inspection and leak testing are performed on all flight units. A fleet leader sectioning plan has been implemented to allow low-pressure oxidizer ducts above the flight life-limit to be used for ground testing. To date, there have been no low-pressure oxidizer duct failures during SSME operations. Implementation of the fleet leader sectioning plan led to the discovery of anomalous conditions in high-time, low-pressure oxidizer ducts. Sectioning of the fleet leader, with 61 tests/operations at Liquid Oxygen (LO_2) operating pressure and 24,472 sec of operation, identified inner ply distortion and a ductile overload tear along bellows weld #4. Pre-sectioning leak tests identified no external leakage. The fatigue crack was across the longitudinal seam weld at the inner ply crown. Sectioning of a second unit with 48 tests and 20,539 sec of operation identified no anomalies. A third unit, with 41 tests and 17,402 sec of operation, was sectioned after the discovery of an external pinhole leak at bellows weld #16. This leak was attributed to a manufacturing defect, and no flow growth was evident. Mass spectrometer leak tests identified a leak at the bellows weld #4 inner ply; however, the leak source was not detectable through visual inspection or by sectioning. There was no record of "frost" during operational testing. Except for the fleet leader, no other fatigue cracks were found.

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
4 (Continued)	Low-pressure oxidizer duct flex joint bellows life issue.	The concern with these findings was that an undetected fatigue crack could propagate through the inner bellows ply, leading to an LO_2 leak into the inner bellows cavity during SSME operation. If this were to occur, the LO_2 would expand during post-test warmup and result in the potential for inner ply deformation and tearing at bellows weld #4. It was postulated that if this phenomenon occurred, the localized deformation (bulges) would be visually apparent external to the duct bellows. The resulting duct bellows structural factor of safety for 2 of the 3 plies remaining (1 ply failure) was calculated to be 1.6 against ultimate and 1.3 against yield.

In comparison to the high-time ducts that were sectioned, the low-pressure oxidizer duct bellows installed on STS-48/OV-103 had relatively low operating times and LO_2 pressure tests/operations. Following is the STS-48/OV-103 data:

SSME	Duct S/N	Tests	Time (sec)
#2019	4877675	25	8,628
#2107	4878908	22	7,612
#2031	4922633	11	5,052

STS-48/OV-103 low-pressure oxidizer duct bellows were visually inspected for external bulges; none were found. The STS-48 SSMEs passed encapsulated helium leak test prior to installation on OV-103.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
4 (Continued)	Low-pressure oxidizer duct flex joint bellows life issue.	Rationale for STS-48 flight was: • STS-48/OV-103 duct bellows had low time compared to the fleet leader found with fatigue cracks; sectioned bellows with 20,539 sec and 17,402 sec of operation were found with no fatigue cracks. • STS-48/OV-103 duct bellows were visually inspected, and SSMEs passed encapsulated helium leak tests. *This risk factor was acceptable for STS-48.*
5	High-Pressure Fuel Turbopump (HPFTP) #5602R1 failure on engine #0215 during ground test. HR No. ME-D1 (All Phases) {AR} *No Space Shuttle Main Engine anomalies were reported on STS-48.*	HPFTP #5602R1 on engine #0215 failed during ground testing on July 24, 1991. The test was terminated 4.33 seconds (sec) after engine start due to loss of fuel flow, and resulted in extensive hot-gas system erosion. HPFTP #5602 was the fleet leader and was not on a flight configuration engine at the time of the failure. The investigation determined that the problem was caused by turbine blade (#48) failure induced by rubbing. Metallurgical analysis concluded that the fracture occurred above the fourth firtree lobe of the blade. Ductile overload was initiated from the blade suction side resulting in the failure. A "thumbnail" feature, recessed below the fracture plane, was found near, but not through, the blade suction-side surface. The "thumbnail" feature was determined to be a pre-existing, hydrogen environment embrittlement crack. (The tip of the crack was blunted and did not progress to failure; however, it clearly weakened the primary fracture plane.)

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
5 (Continued)	HPFTP #5602R1 failure on engine #0215 during ground test.	The firtree casting was also found to have interdendritic, micro-shrink porosity. Multiple cracks, near the blade-pressure side in the region of the second and third lobes, were associated with porosity. Fractography inspection of the third-lobe crack confirmed the existence of hydrogen environment embrittlement. The third-lobe crack originated at the subsurface porosity with visible fracture surface arrest lines. No porosity or cracks were found in sections of 10 other second-stage blades. Two additional fleet leader blades were also inspected for porosity and cracks, with none found. Investigation into the lot from which blade #48, S/N AN652 (35 starts and 13,265 sec of operation), was purchased, determined that 44 of 64 blades had been scrapped due to various manufacturing deficiencies. The location of the remaining 19 blades was determined: one, S/N AN624 with 6 starts and 2,574 sec of operation, was in HPFTP #2323R2 on STS-48/OV-103 engine #2031. Evaluation of other blades in this lot did not identify similar cracking or porosity. There was no indication of a generic problem with blades in this lot. HPFTPs on STS-48/OV-103 were considered acceptable for flight based on the evaluation of all potential failure modes. Operating times and starts were very low compared to the failed unit. All fabrication, inspection, and repair histories of the STS-48/OV-103 HPFTP turbine blades were within the flight turbine experience base.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
5 (Continued)	HPFTP #5602R1 failure on engine #0215 during ground test.	Rationale for STS-48 flight was: • HPFTP #5602R1 had a high-time turbine installed at the time of the failure. • Engine #0215 was not in flight configuration at the time of the failure; a flight engine would have automatically shut down in 1-2 sec after start if the identical failure occurred. • A similar failure in a flight engine would result in an on-pad abort. ***This risk factor was acceptable for STS-48.***
6	SSME controller F24 Digital Computer Unit (DCU) "B" memory alteration issue. HR No. INTG-165B {AR} ***No Space Shuttle Main Engine anomalies were reported on STS-48.***	Engine #2011, controller F24, last flew on STS-41 and was returned to Rocketdyne/Canoga Park after the flight for refurbishment. During preparation for engine #2011 green-run testing at Stennis Space Center (SSC), controller F24 DCU "B" would not cycle during the initial power-up. An unsuccessful attempt was made to perform an autostatic controller memory dump. The memory loader unit was connected, and a controller memory dump was successfully performed. A comparison of the memory dump to the master magnetic tape was performed, resulting in the determination that certain memory locations had been altered. The memory alterations, identified by memory parity error indications, caused DCU "B" to be inoperative. Controller F24 was removed from engine #2011 and sent to Honeywell, the vendor, for analysis.

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
6 (Continued)	SSME controller F24 DCU "B" memory alteration issue.	Visual inspection of controller F24 at Honeywell revealed 2 pins, pin #21 and pin #30, were shorted in the J303 connector. Pin #30 is used to increment the controller program counter during diagnostic checkouts. The short caused the program counter to erroneously increment, resulting in the controller software randomly skipping instructions. DCU "B" eventually halted because it was not cycling properly. The J303 connector is used for ground tests and factory diagnostics only, and was environmentally enclosed for flight. Controller F24 experienced no adverse affects (hardware failures) from the shorted pins; SSME controllers are designed to withstand shorts without damage. Rationale for STS-48 flight was: • Controllers installed on STS-48/OV-103 SSMEs successfully passed all green-run tests. Continued Operational Maintenance Requirements and Specifications Document testing through SSME start would identify a similar problem, resulting in a scrub and analysis. • Connector J303 was not used for flight and was environmentally sealed. ***This risk factor was acceptable for STS-48.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
7	STS-48/OV-103 Main Engine 2 (ME-2) Main Oxidizer Valve Actuator (MOVA) failures. HR No. ME-A2A {C} ME-A2P {C} ME-B4A {AR} ME-B4C {AR} ***No Space Shuttle Main Engine anomalies were reported on STS-48.***	STS-48/OV-103 ME-2 MOVA, Serial Number (S/N) 107, failed to go into hydraulic lockup during nominal Flight Readiness Tests (FRTs) on August 27, 1991. MOVA, S/N 107, had experienced 2 operational engine firings and a total of 819 seconds (sec) of operation prior to this failure. During ME-2 FRT, main oxidizer valve movement was noted due to hydraulic actuation while in the fail-safe configuration. This confirmed that the problem was not a Space Shuttle Main Engine (SSME) controller problem, and the decision was made to remove the main oxidizer valve and actuator. The valve and actuator were returned to Rocketdyne/Canoga Park for failure analysis. A similar failure in flight would not have any adverse effects unless there was a previous Auxiliary Power Unit (APU) failure requiring hydraulic lockup of the SSME. In the case of hydraulic lockup, pneumatic SSME shutdown would be required. This MOVA failure mode would prevent hydraulic lockup and pneumatic SSME shutdown, and would result in potential catastrophic shutdown of the SSME at propellant depletion. This failure mode also raised the concern for a potential generic failure mode in MOVAs. This concern was amplified because the actuator spool/sleeve is identical for the Oxidizer Preburner Oxidizer Valve (OPOV), the Fuel Preburner Oxidizer Valve (FPOV), and the Main Fuel Valve (MFV). Disassembly of S/N 107 determined that the fail-safe bypass spool/sleeve was jammed, preventing movement of the spool to the lockup position. Galling and metal smearing was found on both the spool and the sleeve. No contamination was found during this investigation. There was 1 similar failure in SSME Program history. FPOVA, S/N 090, failed during preflight testing on STS-26. Similar

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
7 (Continued)	STS-48/OV-103 ME-2 MOVA failures.	galling of the actuator spool and sleeve was identified. The S/N 090 failure was attributed to contamination of the fail-safe servoswitch controlling the actuator. Corrective action was implemented after the S/N 090 failure to reduce the potential for introducing contamination into valve actuators. Because no contamination was found in S/N 107, there was a concern that the original S/N 090 diagnosis may have been incorrect and there was another contributor to this failure mode. Analysis into potential failure mechanisms continues. Checkout of the replacement valve and actuator, S/N 057, identified a problem with the servovalve current bias when checked at the 90% commanded position. Current oscillations were witnessed while under control on both channel "A" and channel "B". S/N 057 had experienced only 1 green-run hot-fire prior to installation on STS-48/OV-103 ME-2. S/N 057 was removed and returned to Rocketdyne/Canoga Park for failure analysis. Troubleshooting isolated the failure to the Rotary Variable Differential Transducer (RVDT) used to indicate the angular position of the MOVA. RVDT teardown identified that the epoxy used to attach the mid-bearing and shields to the rotor had expanded. This created the potential for rotor binding and/or shaft deformation. The area of expanded epoxy was removed, the RVDT was reassembled, and testing determined S/N 057 to be normal. The location of RVDTs from the same manufacturing lot as S/N 057 was identified. Those in stock were inspected and found not to have similar epoxy expansion. Three MOVAs with RVDTs from the same lot were found to be on STS-48/OV-103 ME-2 (#2031). Unlike S/N 057, those located on STS-48 had experienced multiple engine hot-fires and tests, and no anomalies had been recorded.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
7 (Continued)	STS-48/OV-103 ME-2 MOVA failures.	MOVA failures are considered screenable prior to launch. SSME green-run engine firings and prelaunch SSME FRTs are performed to identify similar valve problems. To date, all failures have been found during these tests. Additionally, fail-safe servoswitches and bypass valves are transitioned from "off" to "on" at T–4 minutes (min) as a last Launch Commit Criteria (LCC) check prior to engine start. Failure at this point in the countdown will result in a launch scrub. Rationale for STS-48 flight was: • Main oxidizer valve failures are screenable prior to launch; all similar failures were identified during SSME green-run tests or FRTs. A similar S/N 107 failure would require a previous APU failure to be evident and lead to a catastrophic event. • LCC verifies Main Oxidizer Valve (MOV) bypass valve movement from pneumatic shutdown position to normal run position at T–4 min. • S/N 057 was replaced and passed ME-2 FRT requirements. ***This risk factor was resolved for STS-48.***

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1	Nozzle fixed housing, joint #5 secondary seal integrity issue. HR No. BC-04 DCN-113 {C} BN-03 Rev. D {AR} ***No Solid Rocket Motor anomalies were reported on STS-48.***	Integrity of the nozzle joint #5 secondary seal came into question due to the recent discovery that the nozzle fixed housing bolt chamfer and spot face is adversely affected by glass beading operation during refurbishment. The glass beading operation caused metal displacement from the edge of the chamfer area. Corrective action was identified to avoid metal displacement in this area; however, this action was not implemented at the time of STS-48 nozzle refurbishment. Packing with retainers acts as a secondary seal on the 72 fully-threaded bolts at joint #5. The sealing surface comprises overlapping elastomer on the spot face and any additional area on the chamfer. This configuration was questioned prior to STS-26. Packing with retainer testing was performed to verify sealing integrity. Flaw sizes up to 0.007-inch depth were introduced on the retainer in combination with flaw sizes up to 0.003-inch depth on the associated washer. This test confirmed metal-to-metal sealing for the joint #5 configuration for full motor pressure and joint movement. In addition, 39 ground-test and flight nozzles have been fully inspected to date. No soot was observed in this area. An intentionally flawed joint on a test motor was the only nozzle that allowed pressure to the primary O-ring. In this case, the primary O-ring sealed, and no anomalous conditions, such as sooting, were observed. During buildup, all nozzle joints are leak tested. The maximum allowable leak rate for joint #5 is 0.084 standard cubic centimeters per second (sccs) at 920 pounds per square inch gage (psig) and 0.0082 sccs at 30 psig. Leak tests also include bubble tests, with no bubbles allowed. All STS-48 nozzle joints, including joint #5, successfully passed required leak tests.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1 (Continued)	Nozzle fixed housing, joint #5 secondary seal integrity issue.	An anonymous letter, dated August 30, 1991 and addressed to the Director, Space Shuttle Program, identified several concerns that an individual or individuals had concerning this risk factor. These concerns were reevaluated by Thiokol Corporation management, Marshall Space Flight Center Redesigned Solid Rocket Motor (RSRM) Project Management, the System Safety Review Panel, and the Space Shuttle Program Management. The result of this reevaluation found that there were no residual concerns with STS-48 nozzle joints, including joint #5; STS-48 was safe to fly. Rationale for STS-48 flight was: • Packing with retainer provides a metal-to-metal seal at joint #5 under dynamic load conditions. • STS-48 nozzle joints passed all required leak tests. • All technical information and presentation material used in the assessment of this issue were re-reviewed to confirm that valid and complete data were used in the evaluation of this issue. ***This risk factor was resolved for STS-48.***

RESOLVED STS-48 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2	Test Evaluation Motor-8 (TEM-8) forward segment propellant stress relief flap erosion. HR No. BC-10 Rev. D {C} ***No Solid Rocket Motor anomalies were reported on STS-48.***	Post-test teardown and evaluation of TEM-8 identified abnormal insulation erosion that had the potential for violating thermal and erosion safety factors. TEM-8 was in a High Performance Motor (HPM) configuration for the static test and was the fleet leader for age at firing, 5 1/2 years. Inspection found that the forward segment propellant stress relief flap was partially missing. Flap material was missing on the top portion of the segment, in varying depths across a circumferential arc from 94° to 230°. The 180° position was on the top with the TEM-8 motor in the horizontal position for the static test firing. Three char pattern lobes were located at the 118° (small), 176° (large), and 212° (large) position on the flap. The Nitrite Butadiene Rubber (NBR) inhibitor on the opposite side of the joint slot appeared normal, indicating no jet-impingement effects. Normal amounts of castable inhibitor also remained. Erosion in the lobe areas and at missing flap locations resulted in a calculated Factor of Safety (FOS) of 2.45. A review of TEM-8 performance data indicated no noticeable change in motor pressure during the test. The investigation into this anomalous erosion determined that the abnormal erosion was caused by early propellant initiation on the inboard portion of the castable inhibitor at the 176° and 212° locations. The most probable cause for the large lobes at these locations was anomalies in the castable inhibitor. Castable inhibitor anomalies can result from thin spots in the inhibitor, unbonds, or defects that act as a fuse to prematurely ignite surrounding propellant. Propellant burning at the 118° location occurred later in the motor burn and initiated on the flap, 5.5 inch forward

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
2 (Continued)	TEM-8 forward segment propellant stress relief flap erosion.	of the flap-to-inhibitor interface. It was believed that bond line failures at the flap-to-propellant or the castable inhibitor-to-propellant interfaces at this location led to the abnormal erosion pattern in this area. There have been numerous processing enhancements implemented between the HPM configuration and the Redesigned Solid Rocket Motor (RSRM) configuration used for flight. Process controls and insulation inspections are more extensive for RSRMs. In the area of anomalous insulation erosion on TEM-8, RSRMs are 100% x-ray and ultrasonically inspected. No abnormal erosion patterns similar to those experienced in TEM-8 have been witnessed in the RSRM flight history, and all 37 RSRMs have met the required insulation FOS. There was no anomalous performance history of forward castable inhibitor in any previous HPM or RSRM, a total of 103 motors. The RSRM insulation design is considered fail-safe for a total inhibitor failure at motor ignition. Rationale for STS-48 flight was: • TEM-8 exceeded the insulation FOS with the anomalous erosion patterns. • RSRM insulation design was sufficient to handle anomalous erosion patterns as experienced on TEM-8. • RSRM process controls and inspections far exceed those of the HPM configuration. ***This risk factor was resolved for STS-48.***

SECTION 5

STS-43 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-43/OV-104 mission, the previous Space Shuttle flight. Each anomaly is briefly described, and risk acceptance information and rationale are provided.

Hazard Reports (HRs) 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.

The following risk factor in this section represents a low-to-moderate increase in risk above the Level I approved Hazard Baseline. The NASA Safety Community assessed the relative risk increase of each and determined that the associated increase was acceptable.

Orbiter 5	Contamination and leakage of the Liquid Hydrogen 4-inch disconnect.

SECTION 5 INDEX

STS-43 INFLIGHT ANOMALIES

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1	Water Spray Boiler (WSB) #2 did not provide lube oil cooling on ascent. IFA No. STS-43-V-02 HR No. ORBI-121 {AR} ***No WSB lube oil cooling anomalies were reported on STS-48.***	During ascent, WSB #2, Serial Number (S/N) 018, gave no indication that it was cooling lube oil from Auxiliary Power Unit (APU) #2, S/N 208. Lube oil return temperatures reached 323°F before APU #2 was shut down; lube oil temperatures should not exceed 250°F. Instrumentation indicated that WSB #2 core temperature steadily decreased through ascent, reaching a low of 31°F after Main Engine Cutoff (MECO), indicating freeze-up of the WSB #2 spray bar. Switching between controller "A" and "B" had no effect on lube oil cooling. The APU #2 gearbox forward bearing temperature reached 351°F. Because the gearbox temperatures exceeded 350°F, Flight Rule 10-14 was invoked and APU #2 was shut down directly after MECO. The APU gearbox is certified to 400°F. During on-orbit flight control checkout, WSB #2/APU #2 were operated for 11 minutes. Again, no lube oil cooling was observed. The crew switched between controller "A" and "B" with no effect. APU #2 gearbox temperatures reached 341°F during this period, and the lube oil return temperature topped out at 305°F. Because WSB #2 cooling failure was confirmed, Flight Rule 10-23 was invoked to allow for late turn-on of APU #2 [at Terminal Area Energy Management (TAEM)]. Lube oil temperatures again ran high until APU #2 was shut down at wheel stop. The lube oil and the hydraulic spray valves were removed and replaced after troubleshooting at the Kennedy Space Center (KSC). APU/WSB system #2 will be hot-oil flushed prior to the next flight of OV-104, STS-44. A similar, less severe WSB #2 freeze-up occurred on the last OV-104 flight, STS-37. During STS-37, inflight checkout determined the maximum lube oil return temperature was 279°F. The Operational Maintenance Requirements and

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	WSB #2 did not provide lube oil cooling on ascent.	Specifications Document (OMRSD) requires a hot-oil flush of the APU/WSB in the event that lube oil return temperatures exceed 275°F during a mission. However, because the exceedance was only 4°F above the requirement, and because the lube oil out pressure was nominal (past WSB problems associated wax buildup in the lube oil with low cooling), an exception to the requirement was approved for STS-43. Pre-STS-43 WSB #2 checkout included valve flow checks and electrical function checks; no anomalies were reported. Past flights of OV-103 have shown no similar APU #1/#2 or WSB #1/#2/#3 anomalies which could be attributed to contamination or spray bar freeze-up. APU #3, recently removed from OV-102 and installed on OV-103, operated with no anomalies on STS-40. APU #3 was hot-fired at the pad in preparation for STS-48, with no lube oil cooling problems noted. All appropriate OMRSD ground checkouts were performed on STS-48/OV-103 APUs/WSBs.

STS-43 INFLIGHT ANOMALIES

<table>
<tr><th>ELEMENT/
SEQ. NO.</th><th>ANOMALY</th><th>COMMENTS/RISK ACCEPTANCE
RATIONALE</th></tr>
<tr><td><u>ORBITER</u></td><td></td><td></td></tr>
<tr><td>1 (Continued)</td><td>WSB #2 did not provide lube oil cooling on ascent.</td><td>Rationale for STS-48 flight was:

• The anomalous APU lube oil and gearbox temperatures were within the APU certification limits.

• Appropriate flight rules were invoked to limit damage to APU #2/ WSB #2. Identical flight rules were in place for STS-48 if a similar anomaly occurred.

• Past OV-103 missions have demonstrated proper APU/WSB operation.

This anomaly/risk factor was resolved for STS-48.</td></tr>
<tr><td>2</td><td>Power Reactant Supply and Distribution (PRSD) Hydrogen (H_2) tank #1, heater "B" failed off.

IFA No. STS-43-V-04

HR No. ORBI-089A {AR}
ORBI-094 {AR}

No PRSD H_2 tank heater anomalies were reported on STS-48.</td><td>PRSD H_2 tank #1, heater "B" failed off on flight day #2. Tank #1 pressure rise rate confirmed that only 1 of 2 heaters was operational. An attempt to manually bring heater "B" back on-line was unsuccessful. The decision was made to continue operations on heater "A", but also to use H_2 from tank #1 at a slightly higher rate to protect against the potential for tank #1 heater "A" failure to adversely impact consumable margins. No further PRSD heater problems were experienced during the remainder of the flight.

Troubleshooting efforts at Kennedy Space Center identified that a 1-ampere fuse had blown in the circuit between the tank #1 heater "B" switch and the associated Remote Power Controller (RPC). The fuse was replaced, and tank #1 heater "B"</td></tr>
</table>

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	PRSD H_2 tank #1, heater "B" failed off.	operated nominally. Laboratory analysis indicated that the fuse was defective; this was not considered to be a generic failure problem. Rationale for STS-48 flight was: • There were no previous PRSD tank heater problems reported on OV-103. • There are redundant heater elements in each PRSD H_2 tank, and the PRSD H_2 tanks are redundant. • Flight Rule 9-161, "Cryo Heater Management for Orbit", was in place to handle the loss of H_2 tank heaters. ***This anomaly/risk factor was resolved for STS-48.***

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Power Reactant Supply and Distribution (PRSD) Hydrogen (H_2) manifold #1 isolation valve failed to close. IFA No. STS-43-V-09 HR No. ORBI-286A {AR} ***No PRSD H_2 manifold isolation valve anomalies were reported on STS-48.***	PRSD H_2 manifold #1 isolation valve failed to close when commanded on flight day #7. The isolation valve had successfully closed 5 times during the mission prior to the occurrence of this failure. The H_2 isolation valve that failed on STS-43 is the same model as an Oxygen (O_2) isolation valve that failed to close when commanded on STS-34/OV-104 and STS-37/OV-104. The H_2 manifold isolation valve was left open for the remainder of the STS-43 mission. The isolation valve, in the H_2 or O_2 position, is used to isolate PRSD system leaks. There are redundant manifold isolation valves in each system. There have been no similar PRSD manifold isolation valve anomalies on OV-103. This anomaly could not be repeated during Kennedy Space Center postflight troubleshooting. Instrumentation of the valve command circuit was approved for the next flight of OV-104. Rationale for STS-48 flight was: • PRSD manifold isolation valves are redundant. • Leak checks were successfully performed on the STS-48/OV-103 PRSD system. • There was no history of similar PRSD manifold isolation valve failures on OV-103. ***This anomaly/risk factor was resolved for STS-48.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Auxiliary Power Unit (APU) #1, Serial Number (S/N) 305, anomalous Gas Generator (GG) chamber pressure during entry. IFA No. STS-43-V-12 HR No. ORBI-031 {AR} ***No erratic APU GG chamber pressure anomalies were reported on STS-48.***	Postflight evaluation of APU #1, S/N 305, data identified the potential for Pulse Control Valve (PCV) leakage for a period of 35 seconds during reentry. This was demonstrated by the GG chamber pressure traces returning to a non-zero level after each pulse. The GG chamber pressure trace was normal prior to the anomaly and gradually returned to normal soon thereafter. At no time did APU #1 performance decrease or turbine speed increase to the point that crew action was required. The most probable cause of this anomaly was initially thought to be transient contamination that prevented the PCV from closing completely. The concern with this anomaly was the potential for a "smart particle" (contamination) to become lodged on the Shutoff Valve (SOV) seat and result in uncontrolled APU turbine overspeed. APU #1 was removed and sent to Sundstrand, the vendor, for failure analysis. Sunstrand found that the PCV and SOV were leaking beyond specification due to a known age/cycle-related failure mode of the valve seats. Rationale for STS-48 flight was: • OV-103 APUs successfully passed Gas Generator Valve Module liquid leak checks, indicating no PCV or SOV anomalies. • The SOV provides turbine speed control (high-speed mode) in the event the PCV fails open. ***This anomaly/risk factor was acceptable for STS-48.***

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	Contamination and leakage of the Liquid Hydrogen (LH_2) 4-inch disconnect. IFA No. STS-43-V-13 HR No. INTG-041 {AR} ORBI-035A {AR} ***No anomalies associated with the LH_2 4-inch disconnect were reported on STS-48.***	Postlanding inspection of STS-43/OV-104 identified an audible leak emanating from the LH_2-side, 4-inch disconnect. A piece of plastic flapper valve seal material was found wedged in the flapper. The wedged seal material represented approximately 12.5% of the total flapper valve seal. Damage to the seal most likely happened under cryogenic conditions after Main Engine Cutoff (MECO); the 4-inch flapper valve is closed immediately following shutdown of 1 or more Space Shuttle Main Engines (SSMEs) and prior to External Tank (ET) separation. Flapper closure is effected to prevent overboard hydrogen leakage and loss of helium at ET separation. The concerns associated with this anomaly are the ability of the flapper valve to seal properly. In the case of an on-pad abort, a partially-open 4-inch flapper valve could lead to the failure to isolate LH_2 from a shutdown SSME. In this case, the 4-inch flapper valve in the ET disconnect provides redundancy. Failure of the 4-inch flapper to seal properly post-MECO and after ET separation represents the worst-case, Crit 1 scenario for a partially-open 4-inch flapper valve. The concern here is with the potential for hydrogen ingestion into the aft compartment and helium depletion. For nominal End-Of-Mission (EOM) entry, Abort-Once-Around (AOA), or Abort-To-Orbit (ATO) mission profiles, failure of the 4-inch flapper to

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	Contamination and leakage of the LH_2 4-inch disconnect.	seal properly is not considered catastrophic because the amount of hydrogen left in the system is minimal. In the case of a Return-To-Launch Site (RTLS) or a Transoceanic Abort Landing (TAL) contingency, the risk of hydrogen ingestion increases due to the amount of hydrogen remaining in the system at the time the ET umbilical doors close. For a TAL contingency, the residual risk was originally identified and baselined as being lower because there is approximately 3 pounds-mass (lbm) of hydrogen in the system when the ET umbilical doors close [MECO + 1 minute]. It was originally believed that the aft compartment helium purge during entry should dilute this amount of hydrogen in the aft compartment to an extent and reduce the risk of ignition. Recently completed analysis, discussed below, indicated that the aft compartment helium purge is not adequate to eliminate the risk of a hydrogen fire. For an RTLS contingency, as much as 150 lbm of hydrogen can remain in the LH_2 manifold system when the ET umbilical doors are closed [MECO + 55 seconds (sec); Major Mode 602 (MM602) + 30 sec]. Since RTLS dump flowpaths are open at this time, it is not known how much hydrogen might leak through a broken seal into the aft compartment. Hydrogen accumulation in the aft compartment, coupled with the presence of ignition sources and air during reentry, presents the potential for a flammability hazard.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	Contamination and leakage of the LH_2 4-inch disconnect.	A similar blowing leak through the 4-inch flapper valve was experienced on STS-26/OV-103. Troubleshooting determined that this problem was due to improper flapper valve shimming. The build drawings provided insufficient guidance to shim the flapper valve properly. A correction was made to the drawings to eliminate the potential for future improper shimming. The STS-26/OV-103 and the STS-43/OV-104 anomalies are not considered related; however, damage to the 4-inch flapper seal is in the identical location on the seal. Recent kinematic analysis confirmed the potential for interference during valve closure due to improper shimming of the flapper. Excessive flapper angulation, or "wobble", can lead to premature seal-to-seat contact. Computer-aided drawing analysis confirmed that premature contact is exaggerated under cryogenic conditions. A tool has been developed to accurately measure proper seal clearance in ambient temperatures; however, this tool was not available for STS-48/OV-103 utilization. For STS-48/OV-103, the 4-inch flapper valve seals could not be inspected because the Orbiter and ET were already mated. A large volume decay/leak test was performed to provide confidence that the flapper valve seal was functioning properly. No problems were identified with OV-103 flapper valves/seals since the proper shimming of the 4-inch flapper valve after STS-26. In comparison to the STS-43/OV-104 4-inch flapper seal that had 9 flights and 21 cryogenic cycles prior to the anomaly, the STS-48/OV-103 4-inch flapper seal had 5 flights and 6 cryogenic cycles.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	Contamination and leakage of the LH_2 4-inch disconnect.	Several options were considered to mitigate the residual risk for the RTLS contingency considered. One option was to implement software patches in the Primary Avionics System Software (PASS) and in the Backup Flight Software (BFS) to accelerate the opening of the LH_2 inboard fill and drain valve (PV12), the LH_2 outboard fill and drain valve (PV11), and the LH_2 topping valve (PV13) 30 sec prior to closure of the ET umbilical doors at the transition to MM602. These valves are currently commanded open 80 sec after transition to MM602, 50 sec after the command to close the ET umbilical doors. The proposed software patches would inhibit a timer that now delays valve opening. Earlier valve opening would reduce the amount of hydrogen in the LH_2 manifold to less than 3 lbm; similar to the TAL contingency case. There was concern with installing and checking the software patches required to implement this option in time for STS-48 launch. A second option requires the crew to manually set the Main Propulsion System (MPS) dump switch on Panel R2 to "start" after MECO, thereby invoking the contingency RTLS dump procedure (Panel R2 is adjacent to the pilot's right leg). This action results in a 20-sec unpressurized hydrogen dump through the RTLS dump line and later through the fill and drain line at the MM602 transition. The amount of hydrogen remaining in the LH_2 manifold prior to ET umbilical door closure was estimated to be less than 3 lbm. Analysis of this option led to the determination that this action would result in inhibiting the automatic dump of oxygen in the Liquid Oxygen (LO_2) manifold, approximately 4,000 lbm oxygen. This raised the concern for adverse effects on vehicle weight and Center of Gravity (CG) distribution. To allow for dumping of oxygen, an additional step was proposed to require the crew to manually move the MPS dump switch to "GPC" (General Purpose Computer) between transition to MM602 and MM602 + 20 sec. This

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	Contamination and leakage of the LH_2 4-inch disconnect.	action returns LO_2 dump to normal software control. The proposed single-step and 2-step manual procedures were validated in the Shuttle Mission Simulator. STS-48/OV-103 weight and CG analysis determined that there would be no adverse effects on the vehicle if the 4,000 lbm oxygen remained in the LO_2 manifold during an RTLS contingency. For this reason, the 2-step procedure was deemed unnecessary for STS-48. The decision was made at the STS-48 Flight Readiness Review Action Item Closeout meeting to implement the single-step procedure in the event of an RTLS contingency, crew movement of the RTLS dump switch to "start" after MECO. This action was added to the pilot's flip checklist as the first entry after MECO. Additionally, a ground call would instruct the pilot to perform this procedure. An integrated simulation with the STS-48 crew and the Mission Control Center was performed prior to launch. Investigation into the 4-inch flapper seal failure mode(s) and corrective action(s) continues. Additionally, the decision was made to continue development and test of the software patches for implementation by STS-44/OV-103, the next scheduled Shuttle mission.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	Contamination and leakage of the LH_2 4-inch disconnect.	Analysis of potential adverse effects of allowing 3 lbm of hydrogen to leak into the aft compartment was recently undertaken as the result of the decision to require crew action for an RTLS contingency. This analysis determined that 3 lbm of hydrogen residuals would pose a flammability hazard for approximately 120 sec during reentry. At 56,000 feet (ft) altitude, residual hydrogen in the manifold would yield 10-11% hydrogen concentrations in the aft compartment. On ignition, this could produce a flammability hazard of 8,500 British Thermal Units (Btu) and a delta-pressure increase of 2 pounds per square inch (psi) in aft compartment pressure. The flammability hazard continues through 17,000 ft, where aft compartment hydrogen concentrations drop below 4%. At 17,000 ft, ignition can produce 29,000-Btu and 8.6-psi delta-pressure increase, resulting in a 5-psi overpressure in the aft compartment. For a TAL contingency, there is a similar flammability hazard. To mitigate the residual flammability risk, an additional procedural step was recommended to require the pilot to manually close the inboard fill and drain valve 20 sec after initiation of the dump. This step would result in closing the topping valve, isolating the 4-inch recirculation line, and isolating 90% of the 3 lbm residual hydrogen. A presentation, outlining the residual flammability hazard, was made to the Mission Management Team (MMT) and the Space Shuttle Program Director at the L-2 Day Review.

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	Contamination and leakage of the LH_2 4-inch disconnect.	Rationale for STS-48 flight was: • A large volume decay/leak test was performed on the STS-48/OV-103 4-inch flapper valve and provided confidence that the seal was functioning properly at that time. • A single-step procedure was implemented for an RTLS contingency to reduce the amount of potential hydrogen accumulation in the aft compartment and protect against failure of the LH_2 4-inch flapper valve seal. • A recommendation was made to add a second step to the single-step procedure to further reduce the identified flammability hazard. ***This anomaly/risk factor was acceptable for STS-48.***
6	Right-Hand (RH) outboard brake pressure was low. IFA No. STS-43-V-14 ***No brake pressure anomalies were reported on STS-48.***	Postflight data review indicated that the RH outboard brake pressure #1 was approximately 200 pounds per square inch absolute (psia) lower than RH outboard brake pressure #2. Maximum delta-pressure between brakes should be in a range from 1-psia high to 72-psia low. This brake pressure anomaly did not adversely affect the Orbiter rollout distance. A similar anomaly occurred on the last flight of OV-104, STS-37, when the RH outboard brake pressure was approximately 100-psia low. Post-STS-37 troubleshooting could not duplicate the problem. The OV-104

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	RH outboard brake pressure was low.	brake and anti-skid system was tested and found to be functioning within specification prior to STS-43. Carbon brakes were first installed on OV-104 prior to STS-37. Carbon brakes have sufficient energy capability to accommodate a 200-psia delta-pressure during the rollout braking phase of the mission. OV-103 had a similar problem during STS-41; out-of-specification brake pressure dispersions of approximately 200-240 psia were experienced on the 4 left brake pressures and the 4 right brake pressures. STS-41 was the second flight of OV-103 with carbon brakes. Troubleshooting isolated the problem to a faulty brake/skid servovalve (LV24). No problems were experienced during the previous OV-103 flight, STS-39. Heavy-braking tests performed on STS-39 completed the suite of carbon brake detailed test objectives for OV-103. Postflight ground troubleshooting on STS-43/OV-104 isolated the problem to a servovalve. The OV-104 brake servovalve module was removed and replaced, and will undergo failure analysis. Rationale for STS-48 flight was: • Carbon brakes have sufficient energy margin to accommodate delta-pressures in the range of 200 psia with no adverse effects. • There were no similar brake pressure problems on STS-39, the last OV-103 flight. ***This anomaly/risk factor was resolved for STS-48.***

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	STS-43/OV-104 Main Propulsion System (MPS) Liquid Oxygen (LO_2) bleed/check valve failure. IFA No. STS-43-V-16 HR No. INTG-023 {AR} ***No anomalies associated with the MPS LO_2 bleed/check valves were reported on STS-48.***	Review of STS-43/OV-104 entry MPS repressurization data found that the LO_2 bleed/check valve (CV35) was not checking system pressure. The bleed/check valve is used to provide LO_2 recirculation/bleed flow during tanking and prelaunch engine conditioning and provides Pogo recirculation flow during engine operation. The valve is designed with a 4-standard cubic feet per minute (scfm) backflow orifice to allow helium into the engine and Orbiter feedlines for purging and propellant system repressurization prior to entry. The bleed/check valves close after engine shutdown to isolate a failed engine from the LO_2 supply. Upon removal of the LO_2 bleed/check valve, it was discovered that both tangs at the end of the valve springs were missing. Visual inspection of the valve indicated wear on the tang/flapper, demonstrating that the valve was properly installed (the only other bleed/check valve failure in the program was attributed to improper installation). The valve was returned to the vendor, Parker-Hannifin, for analysis to determine the cause of the failure and to determine if the spring material was in accordance with design. The valve, including the spring, is LO_2-compatible by design. In addition to the LO_2-compatibility issue, there initially was concern relative to the effect the missing tangs would have on downstream Space Shuttle Main Engine (SSME) components. Analysis by the SSME Project determined that small, wire-size pieces of the spring could be ingested by the engine with no adverse consequence. The only potential for Crit 1 effects on the SSMEs is in the event that an SSME is prematurely shut down and sustains uncontained engine damage. Only then could Pogo return flow past the failed bleed/check valve from 1 of the remaining operational engines.

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
7 (Continued)	STS-43/OV-104 MPS LO_2 bleed/check valve failure.	Rationale for STS-48 flight was: • There were no adverse effects identified during a nominal mission; an SSME could ingest tang-size debris. • Two or more independent failures were required to get into a Crit 1 situation. ***This anomaly/risk factor was acceptable for STS-48.***

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1	Space Shuttle Main Engine (SSME) #3 controller Digital Computer Unit (DCU) failure on channel "A". IFA No. STS-43-E-01 HR No. INTG-165B {AR} ***No SSME anomalies were reported on STS-48.***	Engine #2028, controller F21, experienced a DCU channel "A" hold without a power loss prior to the start of Liquid Oxygen (LO_2) replenish. A successful switchover to channel "B" was performed at T–4.5 hours before the planned liftoff. A Failure Identification (FID) 001-000 was reported on the Vehicle Data Table (VDT), and a Major Component Failure (MCF) was reported on the engine status word. A memory dump of the faulty controller was performed, and analysis identified a parity error as the cause of the DCU channel "A" halt. The launch was scrubbed, and the discrepant controller was replaced with a spare (F18). The failed controller was sent to the vendor, Honeywell, for failure analysis. The parity error was isolated to hardware failure in the plated-wire memory bit sense line circuit, specifically a bit-6 sense line open circuit. Hardware disassembly revealed a broken "blind" lap solder joint connection of the bit jumper cable to the half-stack, which is not a generic design problem. The solder joint cannot be visually inspected after completion of the joint due to ribbon wire insulation remaining on the top half of the conductor. The defective solder joint was caused by improper reflow (insufficient heat) during build. The resultant high resistance produced the observed parity error and a DCU halt. The solder joint in question is 1 of 8,960 "blind" solder joints in each controller. This was the first solder joint failure that resulted in a controller-level failure, representing over 54,500 field hours and 24,443 factory hours of controller operation. There were 3 previous "blind" lap solder joint failures detected during subassembly build; however, all were identified prior to half-stack assembly into an engine controller.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1 (Continued)	SSME #3 controller DCU failure on channel "A".	A single-channel failure of the DCU results in the loss of redundancy and the controller posts an MCF. If the failure occurs before Solid Rocket Booster (SRB) ignition, the result is an on-pad abort. If the failure occurs before engine start command, the result is a launch scrub. A dual-channel failure during engine start or mainstage results in an engine pneumatic shutdown. There have been no DCU switchovers during engine start or mainstage in the program history. The calculated probability of a similar failure, based on the single controller level failure is: 1 in 330 flights for a launch delay, 1 in 9.8×10^6 flights for a pad abort, and 1 in 126,000 flights for loss of redundancy in flight. Since this issue was considered unique and understood, the problem did not pose a generic failure concern and was, therefore, eliminated as a flight constraint to the STS-43 mission. Rationale for STS-48 flight was: • This was the first "blind" solder joint failure in the field in SSME Program history. • Prelaunch, the Launch Commit Criteria protects against a launch with loss of DCU/controller redundancy. ***This anomaly/risk factor was resolved for STS-48.***

SECTION 6

STS-39 INFLIGHT ANOMALIES

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

Hazard Reports (HRs) 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.

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STS-39 INFLIGHT ANOMALIES

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Flash Evaporator System (FES) feedline "A", system #2, heater failure. IFA No. STS-39-V-01 HR No. INTG-164 {C} ORBI-276B {C} ***No FES anomalies were reported on STS-48.***	Prior to entering the tanking phase of the second launch countdown, FES feedline "A", system #2, heater failed. A review of data traces revealed a 20-24 ampere (amp) current spike for 5-10 milliseconds on main bus "B". This is an indication of a short to ground, causing the 10-amp line fuse to open and remove heater power. While the FES feedline heaters are Crit 2R3, there was concern that the short might 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 stepping on the feedline heater wire harness during repair of the secondary seal cavity pressure sensor caused the first launch attempt to be scrubbed. The 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 undertaken 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, which 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

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
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. Rationale for STS-48 flight was: • Troubleshooting isolated the anomaly to ALCA #2 and a wire harness between ALCA #2 and FES #2. ALCA #2 and the wire harness were replaced, and FES #2 heaters were retested successfully. ***This anomaly/risk factor was resolved for STS-48.***
2	Auxiliary Power Unit (APU) #2 Fuel Pump/Gas Generator Valve Module (FP/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 valve anomalies were reported on STS-48.***	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 cool the APU #2 FP/GGVM. Cooling system "A" spray valve LV25 had failed closed. FP/GGVM cooling is required in the contingency that an abort from orbit is required soon after APU shutdown. Without additional cooling, the FP/GGVM takes approximately 6 hours to cool sufficiently beyond the point of potential hydrazine detonation and subsequent fire or explosion in the APU, possibly causing loss of the vehicle and crew.

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	APU #2 FP/GGVM coolant system "A" valve did not open.	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 STS-48/OV-103 flight. Rationale for STS-48 flight was: • LV25 had exceeded the life-limit criteria for APU FP/GGVM cooling spray valves prior to launch. LV25 was replaced prior to STS-48. ***This anomaly/risk factor was resolved for STS-48.***
3	Reaction Control System (RCS) vernier thruster F5R fuel injector temperature biased low. IFA No. STS-39-V-03 HR No. ORBI-056 {C} ***No RCS vernier thruster anomalies were reported on STS-48.***	During firing of RCS vernier thruster F5R, the fuel injector temperature read 30-40°F lower than the oxidizer injector temperature. Both the oxidizer and fuel injector temperatures should not vary more than 10°F. Because there were no apparent thruster heater failures or leaks detected, it is believed that this anomaly was caused by a sensor or instrumentation error. Loss of a vernier thruster is a Crit 2/2 failure mode; loss of mission capability. There have been 2 previous vernier thruster oxidizer/fuel injector temperature anomalies similar to this occurrence: 1 on STS-3 and 1 on STS-4. Both anomalies were determined to be the result of poor sensor thermal conductivity that occurs in the vacuum of space. The corrective action taken to overcome the thermal

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)	RCS vernier thruster F5R fuel injector temperature biased low.	conductivity problem was to add thermal grease to the exterior of the sensor probe and the sensor injector well. It was determined that the sensor in thruster F5R was of the old configuration and did not have the added thermal grease. Postflight troubleshooting determined that this anomaly, as in the 2 previous cases, was caused by poor contact between the sensor and the thruster. There was not a true temperature degradation in thruster F5R on STS-39. Rationale for STS-48 flight was: • Loss of a vernier thruster is a Crit 2/2 failure. • This anomaly was an instrumentation error caused by poor sensor thermal conductivity. ***This anomaly/risk factor was acceptable for STS-48.***

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} ***No OPS recorder anomalies were reported on STS-48.***	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 flight day 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 Payload Forward 2 (PF2) MDM, Serial Number (S/N) 72, the MDM between the General Purpose Computers and OPS recorder #2, could be the cause of the anomaly. It was believed that the PF2 MDM 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. The PF2 MDM performs only Crit 3/3 functions.

C-2

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 the 2 failures was in 1985 and was with PF2 MDM, 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 hybrid circuit may have caused both the 1985 anomaly and the STS-39 anomaly. PF2 MDM, S/N 72, was removed from OV-103 at Kennedy Space Center and sent to the vendor for further failure analysis. There was no indication of a generic MDM problem. Rationale for STS-48 flight was: • The cause of this anomaly was determined to be the PF2 MDM, S/N 72. PF2 MDM, S/N 72, was removed from OV-103. • There was no generic MDM problem. ***This anomaly/risk factor was resolved for STS-48.***

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	Supply water dump nozzle temperature drop. IFA No. STS-39-V-08 ***During the 2 post-bakeout periods following supply dumps #4 and #5, the supply water line and nozzle temperatures indicated water leakage past the dump valve (IFA No. STS-48-V-04). The crew successfully purged the line of any free water and isolated the line by closing the dump line isolation valve. See Section 7, Orbiter 4, for more details.***	Approximately 20 minutes (min) into supply water dump #5, the water dump nozzle temperature rapidly decreased approximately 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. Troubleshooting at Kennedy Space Center was unsuccessful in reproducing this anomaly. This anomaly was considered to be caused by a transient effect. 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 waste water dump lines.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	Supply water dump nozzle temperature drop.	Rationale for STS-48 flight was: • Supply water dumping was not inhibited by the nozzle temperature anomalies on STS-39. There are alternate supply water dump methods available. • Troubleshooting was unsuccessful in reproducing this anomaly. On-orbit operation indicated that this was a transient problem. • Supply water dumping is a Crit 1R3 function. ***This anomaly/risk factor was acceptable for STS-48.***

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	Auxiliary Power Unit (APU) #2 lube oil outlet pressure was low. IFA No. STS-39-V-11 (STS-39-K-01) ***No APU anomalies were reported on STS-48.***	During entry, APU #2, Serial Number (S/N) 301, lube oil outlet pressure was low, reading 25 pounds per square inch absolute (psia); nominal outlet pressure is 40-50 psia. Additionally, the minimum delta-pressure (ΔP) across the pump was 20 psia during entry; nominal is 25-30 psia. The pump was certified to a ΔP of 25-30 psia and qualified to a minimum ΔP of 23 psia. During the pressure anomalies, APU #2 gearbox bearing temperatures #1 and #2 were within limits at 308°F maximum.Flight rules require APU shutdown if the lube oil outlet temperature or gearbox bearing temperatures exceed 425°F. Loss of lube oil flow can result in high gearbox temperatures, leading to APU shutdown and loss of critical APU function. This was the first APU lube oil outlet pressure-low anomaly in the program. Previous lube oil anomalies were related to high lube oil outlet pressures. Troubleshooting at Kennedy Space Center (KSC) determined that APU #2 was not properly serviced with lube oil prior to STS-39 launch, resulting in a low quantity of lube oil in APU #2. For this reason, APU #2, S/N 301, was not removed from OV-103 prior to STS-48. This Orbiter IFA was transferred to a KSC IFA (STS-39-K-01) for procedural corrective action. Rationale for STS-48 flight was: • This anomaly was the result of improper preflight lube oil servicing; servicing was properly performed on STS-48/OV-103. ***This anomaly/risk factor was resolved for STS-48.***

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 MLG anomalies were reported on STS-48.***	Postflight inspection of the MLG tires found that the RH outboard tire showed signs of significant wear. The outer 3 plies were excessively worn; 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 feet 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 Shuttle Landing Facility (SLF) runway at the Kennedy Space Center (KSC) may have contributed to the excessive tire wear. Previous experience with tire wear was limited to localized spin-up spots in the MLG tires; there was 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 has concluded that excessive tire wear was the result of environmental and crew performance dispersions. All MLG tires are replaced between flights.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)	RH outboard MLG tire excessive wear.	Rationale for STS-48 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. Observed wear conditions were well within the tire capability. • Excessive wear was the result of environmental and crew performance dispersions. ***This anomaly/risk factor was acceptable for STS-48.***
8	Loss of communications during entry. IFA No. STS-39-V-13 ***No similar anomaly was reported on STS-48.***	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 attitude of the Orbiter during the high-inclination entry and the relative position of the Tracking and Data Relay Satellite (TDRS) during that portion of the reentry; no hardware or software problems were found. Because of these conditions, the S-band antenna used for 2-way data and voice communications between the Orbiter and the ground was not in the required line-of-sight with the TDRS. Onboard navigation control would have provided sufficient data to the Commander and Pilot to achieve a safe landing if a hardware or software problem had caused the communications loss. Data dropouts were typical during Space Shuttle missions prior to the TDRS System availability and use.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)	Loss of communications during entry.	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-48 flight was: • STS-48 mission planning accounted for the potential for loss of communications during reentry. • Onboard navigation is sufficient to guide the Space Shuttle crew to a safe landing. ***This anomaly/risk factor was resolved for STS-48.***
9	The Pilot's Rotational Hand Controller (RHC) bottomed-out during entry. IFA No. STS-39-V-14 HR No. ORBI-152 {C} ***No RHC anomalies were reported on STS-48.***	During entry, the Pilot was in control of the vehicle until approximately 3 minutes prior to touchdown. At that time, the Pilot turned control of the vehicle over to the Commander, who completed the landing. When the Pilot took his hand off the RHC, it dropped into the slot and bottomed-out. The Pilot said that he tried to pull it back up and lock it in place, but was unable to lock it because the adjustment knob was jammed. Postlanding inspection found that the adjustment knob was in the full counterclockwise, or loose position. The Commander and Pilot RHCs are adjustable in the up and down direction, and 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

STS-39 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
9 (Continued)	The Pilot's RHC bottomed-out during entry.	adjustment, is a standard friction-type knob; turn it counterclockwise to loosen, clockwise to tighten. The Commander and Pilot nominally adjust their respective RHC to best fit their relative hand position prior to entry. The STS-39 Pilot would have been able to use his RHC, if required, even though it was not adjusted to the optimum height. Rationale for STS-48 flight was: • The positional adjustment anomaly would not impede the proper and accurate use of the RHC. ***This anomaly/risk factor was resolved for STS-48.***

SECTION 7

STS-48 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-48/OV-103 mission. Each anomaly is briefly described.

Hazard Reports (HRs) 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.

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STS-48 INFLIGHT ANOMALIES

STS-48 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Smoke detector false alarms. IFA No. STS-48-V-01 HR No. ORBI-259B {AR}	Smoke detector B in avionics bay #1 sounded 5 false alarms during prelaunch. At no time were the alarms associated with high smoke concentrations. The Circuit Breaker (CB) was pulled to eliminate nuisance alarms during launch and crew sleep periods. On Flight Day 3, another false alarm occurred, and the crew pulled the CB for the remainder of the flight to avoid further nuisance alarms. Redundant smoke detectors are available in avionics bays #1 and #3. Problem analysis indicated that the smoke detector electronics is sensitive to fan noise. Similar false alarms occurred during STS-38 and STS-32 missions.
2	External Tank (ET) umbilical door centerline latch drive motor #2 phase B current low. IFA No. STS-48-V-02 HR No. ORBI-302B {AR}	During ET umbilical door closing, the forward ET door centerline latch drive motor #2 failed to draw current on phase B. The centerline latch performed nominally with this failure. If one motor fails, the second motor is capable of releasing the centerline latch. The purpose of the centerline latches is to hold the ET doors open. These latches must be released before the doors can be closed. If the door cannot be closed, it would result in loss of vehicle/crew due to overheating during entry (Crit 1R2). Troubleshooting at Kennedy Space Center indicated the problem to be a failed relay in the aft Motor Control Assembly #2.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	ET umbilical door centerline latch drive motor #2 phase B current low.	Each ET door centerline latch has 2 redundant motors. The motors use separate power and control sources to retain redundancy. Loss of 1 phase of an ET centerline latch motor is a Crit 1R2 failure mode. If one motor loses 1 phase, and the second motor has all 3 phases, the ET latch actuator can operate properly. The hybrid relays, used in series to connect 3-phase alternating current power to each centerline latch motor, are also Crit 1R2. The door can also be closed with power from only 1 motor (3 good phases) in 12 seconds.
3	Fuel cell 1# Oxygen (O_2) reactant valve "closed" indication. IFA No. STS-48-V-03 HR No. ORBI-094 {AR}	The valve status remained "closed" when it should have changed to "open". Flow and pressure readings confirmed that the valve functioned as required. Absence of a corresponding decrease in O_2 flow or fuel cell coolant pressure verified that the valve was open. A reactant valve actually closing would have shut down the fuel cell. The indication later changed to "open" when an adjacent manifold valve was closed. It was suspected that the panel shook and caused a loose connection within the valve position switch. An open or break in the circuit gives the crew a "closed" indication. The main A and main B busses were tied together in case of a hard failure to prevent electrical equipment shutdown in the event of a fuel cell shutdown. The crew confirmed the onboard talkback also indicated "open". The main A and main B busses remained tied together until deorbit preparation. Postflight troubleshooting could not duplicate the problem after repeated cycling of the main C bus circuit breaker. The most probable cause of this anomaly was nonconductive contamination of the circuit breaker contacts.

STS-48 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4	Supply water dump valve leakage. IFA No. STS-48-V-04 HR No. ORBI-254 {C}	During the 2 post-bakeout periods following supply dumps #4 and #5, the supply water line and nozzle temperatures indicated water leakage past the dump valve. During the fifth supply water dump, an unusual heating cycle for the nozzles was noted. Nozzle heaters were left "on" following the dump in order to perform the bakeout. The nozzles appeared to respond nominally to the bakeout procedure. However, the data indicated sporadic leakage into the dump line through the dump valve; this is a very probable source for ice formation. A purge of the line was initiated to clear any water and prevent freezing of the line, dump valve, and dump nozzle. The crew successfully purged the line of any free water and isolated the line by closing the dump line isolation valve. If the nozzles were ruled failed, excess supply water could be dumped into the flash evaporator system, with the alternative of utilizing the wastewater dump line via the wastewater contingency crosstie (Flight Rule 9-919). Similar problems were experienced on STS-39/OV-103. Post-STS-39 troubleshooting could not reproduce the anomaly. An unused fuel cell end heater discreet measurement was spliced into the supply water nozzle heater power circuit for STS-48 to monitor heater circuit status. As indicated above, the problem recurred during STS-48. The discreet measurement demonstrated, however, that the nozzle heater was receiving power during the anomalies. Based on this information, the supply water dump nozzle assembly was replaced during the STS-42 flow.

STS-48 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	Hydraulic system #2 unloader valve leakage. IFA No. STS-48-V-05	The accumulator was recharged 4 times with circulation pumps and showed pressure decay as high as 369 pounds per square inch per hour (psi/hr) (specification limit = 48 psi/hr). The accumulator pressure increased during circulation pump operation, then dropped within an hour. Varying unloader valve leakage rates were observed over several circulation pump cycles. During Flight Control System (FCS) checkout, the unloader valve seat was flushed of contamination via hydraulic pressure supplied by the Auxiliary Power Unit. The decay rate after FCS checkout was reduced to 24 psi/hr, within the allowable limit of 48 psi/hr. The unloader valve is known to be sensitive to leakage caused by contamination. The problem was reproduced during ground turnaround testing and, the valve was removed and replaced.
6	Forward Reaction Control System (FRCS) manifold isolation valve operating on 2 of 3 phases. IFA No. STS-48-V-06 HR No. ORBI-127 {C} ORBI-244 {AR}	During entry/landing, the FRCS manifold #2 isolation valves (LV127, LV128) operated on only 2 alternating current phases (A and B). No current was shown on phase C for valve open and close. The valves functioned nominally when the isovalves were closed during entry. The same anomaly was observed when the valves were cycled postlanding. RCS isolation valves provide leak isolation for a failed-open thruster or a downstream leak. They are normally open throughout the mission.

STS-48 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)	FRCS manifold isolation valve operating on 2 of 3 phases.	The Circuit Breaker (CB) for phase C was cycled pre-ferry flight, and the valves operated on 3 phases upon return to Kennedy Space Center (KSC). The anomaly could not be duplicated at KSC. Contamination of the CB was considered to be the most likely cause of the problem due to previously seen failures of low-current Cbs. This condition can be cleared by cycling the CB when contamination is suspected. Manifold isolation valves can be operated on 2 of 3 phases. Tank isolation valves can be closed to prevent propellant loss in case of a leak.
7	Extraneous body flap motion. IFA No. STS-48-V-07 HR No. ORBI-025 {C}	During entry, at a relative velocity of 23,000 feet per second, extraneous body flap motion was observed. The cycling continued for about 1 minute. The motion was caused by OI-20 I-loads that were designed to provide thermal protection for the Space Shuttle Main Engine bells. The potential for this motion was understood before the flight. The motion is Center of Gravity (CG) dependent and is more noticeable at aft CGs. This condition could occur during entry. The body flap puts a small load on the Auxiliary Power Unit/hydraulic system.

STS-48 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1	Left-Hand (LH) frustum upper Right-Hand (RH) Booster Separation Motor (BSM) aeroheat shield cover missing. IFA No. STS-48-B-01 HR No. B-60-12 Rev. D-DCN9 {AR} C-00-04 Rev. B-DCN4 {C}	Postflight inspection of the STS-48 Solid Rocket Boosters found a LH frustum BSM aeroheat shield cover was missing. The upper RH hinge assembly and cover separated from the attach ring. These covers have been lost on 7 previous flights. All of these incidents were closed as reentry, water impact, or retrieval issues. Materials and processes investigation found the following: a scar on the BSM exit cone that indicates the cover opened normally; the attach ring failed in the closing direction which verifies the loss of the cover as a post-BSM firing event; the attach ring surfaces were free of aluminum which indicates that initial fracture occurred after BSM firing; and metallurgical analysis revealed no material defects. Analysis indicated that there are no ascent loads sufficient to cause loss of the cover. All possible failure modes during separation were examined and eliminated as a cause of the anomaly. Rebound energy from the cover hitting the frustum is insufficient to fracture the attach ring or fasteners. However, sufficient loads can be generated during Max Q reentry to break the attach ring and fasteners.
2	Right-Hand (RH) lower strut black mark. IFA No. STS-48-B-02 HR No. B-30-06 Rev. C-DCN4 {C}	During open assessment, a black mark was noted on the RH lower separation face. This condition was noted on an otherwise clean strut. The strut separation face is closed out with a Room-Temperature Vulcanizing (RTV)-133 environmental seal. Materials and processing analysis indicated that the black area had the appearance of black carbon soot, along with a brown background that appeared to be grease.

STS-48 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
2 (Continued)	RH lower strut black mark.	Carbon soot could not be positively identified. Chemical analysis indicated a strong presence of calcium and zinc, and secondary peaks of sulfur; these elements were roughly in proportion to those found in Conoco grease. The strut adjustment nut is installed with Conoco grease. Thermal analysis was performed of the heat flow in the area of the strut. The separation bolt showed no overall temperature increase. Worst-case initial temperature was 100°F; maximum qualified operating temperature is 120°F. The NASA Standard Initiator (NSI) cable showed a temperature rise of 10°F. Worst-case initial temperature is 100°F; maximum qualified operating temperature is 120°F. The NSI cable is qualified to 200°F. The maximum temperature at the separation face of the strut is 250°F at separation. The most probable scenario was as follows. Grease was inadvertently left on the strut during assembly. Grease should be cleaned from the RTV fillet area. The mating surfaces were contaminated with grease outboard from the black mark, and the RTV did not adhere. During temporary pressure reversal during ascent, soot flowed across the grease.

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
ACRIM II	Active Cavity Radiometer II
AFB	Air Force Base
ALCA	Aft Load Controller Assembly
AMOS	Air Force Maui Optical Site
amp	Ampere
AOA	Abort-Once-Around
APM	Ascent Particle Monitor
APU	Auxiliary Power Unit
AR	Accepted Risk
ATO	Abort-To-Orbit
ATP	Acceptance Test Program
BFS	Backup Flight Software
BITE	Built-In-Test Equipment
BSM	Booster Separation Motor
Btu	British Thermal Unit
C	Controlled
CA	California
CB	Circuit Breaker
CG	Center of Gravity
CLAES	Cryogenic Limb Array Etalon Spectrometer
CPU	Central Processing Unit
CREAM	Cosmic Radiation Effects and Activation Monitor
CRT	Cathode Ray Tube
DAR	Deviation Approval Request
DC	Direct Current
DCU	Digital Computer Unit
DEU	Display Electronics Unit
DOLILU	Day-Of-Launch I-Load Update
DR	Discrepancy Report

APPENDIX A

LIST OF ACRONYMS - CONTINUED

EAFB	Edwards Air Force Base
ECP	Engineering Change Proposal
EDM	Electrical Discharge Machining
EDT	Eastern Daylight Time
EIU	Engine Interface Unit
EMDM	Enhanced Multiplexer-Demultiplexer
EOM	End-Of-Mission
ESC	Electronic Still Camera
ET	External Tank
FASCOS	Flight Acceleration Safety Cutoff System
FCS	Flight Control System
FD	Flight Day
FES	Flash Evaporator System
FID	Failure Identification
FMEA	Failure Modes and Effects Analysis
FMEA/CIL	Failure Modes and Effects Analysis/Critical Items List
FOS	Factor of Safety
FP	Fuel Pump
FPOV	Fuel Preburner Oxidizer Valve
FR	Flight Rule
FRCS	Forward Reaction Control System
FRR	Flight Readiness Review
FRT	Flight Readiness Testing
ft	Feet
GG	Gas Generator
GGVM	Gas Generator Valve Module
GHe	Gaseous Helium
GN_2	Gaseous Nitrogen
GPC	General Purpose Computer
H_2	Hydrogen
HALOE	Halogen Occultation Experiment
HCF	High-Cycle Fatigue
HPFTP	High-Pressure Fuel Turbopump
HPM	High Performance Motor
HPOTP	High-Pressure Oxidizer Turbopump
HR	Hazard Report
hr	Hour

APPENDIX A

LIST OF ACRONYMS - CONTINUED

HRDI	High Resolution Doppler Imager
HSD	Horizontal Situation Display
IBM	International Business Machines
IFA	Inflight Anomaly
IM	Instrument Module
INSRP	Interagency Nuclear Safety Review Panel
INTG	Integration
IOM	Input/Output Module
IPMP	Investigation into Polymer Membranes Processing
IPV	Inter-Process Variable
ISAMS	Improved Stratospheric and Mesospheric Sounder
IV&V	Independent Verification and Validation
JSC	Johnson Space Center
KSC	Kennedy Space Center
lb	Pound
lbm	Pounds-Mass
LCC	Launch Commit Criteria
LD	Leak Detector
LH	Left-Hand
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LPFTP	Low-Pressure Fuel Turbopump
LPOTP	Low-Pressure Oxidizer Turbopump
LSFR	Launch Site Flow Review
L-2	Launch Minus 2 Day
MACS	Modular Attitude Control System
MAF	Michoud Assembly Facility
MCC	Main Combustion Chamber
MCDS	Multifunctional CRT Display System
MCF	Major Component Failure
MCIU	Manipulator Controller Interface Unit
MDM	Multiplexer-Demultiplexer
ME-2	Main Engine 2
MEC	Master Events Controller
MECO	Main Engine Cutoff

APPENDIX A

LIST OF ACRONYMS - CONTINUED

MFV	Main Fuel Valve
MIA	Multiplexer Interface Adapter
MIM	MCDS Input Module
min	Minute
MLG	Main Landing Gear
MLP	Mobile Launch Platform
MLS	Microwave Limb Sounder
MM602	Major Mode 602
MMS	Multi-mission Modular Spacecraft
MMT	Mission Management Team
MODE-0	Middeck 0-Gravity Dynamics Experiment
MOV	Main Oxidizer Valve
MOVA	Main Oxidizer Valve Actuator
MPS	Main Propulsion System
MPTA	Main Propulsion Test Article
MSE	Mission Safety Evaluation
MUM	MCDS Update Module
NASA	National Aeronautics and Space Administration
NBR	Nitrite Butadiene Rubber
NLG	Nose Landing Gear
nm	Nanometer
NSI	NASA Standard Initiator
NSRS	NASA Safety Reporting System
O_2	Oxygen
OA-1	Operational-Instrumentation Aft #1
OMRSD	Operational Maintenance Requirements and Specifications Document
OPOV	Oxidizer Preburner Oxidizer Valve
OPS	Operational Sequence Operations
ORBI	Orbiter
OSMQ	Office of Safety and Mission Quality
OV	Orbiter Vehicle
P/N	Part Number
PAR	Prelaunch Assessment Review
PARE	Physiological and Anatomical Rodent Experiment
PASS	Primary Avionics System Software
PCG	Protein Crystal Growth

APPENDIX A

LIST OF ACRONYMS - CONTINUED

PCV	Pulse Control Valve
PEM	Particle Environment Monitor
PF2	Payload Forward 2
POR	Power-On Reset
ppm	Parts Per Million
PR	Problem Reports
PRCB	Program Requirements Control Board
PRSD	Power Reactant Supply and Distribution
psi	Pounds Per Square Inch
psi/hr	Pounds Per Square Inch Per Hour
psia	Pounds Per Square Inch Absolute
psig	Pounds Per Square Inch Gage
PSIG	Propulsion Systems Integration Group
RCS	Reaction Control System
RDVT	Rotary Variable Differential Transducer
RH	Right-Hand
RHC	Rotational Hand Controller
RME	Radiation Monitoring Equipment
RMS	Remote Manipulator System
RPC	Remote Power Controller
RSRM	Redesigned Solid Rocket Motor
RTLS	Return to Launch Site
RTV	Room-Temperature Vulcanizing
S/N	Serial Number
SAM	Shuttle Activation Monitor
scch	Standard Cubic Centimeters Per Hour
sccs	Standard Cubic Centimeters Per Second
scfm	Standard Cubic Feet Per Minute
scim	Standard Cubic Inches Per Minute
SCU	Sequence Control Unit
sec	Second
SLF	Shuttle Landing Facility
SMS	Shuttle Mission Simulator
SOLSTICE	Solar/Stellar Irradiance Comparison Experiment
SOV	Shutoff Valve
SR&QA	Safety, Reliability and Quality Assurance
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor

APPENDIX A

LIST OF ACRONYMS - CONTINUED

SSC	Stennis Space Center
SSME	Space Shuttle Main Engine
SSRP	System Safety Review Panel
SUSIM	Solar Ultraviolet Spectral Irradiance Monitor
TAEM	Terminal Area Energy Management
TAL	Transatlantic Abort Landing
	Transoceanic Abort Landing
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TEM-8	Test Evaluation Motor 8
UARS	Upper Atmosphere Research Satellite
VDT	Vehicle Data Table
WINDII	Wind Imaging Interferometer
WSB	Water Spray Boiler