MISSION SAFETY EVALUATION REPORT FOR STS-43

Postflight Edition: October 31, 1991

Safety Division

Office of Safety and Mission Quality

National Aeronautics and Space Administration

Washington, DC 20546

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EXECUTIVE SUMMARY

STS-43 launch, originally scheduled for July 24, 1991, was initially delayed 24 hours due to a problem in the electrical circuit controlling Orbiter/External Tank (ET) separation. A further launch postponement occurred the next day when channel "A" on the Main Engine Controller (MEC) for Space Shuttle Main Engine (SSME) #3 went to halt during the countdown. Data indicated that a hardware failure of the MEC had occurred; both channels "A" and "B" must be operational prior to launch. The malfunctioning MEC was changed out and retested, and STS-43 launch was rescheduled for August 1, 1991. After an additional 1-day delay due to rain and thunderstorm activity in the Kennedy Space Center (KSC) launch area, STS-43/*Atlantis* was launched from KSC Launch Complex 39A at 11:02 Eastern Daylight Time (EDT) on August 2, 1991.

The STS-43 flight included several new or recent modifications to improve performance and safety. This was the first mission flown with the OI-20 flight software designed for use with the new AP-101S General Purpose Computer (GPC); OI-20 provides enhancements that improve overall safety by adding operational capability, monitoring capability, and automation in time-critical sequences. Cadmium plating was eliminated on the Solid Rocket Motor (SRM) igniter inner and outer gasket retainers; this eliminated the potential for cadmium-induced embrittlement of the hardware. STS-43 was the first flight of the new tire pressure transducer that allows the ground to more accurately determine the tire pressure leak rate. The STS-43 Gaseous Oxygen (GOX) Flow Control Valves (FCVs) were shimmed to a fixed 78% orifice. This is the same configuration as STS-40, the first flight with this fixed 78% orifice that will become the final configuration for the Orbiter fleet. Performance on STS-43 was similar to the satisfactory performance experienced on STS-40.

Atlantis carried the fourth Tracking and Data Relay Satellite (TDRS-E) into orbit to update NASA's Tracking and Data Relay Satellite System (TDRSS), resulting in 2 operating satellites plus a complement of 2 spares in this space network. Following TDRS deployment, *Atlantis* moved to a safe separation distance prior to the Inertial Upper Stage (IUS) SRM-1 burn that placed the TDRS-E satellite into a transfer orbit. All IUS-sequenced events associated with the transfer orbit occurred on time, and performance was nominal. The IUS SRM-1 and SRM-2 burns, coupled with their respective follow-on Reaction Control System (RCS) burns, were well within pre-mission predicted limits. As a result, the IUS successfully placed TDRS-E in a 19,335.5-Nautical Mile (NM) by 19312.9-NM geosynchronous orbit above the Pacific Ocean southwest of Hawaii.

Along with the TDRS-E/IUS in *Atlantis'* cargo bay for STS-43 were the following payloads: the Shuttle Solar Backscatter Ultraviolet (SSBUV) experiment, the Space Station Heat Pipe Advanced Radiator Element-II (SHARE-II), the Optical Communications Through the Shuttle Window (OCTW) flight demonstration, and the Tank Pressure Control Experiment (TPCE). A number of other experiments were carried in *Atlantis'* middeck.

All systems and payloads operations were generally satisfactory during the 9-day mission. *Atlantis* landed at the KSC Shuttle Landing Facility (SLF) at 8:23 a.m. EDT for a total mission elapsed time of 8 days, 21 hours, 21 minutes, and 25 seconds. **This was the first planned landing at KSC's SLF since return-to-flight.** However, 2 other Shuttle missions landed at KSC since return to flight; both were diverted to KSC due to bad weather at Edwards Air Force Base (EAFB), California.

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-43 flight were also documented prior to the STS-43 Flight Readiness Review (FRR) (FRR Edition) and the STS-43 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-43 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-43 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-43 launch, that were impacted or repeated by anomalies reported for the STS-43 flight.

Section 4 - Contains a list of safety risk factors that were considered resolved for STS-43.

Section 5 - Contains a list of Inflight Anomalies (IFAs) that developed during the STS-40 mission, the previous Space Shuttle flight.

Section 6 - Contains a list of IFAs that developed during the STS-37 mission, the previous flight of the Orbiter Vehicle (OV-104).

Section 7 - Contains a list of IFAs that developed during the STS-43 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-43 MISSION SUMMARY

2.1 Summary Description of the STS-43 Mission

STS-43 launch, originally scheduled for July 24, 1991, was initially delayed 24 hours due to a problem in the electrical circuit controlling Orbiter/External Tank (ET) separation. A further launch postponement occurred the next day when channel "A" on the Main Engine Controller (MEC) for Space Shuttle Main Engine (SSME) #3 went to halt during the countdown. Data indicated that a hardware failure of the MEC had occurred; both channels "A" and "B" must be operational prior to launch. The malfunctioning MEC was changed out and retested, and STS-43 launch was rescheduled for August 1, 1991. After an additional 1-day delay due to rain and thunderstorm activity in the Kennedy Space Center (KSC) launch area, STS-43/*Atlantis* was launched from KSC launch complex 39A at 11:02 Eastern Daylight Time (EDT) on August 2, 1991.

The STS-43 flight included several new or recent modifications to improve performance and safety. This was the first mission flown with the OI-20 flight software designed for use with the new AP-101S General Purpose Computer (GPC); OI-20 provides enhancements that improve overall safety by adding operational capability, monitoring capability, and automation in time-critical sequences. Cadmium plating was eliminated on the Solid Rocket Motor (SRM) igniter inner and outer gasket retainers; this eliminated the potential for cadmium-induced embrittlement of the hardware. STS-43 was the first flight of the new tire pressure transducer that allows the ground to more accurately determine the tire pressure leak rate. The STS-43 Gaseous Oxygen (GOX) Flow Control Valves (FCVs) were shimmed to a fixed 78% orifice. This is the same configuration as STS-40, the first flight with this fixed 78% orifice that will become the final configuration for the Orbiter fleet. Performance on STS-43 was similar to the satisfactory performance experienced on STS-40.

During ascent, Water Spray Boiler (WSB) #2 failed to provide cooling to the Auxiliary Power Unit (APU) lube oil. The crew configured the WSB controller from system A to B with the same results. The APU #2 gearbox forward bearing temperature reached 351°F; Flight Rule (FR) 10-14 was invoked, and APU #2 was shut down directly after Main Engine Cutoff (MECO). The APU gearbox is certified for 400°F. Data indicated that freezing occurred on the water spray bar. WSB undercooling occurred on STS-35/OV-102, STS-37/OV-104, and STS-38/OV-104. FR 10-23 indicates that an APU without WSB cooling has about 11 minutes of allowable run-time. In this case, the affected APU/hydraulic system activation was delayed for entry until Terminal Area Energy Management (TAEM). FR 10-85 states that for loss of a single WSB, the flight will continue to a nominal End-Of-Mission (EOM) provided there is no subsequent loss of redundancy on one of the remaining APU/hydraulic/WSB systems; if redundancy is lost on a remaining system, Minimum Duration Flight (MDF) is invoked. On Flight Day (FD) 1 indications were that WSB #2, that froze up during ascent, had now thawed. No more action was planned until Flight Control System (FCS) checkout, at which time APU #2 was to be used.

Atlantis carried the fourth Tracking and Data Relay Satellite (TDRS-E) into orbit to update NASA's Tracking and Data Relay Satellite System (TDRSS), resulting in 2 operating satellites plus a complement of 2 spares in this space network. The TDRS-E satellite/Inertial Upper Stage (IUS) booster was deployed from the Shuttle's payload bay approximately 6 hours and 12 minutes after launch. Following TDRS deployment, *Atlantis* moved to a safe separation distance prior to the IUS SRM-1 burn that placed the TDRS-E satellite into a transfer orbit. All IUS-sequenced events associated with the transfer orbit occurred on time, and performance was nominal. The IUS SRM-1 and SRM-2 burns, coupled with their respective follow-on Reaction Control System (RCS) burns, were well within pre-mission predicted limits. As a result, the IUS successfully placed TDRS-E in a 19,335.5 -Nautical Mile (NM) by 19312.9-NM geosynchronous orbit above the Pacific Ocean southwest of Hawaii.

Along with the TDRS-E/IUS in *Atlantis'* cargo bay for STS-43 were the following payloads: the Shuttle Solar Backscatter Ultraviolet (SSBUV) experiment, mounted in 2 Get-Away-Special (GAS) containers, which was used to aid in calibrating ultraviolet satellites already in orbit that assist in measuring the Earth's ozone layer and provide solar irradiance data; the Space Station Heat Pipe Advanced Radiator Element-II (SHARE-II), a modification of the SHARE-II payload flown on STS-29, used to demonstrate and quantify the microgravity thermal vacuum performance of a high-capacity, space-constructible, heat pipe radiator element with no moving parts that may be used to cool Space Station Freedom; the Optical Communications Through the Window (OCTW) flight demonstration that uses fiber optics for data communications onboard the Shuttle; and the Tank Pressure Control Experiment (TPCE), contained in GAS hardware, to study the effects of microgravity on the thermal stratification of fluids and to determine the effectiveness of jet-induced mixing for controlling tank pressure.

In *Atlantis'* middeck was the Auroral Photography Experiment-B (APE-B), an Air Force-sponsored experiment to study the Earth's auroras (Northern and Southern Lights); the Bioserve Instrumentation Technology Associates Materials Dispersion Apparatus (BIMDA), consisting of apparatus for conducting experiments in protein crystal growth, zeolite crystal formation, collagen and virus assembly, interferon induction, seed germination, cell fixation, and fluid sciences/diffusion; the Investigations into Polymer Membrane Processing (IPMP) experiment, a test of manufacturing polymers in orbit; the Protein Crystal Growth-III (PCG-III) experiment, a device used to grow crystals in microgravity; the Space Acceleration Measurement System (SAMS), a device used to measure accelerations and disturbances to weightlessness during *Atlantis'* stay in orbit; and the Solid Surface Combustion Experiment (SSCE), a test of the way materials burn in weightlessness.

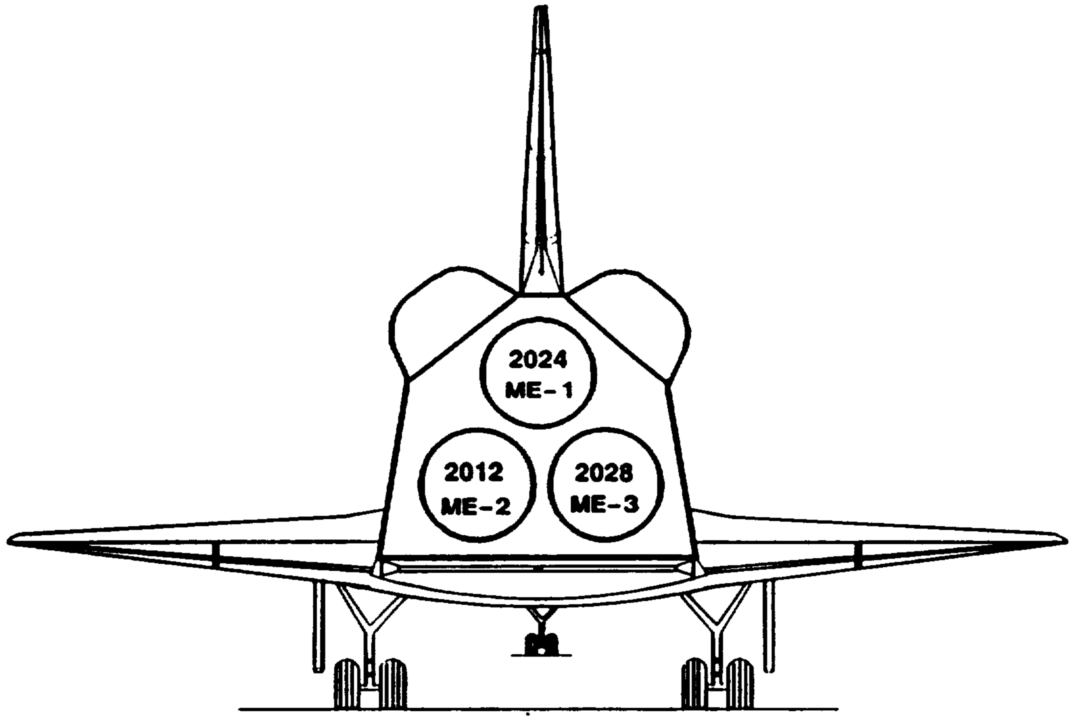
Operation of all systems was generally satisfactory during the 9-day mission (the inflight anomalies encountered are addressed in Section 7). FCS checkout was performed on FD 8 using APU #2, and the failure of WSB #2 was confirmed. On FD 9, as planned, the affected APU was started at the TAEM interface (Mach 2.5) and shut down at wheel stop. The remaining 2 APUs were run in their nominal configuration, speed normal and auto shutdown enabled. APU #1 was started at Time of Ignition (TIG) – 5 minutes, and APU #3 was started at Entry Interface (EI) – 13 minutes. Approximately 11 RCS thrusters exhibited lower than normal temperatures. This was thought to have been caused by moisture trapped in the thrusters from the rains at KSC prior to launch.

The mission's final experiments included a good test of new digital autopilot software which will reduce the amount of thruster fuel used and the amount of jarring caused by thruster firings. The crew executed 22 tests of this new system, which is being considered for flight on the next Shuttle mission. Performance of both SHARE-II heat pipes closely matched preflight predictions during both the bay-to-Earth and tail-to-Sun data takes. A final test was also performed of the prototype space station cooling radiators in which the two different design radiators were deliberately reprimed. As expected, the graded grove design heat pipe was able to reprime itself, while the monogrove design did not exhibit this capability.

Atlantis landed satisfactorily at the KSC Shuttle Landing Facility (SLF) at 8:23 a.m. EDT for a total mission elapsed time of 8 days, 21 hours, 21 minutes, and 25 seconds. This was the first planned landing at KSC's SLF since return to flight. Two other Shuttle missions landed at KSC since return to flight; both were diverted to Kennedy due to bad weather at Edwards Air Force Base (EAFB), California.

2.2 Flight/Vehicle Data

- Launch Date: August 2, 1991
- Launch Time: 11:02 a.m. EDT
- Launch Site: KSC Pad 39A
- RTLS: Kennedy Space Center, Shuttle Landing Facility
- TAL Site: Banjul, The Gambia
- Alternate TAL Sites: Ben Guerir, Morocco; Moron, Spain
- Landing Site: Kennedy Space Center, Runway 15
- Landing Date: August 11, 1991
- Landing Time: 8:23 a.m. EDT
- Mission Duration: 8 Days, 21 Hours, 21 Minutes
- Crew Size: 5
- Inclination: 28.45°
- Orbit: 160 x 160 Nautical Miles/Direct Insertion
- Orbiter: OV-104 (9) *Atlantis*
- ET-47
- SRBs: BI-045
- RSRM Flight Set #17
- MLP #1



ENGINE	#2024	#2012	#2028
POWERHEAD	#2026	#2025	#4005
MCC*	#2013	#2020	#2018
NOZZLE	#4006	#4002	#4012
CONTROLLER	F5	F16	F18
FASCOS*	#01	#09	#21
HPFTP*	#6102R3	#4007R3	#6009
LPFTP*	#2131	#4006	#2228
HPOTP*	#4009R3	#2425R1	#2405R1
LPOTP*	#2028	#2213	#2027R1

* Acronyms can be found in Appendix A.

2.3 Miscellaneous Items of Interest for the STS-43 Mission.

- First Use of OI-20 Flight Software. STS-43 is the first mission using the OI-20 flight software. OI-20 was designed to be used with the new General Purpose Computers (GPCs), AP-101S. There are several enhancements provided in OI-20 that improve overall safety by adding operational capability, monitoring capability, and automation in time-critical sequences. Following is a list of some of the OI-20 safety enhancements:

  - Provides the Orbiter crew visibility concerning the status of the Flight Control System actuators, Space Shuttle Main Engine (SSME) actuators, and Line Replaceable Unit (LRU) selection/deselection status during Operations Sequence (OPS) #1 (ascent) and OPS #6 (abort). Previously, only the Mission Control Center had this visibility and would have to make a call to the crew.

  - Provides the capability to allow the Orbiter crew to select "normal" or "alternate" body flap schedules and body bending filters through an item entry. This will reduce the potential for SSME heating and expand the contingency abort envelope.

  - Automatically transitions primary and secondary runway and TACAN selection when transitioning from OPS #1 (ascent) to OPS #3 (entry) during a Transoceanic Abort Landing (TAL) contingency. Currently, the Orbiter crew must reselect the secondary runway and TACAN after OPS #1 to OPS #3 transition. Failure to accomplish this transition could result in the loss of vehicle and crew.

  - Provides for automatic External Tank (ET) umbilical door closure on entry in OPS #6 (abort) during a TAL contingency. This change eliminates the requirement for the Orbiter crew to manually close the ET umbilical doors during a time-critical abort situation.

  - Adds low thrust-to-weight guidance constraint that allows the thrust vector to be pointed vertical, but not retrograde. This allows maximization of the vehicle's ability to climb out to a safe altitude.

  - Adds a parameter to improve Return-To-Launch-Site (RTLS) monitoring by the Orbiter crew and to provide additional information needed to determine if a downmode to a contingency procedure is required.

  - Provides adaptive guidance enhancements to reduce engine-out transients with Solid Rocket Booster (SRB) dispersions.

- Adds closed-loop Reaction Control System (RCS) pulse-mode to limit duration/frequency of thruster firings to support Remote Manipulation System (RMS)/payload operations.

- Adds smart body flap to proportion control between elevons and body flaps and eliminated need for the crew to interact with elevon scheduling.

- Provides desensitization of the Rotational-Hand Controller (RHC), resulting in more vernier control during slapdown maneuver. Current pitch RHC gain causes some overcontrol during the slapdown maneuver.

- Provides two-fault tolerance in OPS #1 (ascent) and OPS #6 (reentry) for elevon feedback, speedbrake feedback, rudder feedback, RHC, rudder pedal transducer assembly, and the Microwave Scanning Beam Landing System (MSBLS).

- Provides overlay capability in upper GPC memory for stowage and quick retrieval of OPS #3 during OPS #2 (on-orbit) operations. This capability eliminates the need to "freeze dry" OPS #3 in a GPC during on-orbit operations.

- Solid Rocket Motor (SRM) Igniter Gasket Retainer Modification. Blow holes in the igniter joint putty have been experienced on previous flights. One of the effects observed has been erosion of cadmium plating on the SRM igniter retainer gaskets. A redesign effort includes changing the gasket retainer material from cadmium-plated steel to stainless steel. [See STS-35 Mission Safety Evaluation (MSE) Report, Postflight Edition, March 15, 1991, Section 4, SRM 3 for more details.]

  Since the previous SRM set, cadmium plating has been eliminated on the SRM igniter inner and outer gasket retainers [Engineering Change Proposal (ECP) SRM-2418]. This will eliminate the potential for cadmium-induced embrittlement of the hardware. Removal of the cadmium does not affect form, fit, or function. The corrosion preventative grease is the same as currently used (reference TRW-60448, Certification Report for 4130 Steel Gaskets). The new gaskets will be inspected using the existing inspection criteria.

- New Tire Pressure Transducer. STS-34/OV-104 is the first flight of the new tire pressure transducer. This new transducer will allow the ground to more accurately determine the tire pressure leak rate.

- Gaseous Oxygen (GOX) Fixed Orifice Pressurization System. The STS-43 GOX Flow Control Valves (FCVs) are shimmed to a fixed 78% orifice. This is the same configuration as STS-40, the first flight with fixed 78% GOX FCV orifices. Fixed orifice pressurization performance on STS-40 was excellent. Preflight predictions for STS-43 indicate that performance should be similar to STS-40. A fixed 78% GOX orifice is expected to be the final configuration for the Orbiter fleet.

- Payload Return. Payload return will be permitted even though the vehicle touchdown weight exceeds the 230,000-pound limit. The deorbit opportunity will be waved off if prelanding analysis forecasts a violation of 1.85 g for all heading alignment cones for all the runways at the landing site.

2.4 Payload Data

Atlantis will put NASA's fourth Tracking and Data Relay Satellite (TDRS-E) into orbit on Space Shuttle mission STS-43 to update the Tracking and Data Relay Satellite System (TDRSS).

TDRS-E will be deployed from Atlantis about 6 hours after launch and will be boosted to a geosynchronous orbit by an attached Inertial Upper Stage (IUS). TDRS-E will be positioned to remain stationary 22,400 miles above the Pacific Ocean southwest of Hawaii.

The TDRSS, in operation since the eighth Space Shuttle flight, provides almost uninterrupted communications with Earth-orbiting shuttles and satellites and has replaced the intermittent coverage provided by globe-encircling ground tracking stations used during the early space program. The TDRSS will replace the NASA ground-based network of tracking and communication stations located around the world.

Payload Bay:

- TDRS-E will relay signals and data to and from the Shuttle Orbiter, unmanned low-Earth orbiting spacecraft, and Earth-based users without processing. TDRS-E is composed of three distinct modules: a spacecraft module, a payload module and an antenna module. The spacecraft module houses the subsystems that operate the satellite. The payload module is composed of the electronic equipment required to provide communications between the user spacecraft and the ground.

  The antenna module is composed of seven antenna systems: two single access, the multiple access array and space-to-ground link, S-band omni for satellite health and housekeeping, and commercial K-band and C-band antennas. Single Access Antenna (SAA) deployment failure modes were reviewed based on investigation of the Galileo high-gain antenna failure to deploy fully. TRW inspected both SAA deployment mechanisms on TDRS-E; a spoke was found to be out-of-place. A procedure was then generated and exercised on TDRS-F to assure that the spoke could be safely repositioned; the SAA spoke was successfully repositioned and reinspected. The Space to Ground Line (SGL) antenna drive was also reviewed based on the Galileo investigation. Analyses were redone, using bounding conservative assumptions based on stiffness tests, that indicated adequate torque margins for the TDRS-E SGL antenna drive assembly.

- IUS-15, the vehicle to be used on STS-43 to boost TDRS-E into a geosynchronous orbit, is a two-stage rocket weighing approximately 32,500 pounds (lb). Each stage has a solid rocket motor. The IUS is 5.18 meters (17 feet) long and 2.8 meters (9.25 feet) in diameter. It consists of an aft skirt; an aft stage solid rocket motor containing 21,400 lb of propellant generating approximately 42,000 lb of thrust; an interstage; a forward stage solid rocket motor with 6,000 lb of propellant generating approximately 18,000 lb of thrust; and an Airborne Support Equipment (ASE). The reaction control system controls the IUS/TDRS attitude during coasting; roll control during SRM thrusting; and velocity impulses for accurate orbit injection. IUS-15 will carry two reaction control fuel tanks, each containing 120 lb of hydrazine. The ASE is the mechanical, avionics, and structural equipment located in the Orbiter that supports the IUS and the TDRS-E in the Orbiter payload bay and elevates the IUS/TRDS for final checkout and deployment from the Orbiter.

- Shuttle Solar Backscatter Ultraviolet (SSBUV) Experiment -- The SSBUV payload is mounted in 2 Get-Away-Special (GAS) containers. This experiment will check the calibration of the ozone measuring instruments aboard the National Oceanic and Atmospheric Administration (NOAA)-9 and NOAA-11 satellites and the NASA Nimbus-7 satellite to verify the accuracy of the data set of atmospheric ozone and solar irradiance data. The SSBUV measurements will be compared with nearly coincident measurements taken by Solar Backscatter Ultraviolet (SBUV) instruments on these satellites to help scientists solve the problem of data repeatability caused by calibration drift of free-flying SBUV instruments.

- Space Station Heat Pipe Advanced Radiator Element-II (SHARE-II) -- is a modification of the SHARE-II payload that was flown on STS-29. SHARE-II will demonstrate and quantify the microgravity thermal vacuum performance of a high-capacity, space-constructible, heat pipe radiator element for heat rejection as a prelude to development of a Space Station heat rejection system. The two major components of SHARE-II are: the Heat Pipe

Radiator System (HPRS) and the Heat Pipe Instrumentation and Control System (HPICS). The HPRS consists of two 22-foot long heat pipes. The pipes are divided into an evaporator section and a condenser section and are filled with ammonia.

- Optical Communication Through the Shuttle Window (OCTW) Flight Demonstration — is a demonstration system to test the feasibility of using fiber optics to transmit data between the Orbiter crew cabin and the payload bay through an aft Shuttle window. The system consists of a payload bay box and a crew cabin box, each containing a video transmitter/receiver and a digital transmitter/receiver, and two optical couplers. The couplers transmit the fiber optic signal through the window. The video and digital link will be simulated with test signals inside the cabin. The signals will be transmitted via the fiber optic link to the payload bay. In the payload bay, the signals will be received and retransmitted to the crew cabin where they will be analyzed to determine system performance.

- Tank Pressure Control Experiment (TPCE) — is a study to determine the effects of microgravity on the thermal stratification of fluids and to determine the effectiveness of jet-induced mixing for controlling tank pressure. The experiment is contained in GAS hardware, and the working fluid is Freon 113.

Middeck:

- Solid Surface Combustion Experiment (SSCE) — will measure flame spread rate, solid-phase temperature, and gas-phase temperature for flames spreading over a rectangular fuel bed in microgravity to research improvements for fire safety in space travel.

- BioServe Instrumentation Technology Associates (ITA) Materials Dispersion Apparatus (BIMDA) — consists of Materials Dispersion Apparatus (MDA) minilabs and their controller and the Refrigerator/Incubator Module (R/IM) carrier which houses the entire BIMDA hardware. The MDA is a compact mixing device capable of mixing up to 100 samples of any 2 or 3 fluids using the liquid-to-liquid diffusion process. The 4 MDA units are expected to yield over 200 separate data points from experiments conducted in protein crystal growth, zeolite crystal formation, collagen and virus assembly, interferon induction, seed germination, cell fixation, and fluid sciences/diffusion experiments. Another primary element of the BIMDA payload is the bioprocessing testbed that will be used to mix cells with various activation fluids followed by extended periods of metabolic activity and subsequent sampling to determine the response of live cells to various hormones and stimulating agents under microgravity conditions.

Two of the 4 BIMDA MDAs contain fluids which have a relative irritancy rating: moderately high, high, or very high. These fluids were analyzed by a JSC toxicologist. Contact of these fluids with the crewmembers eyes could cause serious injury. This constitutes a catastrophic hazard. Three levels of containment are required to control a potentially catastrophic hazard which results from the release of toxic chemicals. The 2 MDAs containing these fluids do not meet this requirement for triple containment; they have only 2 levels of containment. Review of this problem resulted in agreement that the 2 levels of containment provided will result in satisfactory safety for the crew. A waiver of the triple containment requirement was approved by the Level II Program Requirements Control Board (PRCB) on July 12, 1991.

- Space Acceleration Measurement System (SAMS) — is used to record very low-level accelerations in 3 axes, at up to 3 locations in the middeck, to measure the amount of disturbance to the weightless environment onboard throughout the flight and during specific events including orbital maneuvering system and reaction control system firings.

- Protein Crystal Growth III (PCG III) — is a continuing series of experiments leading toward major benefits in biomedical technology. The experiments on this mission could improve pharmaceutical agents like insulin. The hardware consists of the Protein Crystal Facility (PCF) and R/IM. The experiment will demonstrate the techniques to produce large, high-quality insulin crystals by batch process under controlled conditions. 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.

- 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. This mission will provide additional data on the polymer precipitation process. Sample materials are polysulfone, N-N-dimethylacetamide, and acetone. Following the flight, samples will be tested, and quantitative evaluation consisting of comparisons of the membranes' permeability and selectivity characteristics with those of laboratory-produced membranes will be performed.

- Air Force Maui Optical Site (AMOS) — 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.

- Auroral Photography Experiment-B (APE-B) — is designed to study the geographic extent and dynamics of the aurora (the Northern and Southern lights) by photographing the airglow aurora, auroral optical effects, Shuttle glow phenomenon, and thruster emissions.

- Ultraviolet Plume Instrument (UVPI) — The Orbiter will be used as a calibration target for space-based ultraviolet sensors. Imagery and/or signature data of the Shuttle Orbital Maneuvering System (OMS) and Reaction Control System (RCS) burns during orbital trajectory intersection will be obtained using an instrument on the Low-Power Atmospheric Compensation Experiment (LACE) and an on-orbit Strategic Defense Initiative Organization (SDIO) satellite. No flight hardware is involved.

- Lower Body Negative Pressure Experiment — The safety community has identified a concern with possible adverse effects during the conduct of the test on-orbit. Further coordination between medical, flight crew, and safety has been completed; test criteria have been agreed upon.

SECTION 3

SAFETY RISK FACTORS/ISSUES IMPACTED BY STS-43 ANOMALIES

This section lists safety risk factors/issues, considered resolved (or not a safety concern) for STS-43 prior to launch (see Sections 4, 5, and 6), that were repeated or related to anomalies that occurred during the STS-43 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-37 Inflight Anomalies		
Orbiter 2	Water Spray Boiler (WSB) #2 did not cool Auxiliary Power Unit (APU) lube oil while under operation of controller "A".	On the STS-37 flight, WSB #2 failed to cool APU #2 lube oil after the end of the pool boiling period during ascent. WSB #2 was under the operation of controller "A". The crew was directed to switch to controller "B" when lube oil temperature reached 280° Fahrenheit (F); nominal cooling begins at 250°F. Lube oil temperatures had begun to drop just prior to the crew action. The same anomaly occurred during STS-38, the previous flight of OV-104. The most probable cause of this problem was freezing of the spray bar due to wax buildup in the WSB. During STS-43 ascent, WSB #2, Serial Number (S/N) 018, gave no indication that it was cooling lube oil from APU #2, S/N 208 (IFA No. STS-43-V-02). Lube oil return temperature and gear box forward bearing temperature exceeded the specification limits; APU #2 was shut down directly after MECO. During STS-43 on-orbit flight control checkout, WSB #2/APU #2 were operated for 11 minutes. Again, no lube oil cooling was observed. WSB #2 cooling was confirmed failed, and the decision was made to invoke the flight rule allowing late turn-on of APU #2 [at Terminal Area Energy Management (TAEM)]. Lube oil temperature again ran high until APU #2 was shut down at wheel stop. (See Section 7, Orbiter 1, for more details.)

ITEM		COMMENT
Section 6: STS-37 Inflight Anomalies		
Orbiter 3	Power Reactant Supply and Distribution (PRSD) Oxygen (O_2) manifold valve #2 failed to close when commanded.	On the STS-37 flight, the PRSD O_2 manifold valve #2 failed to close when commanded the first 2 times on orbit. The valve, S/N 28, finally closed on the third command and was left closed for the remainder of the STS-37 mission. O_2 manifold valve #2 is 1 of 2 redundant valves used to isolate the manifold or PRSD supply tank from a system leak. Failure of both manifold valves to close would result in depletion of fuel cell O_2 reactants and potential loss of the 3 fuel cell power plants. No PRSD O_2 manifold valve anomalies were reported on STS-43/OV-104. However, PRSD Hydrogen (H_2) manifold #1 isolation valve failed to close when commanded on STS-43 flight day #7 (IFA No. STS-43-V-09); this H_2 isolation valve was the same model as the O_2 isolation valve that failed on STS-37. This manifold valve anomaly on STS-43 could not be repeated during STS-43 postflight troubleshooting at KSC. The valve command circuit will be instrumented for the next flight of OV-104.

SECTION 4

RESOLVED STS-43 SAFETY RISK FACTORS

This section contains a summary of the safety risk factors that are considered resolved for STS-43. These items have been reviewed by the NASA Safety Community. A description of the risk factor, information regarding problem resolution, and rationale for flight are provided for each safety risk factor. The safety position with respect to resolution 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.

The following risk factors in this section represent 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.

Integration 2	Kennedy Space Center Shuttle landing.
Orbiter 5	Main Propulsion System cryogenic temperature transducer failure.

SECTION 4 INDEX

RESOLVED STS-43 SAFETY RISK FACTORS

SECTION 4 INDEX - CONTINUED

RESOLVED STS-43 SAFETY RISK FACTORS

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Backup Flight Software (BFS) can hang in a "wait" state failure mode. HR No. ORBI-066 {AR} ***No BFS anomalies were reported on STS-43.***	During the investigation of the STS-37 BFS anomaly (see section 6, Orbiter 5) in the Shuttle Avionics Integration Laboratory (SAIL), a new BFS failure mode was discovered. Operations Sequence (OPS)-transition testing was being performed when the BFS hung in a "wait" state during a transition from OPS-0 to OPS-3. This testing was considered abnormal, because the operator rapidly toggled in and out of OPS-3 for 30 transitions; normally only single OPS transitions are performed. Transitions are not performed after Primary Flight System (PFS)/BFS synchronization. This failure occurred only once. The investigation also included a code audit of the BFS and the Primary Avionics Software System (PASS). This audit determined that the BFS only reads the counter once when processing the following critical functions: • Any OPS transition • Input/Output (I/O) resets • Transitions from Halt to Standby mode. The increased processing time of the new General Purpose Computers (GPCs), in combination with the BFS reading the counter only once for these functions, provides a 3-millisecond (ms) window for the BFS to be hung in a "wait" state when rapid, repeated critical functions of these types are performed. PASS was determined through the code audit to read the counter at least twice for all critical functions. A Discrepancy Report (DR) was written to identify and resolve the problems associated with this discovery. Appropriate changes were implemented in the BFS OI-20 software for STS-43.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)	BFS can hang in a "wait" state failure mode.	Rationale for STS-43 flight was: • This problem was corrected in the BFS OI-20 software release to be used on STS-43. ***This risk factor was resolved for STS-43.***
2	Kennedy Space Center (KSC) Shuttle landing. HR No. ORBI-021 {AR} ORBI-179 {AR} ***No problems were encountered with the STS-43/OV-104 landing at KSC.***	The return to KSC as a primary landing site, the STS-39 tire wear results, and the certification plan for improved tires were reviewed by the Space Shuttle Program as a consideration for STS-43/OV-104. The Johnson Space Center (JSC) Mission Operations Directorate (MOD) and the Flight Crew Operations Directorate (FCOD) indicated continued reservations concerning adequate tire wear margins to account for unknowns, unexpected events, pilot techniques, or landing dispersions. MOD and FCOD preferred to maintain Edwards Air Force Base (EAFB) as the primary landing site until improved systems capability, resulting in "real hardware margin," was available (i.e., drag chute or new/modified tire implementation). However, they also indicated End-of-Mission (EOM) return to KSC could be safely accomplished with conservative placards provided no significant unexpected events were encountered. Tire wear associated with landing at KSC continues to be a safety concern. The STS-39/OV-103 Right-Hand (RH), outboard tire-wear anomaly re-alerted the community to the potential for excessive tire wear during landing at KSC. Post-flight inspection determined that the outer 3 plies were excessively worn; Main Landing Gear (MLG) tires have 16 plies. Previous experience with tire wear had been limited to localized spin-up spots in the MLG tires.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)	KSC Shuttle landing.	There was no similar, uniform wear to the extent seen on STS-39 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 EOM landing site. Investigation into the STS-39/OV-103 anomaly determined that the heavy-braking Development Test Objective (DTO) resulted in the excessive wear through 3 plies on both shoulders of the RH tire. There were several areas considered by the Space Shuttle Program in weighing relative safety margins associated with selecting KSC as the primary landing site. The discussion that follows addresses risk contributors and current program efforts to reduce the associated risk. <u>Orbiter Braking</u>. The availability of carbon brakes on the Orbiter has been a requirement for a nominal, EOM landing at KSC since the return to flight. Carbon brakes are now the baseline system for the Orbiter and provide 82-million foot-pounds (ft-lb) of brake energy vs 42-55 million ft-lb for the previously-used beryllium brakes. The carbon brake DTOs were completed on OV-103 with successful results; the DTO data compared favorably with Wright Patterson Air Force Base (WPAFB) dynamometer test results. The last DTO on STS-39/OV-103 included heavy braking, and demonstrated that the anti-skid system cycled properly. All the Orbiters except OV-102 now have carbon brakes; OV-102 will be fitted with carbon brakes during its forthcoming major modification period. Some portion of the DTOs completed on OV-103 will be repeated on the rest of the Orbiter fleet as carbon brakes are installed and used.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)	KSC Shuttle landing.	Tire Wear. The 6-cord tire wear limit has been specified as acceptable based on engineering judgment and WPAFB load tests. The 6-cord wear limit at wheel stop specifies not exceeding 4 cords during peak loads and allowing for 2 cords of wear during the rollout. Tire wear models were developed at Langley Research Center (LaRC) and used at Ames Research Center (ARC) to determine crosswind effects. Ames simulation confirmed that tire wear is influenced by both increasing crosswind and increasing vehicle weight. The 6-cord wear limit has an error band of ± 2 cords that is not accounted for in the Ames simulation. LaRC tests determined that 8-cord wear can survive 245,000 lb/15 knot crosswind conditions. However, no consideration was given to the added effect of braking on tire wear. LaRC tests also demonstrated that additional grinding of the KSC landing surface does not and will not result in reduced tire wear. Improved commercial rubber compound tires that are now under evaluation provide better wear capability than the tires currently used. The LaRC sled tests of existing Orbiter tires show wear to failure before 4-million ft-lb of side energy. This side energy is typical of 3-sigma worst-case wear for 20-knot crosswinds according to Ames simulation results. The LaRC tests show 4/32"-5/32" wear for the improved tire at 4-million ft-lb of side energy. The improved tire has 4/32" of rubber remaining before it is worn to the first cord. KSC was directed to initiate procurement of new commercial tires for delivery as soon as possible. KSC estimated January 10, 1992 as the earliest delivery date to support the STS-50/OV-102 flight scheduled for June 1992. The Orbiter Project schedule objective was to have the new tire certification "Qualification Site Approval" by August 19, 1991. There are 14 tires currently available at WPAFB for

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
2 (Continued)	KSC Shuttle landing.	certification tests. Wind placards have been established to provide margin within predicted crosswind variation after committing to KSC landing (90-minute forecast). These wind placards are: • For a vehicle with landing weight less than 220,000 lb, the maximum allowable crosswind component is 12 knots peak. • For a vehicle with landing weight greater than 220,000 lb, the maximum allowable crosswind component is 10 knots peak. Real-time wind evaluation assumes gusts are in the direction of the average wind. The Orbiter could experience higher crosswinds than predicted based on short-term variability of the direction and magnitude of wind gusts. <u>Drag Chute Implementation</u>. The first Drag Chute is scheduled to fly on STS-49/OV-105 in April 1992. The current flight effectivities for the other Orbiters are: STS-50/OV-102 in June 1992, STS-53/OV-103 in October 1992, and STS-57/OV-104 in mid-1993 following major modification periods.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)	KSC Shuttle landing.	Weather Forecasting. A KSC weather forecast demonstration effort was initiated in June 1988 to provide 24-hour and 90-minute forecasts for each EOM landing site. Using actual Flight Rule (FR) criteria for "go/no-go" determination, this demonstration effort has collected accuracy data for the last 2 years by checking forecasts against actual observations at the "forecast time". Because the majority of these forecasts were made with less than flight/landing-day capabilities (i.e., Shuttle Training Aircraft flight support), the statistical data acquired through this effort is considered worst case. In situations where the 90-minute sunrise forecast predicted a "go" for launch or landing, there was a recorded accuracy of over 95%. Where the forecast determined a "no-go" situation, the forecast accuracy was 77% (meaning 23% of the time a "no-go" determination was made, the "forecast time" weather met the "go" criteria). Compared to 90-minute forecasts for mid-morning and afternoon launches/landings, the sunrise 90-minute forecast accuracy represented the worst case for KSC. When compared to other nominal EOM landing sites, there are roughly 6 times as many incorrect forecasts at KSC as there are now at EAFB; EAFB has 1 every 166 forecasts, while KSC has 1 every 27 forecasts, and Northrup has 1 every 41 forecasts. On any given day the probability of a "no-go" forecast is 1/3 at KSC compared to approximately 1/10 at EAFB. Conditions are sensitive to early morning fog around sunrise at KSC. Therefore, the uncertainty of actually landing in weather predicted at the deorbit commitment is somewhat greater at KSC than

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)	KSC Shuttle landing.	at EAFB. With respect to KSC weather forecasting, returning to KSC as a prime, nominal EOM landing site is considered acceptable based on the 2-year forecast demonstration effort. The demonstrated forecast accuracy will be enhanced on launch/landing day with real-time assessment by the Shuttle Training Aircraft. Directional Control. Acceptable lateral control is provided by either Nose Wheel Steering (NWS) or differential braking, assuming no tire failures prior to or during derotation. Existing FRs do not protect for a tire failure prior to/during derotation. MOD recommendations indicated that differential braking is an adequate backup to NWS, and redundant NWS is not required for primary return to KSC. Current FRs do require NWS prior to a KSC landing pre-deorbit commitment. Lack of runway shoulder bearing capability is not considered a constraint to a nominal landing at KSC because NWS and differential braking are adequate controls against going off the runway. There is a 1992 effort funded to bring the KSC runway shoulders back to the "as-designed" condition. Unknown/Unexpected Events. Concern remains in the Space Shuttle Program community that unknowns/ unexpected events can result in violation of safe tire wear limits. It is desirable to operate with sufficient vehicle capacity remaining to survive unexpected occurrences: landing/deceleration failure, gust magnitude and direction variability, inaccurate forecasts, crew dispersion in response to an unexpected event. STS-39 provided

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
2 (Continued)	KSC Shuttle landing.	new evidence of tire wear due to braking and also showed sensitivity of tire wear to vehicle drift and/or to steering inputs on the KSC crowned runway. From a weather perspective, the STS-39/OV-103 landing was made on a good day. It was recommended by some members of the community that the return to KSC as a primary, nominal EOM landing site be delayed until either the new tire or drag chute is implemented on the Orbiter. The Space Shuttle Program Director requested that Level II look at wind placard considerations that will provide adequate safety margin for landing the Orbiter at KSC under low vehicle return weight/low crosswind conditions. As a result of Level II review, KSC was established as the primary landing site for vehicle weights ≤205,000 lb; EAFB is the secondary landing site. For vehicle weights >205,000 lb, EAFB will be the primary landing site with KSC a secondary site. Revised landing wind constraints were established for KSC EOM: • For vehicle weight ≤205,000 lb, crosswind ≤ 12 knots. • For vehicle weight >205,000 lb, crosswind ≤10 knots. • Peak wind, regardless of direction, must be ≤20 knots. Rationale for STS-43 flight was: • The lower landing weight/wind restrictions provide increased tire wear safety margin for landing at KSC as the primary landing site. ***This risk factor was acceptable for STS-43.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	External Tank (ET) umbilical door lug clevis cracks on STS-37/OV-104. HR No. ORBI-302A {AR} ***No anomalies associated with the Orbiter/ET umbilical door mechanism were experienced on STS-43.***	ET umbilical door lug clevis cracks and displacement observed on OV-103 led to increased inspection and awareness on other Orbiters. Dye-penetrant and etch inspection of OV-104 prior to STS-37 revealed crack initiation on 3 of 4 lug clevises; the fourth clevis showed indications of pitting. Eddy current measurements of the cracks indicated a maximum depth of 0.025" ±0.005". No lug displacement was noted, and adjustments made to the door rigging during the STS-37 flow were considered unaffected by the presence of clevis cracks with the current flaw size. Discovery of these cracks on STS-37/OV-104 was made after the ET umbilical doors were locked open on the centerline latches. Previous inspection of STS-37/OV-104 lug clevises, with the ET umbilical doors at the 90° position, found indications of pitting only. After careful consideration of these conclusions and the independent analysis performed by the Orbiter Project and Headquarters Code QT, the Space Shuttle Program Director and the Associate Administrator for the Office of Safety and Mission Quality accepted the residual risks associated with launching STS-37/OV-104 with starter cracks. Since STS-37, all OV-104 ET door lug housings have been replaced with the modified, "J-leg" lug-configuration housings. This action completed the retrofit of the Shuttle fleet with the modified housing. Rationale for STS-43 flight was: • The ET umbilical door lug housings flown on STS-37 were replaced with the modified, "J-leg" housing configuration. ***This risk factor was resolved for STS-43.***

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2	Potential for new GPCs to erroneously overwrite memory. HR No. ORBI-194 {AR} ***No similar GPC anomalies were reported on STS-43.***	At the STS-37 Level III Orbiter Flight Readiness Review, International Business Machines (IBM) reported a generic hardware problem with the new AP-101S GPCs. The problem occurred when the transition was made from halt (also known as sleep or freeze-dried mode) to an operational mode; the GPC could cause random locations of memory to be overwritten. The problem was first seen with the Shuttle Mission Simulator (SMS); however, it was initially believed to be unique to the SMS. A hardware design change to a single page in the GPCs was identified to fix this problem. OV-104 GPCs were removed during the STS-43 flow and replaced with AP-101S GPCs with this fix. Rationale for STS-43 flight was: • AP-101S GPCs, with the single page change to alleviate this problem, were installed in STS-43/OV-104. ***This risk factor was resolved for STS-43.***
3	Debris found in STS-37/OV-104 window #1. HR No. ORBI-208 {AR} ORBI-339 {AR} ***No problems were experienced with the Orbiter windows on STS-43.***	During STS-37/OV-104 window inspection at the pad, 3 small pieces of Room-Temperature Vulcanizing (RTV) were found between the outer thermal pane glass and the inner pressure pane glass. The largest piece of RTV was estimated to be 3" long and 1/4" wide. The debris was located on the ledge between the outer thermal pane and the inner pressure pane. No debris was witnessed in the window cavity vent area. The concern was that the debris could block the window cavity vents during either ascent or descent. No problems were experienced, however, during the STS-37 mission.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3 (Continued)	Debris found in STS-37/OV-104 window #1.	The window #1 thermal pane was removed during the STS-43 flow to replace the viton seal with a silicon seal and the RTV debris was removed. Rationale for STS-43 flight was: • The RTV debris was removed during the STS-43 flow. ***This risk factor was resolved for STS-43.***
4	OV-105 T-handle access door for the emergency egress window jettison opened during ferry flight. HR No. ORBI-142A {AR} ***No problems with the T-handle access door were reported on STS-43.***	During the initial leg of the OV-105 ferry flight from Palmdale to KSC, the T-handle access door was found unlatched. The door was relatched prior to continuation of the ferry flight. The initial concern was that the OV-105 T-handle access door was rigged, latched, and closed-out prior to the ferry flight in accordance with Operations and Maintenance Instructions (OMIs) used to secure T-handle doors on all Orbiter vehicles; however, it was later reported that a Rockwell/Palmdale procedure was used to close out the OV-105 T-handle access door. The door was resecured at El Paso, Texas, after the first leg of the ferry flight and prior to continuing on to KSC. It was later learned that, following proper closeout of the T-handle access door, a request was made to identify and record the part number of a safety pin that had been installed in the T-handle for the ferry flight. To access the part number, the access door had to be opened. The technician who opened the access door to record the safety pin part number admitted that he had not followed procedures in

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

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4 (Continued)	OV-105 T-handle access door for the emergency egress window jettison opened during ferry flight.	Visual inspection of proper T-handle access door position is performed prior to each launch. Functional open/close, latch/unlatch, and push force verification tests are performed every 5 flights. Door seals that create the preload on the door latch mechanism are also inspected every 5 flights. There is currently no inflight capability to determined if the access door is open. Stress analysis indicates that the load on the T-handle access door latch plunger imposed by launch vibration is 7.2 pounds-force; the force required to overcome the seal preload and unlatch the door is 11.9 pounds-force. All T-handle access door components have been designed with a minimum factor of safety of 1.4. Inspection of the OV-104 T-handle access door was performed during the STS-43 flow. The door was found to be in the proper, latched position. Rationale for STS-43 flight was: • The OV-105 incident was caused by improper securing and closeout of the T-handle access door. • Recent inspection found the STS-43/OV-104 T-handle access door in the proper closed configuration. ***This risk factor was resolved for STS-43.***

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Main Propulsion System (MPS) cryogenic temperature transducer failure. HR No. INTG-023 {AR} ME-C2 (All Phases) {C} ME-D2 (All Phases) {C} ***No cryogenic temperature transducer anomalies were reported on STS-43.***	During the STS-35/OV-102 Hydrogen (H_2) leak investigation in September 1990, a 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. S/N 105, -0021 transducer was removed and sent to the vendor, RDF Corporation of New Hampshire, for failure analysis. Due in part to logistical problems, the results of the RDF failure analysis were not available until May 20, 1991. Full, 360° circumferential cracks were found 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 loss of the 2" mandrel/sheath transducer tip into the oxidizer or fuel feedline during SSME operation. One temperature transducer is located in each of the 3-LH_2 and 3-LO_2 12" feedlines to the SSMEs just prior to the inlet of the Low-Pressure Fuel Turbopumps (LPFTPs) and the Low-Pressure Oxidizer Turbopumps (LPOTPs). There are no filters or screens to prevent a piece of temperature transducer from entering the LPFTP or LPOTP. If a portion of the temperature transducer broke off and entered the LPFTP or LPOTP, the results would be catastrophic. The cracks found on the S/N 105, -0021 transducer led to the requirement to remove and x-ray the OV-102 and OV-103 MPS temperature transducers. All

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	MPS cryogenic temperature transducer failure.	spare temperature transducers available at 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 either found with similar circumferential cracks in the sheath-to-mandrel weld or with indications of cracking. To date, 6 of 17 -0021 MPS temperature transducers have been found with weld cracks. No other ME449-0013 dash number temperature transducers x-rayed to date have been found with cracks. Except for S/N 105, there have been no other -0021 transducers found with cracks in the necked-down area of the mandrel. The -0021 temperature transducers are used exclusively on the LH_2 side of the MPS system. The design of cryogenic temperature transducers now used in the MPS was based on transducers first used on the Apollo/Saturn Program. The -0021 transducer configuration was the replacement for the -0018 temperature transducers that were used in the early stages of the Space Shuttle Program. Early problems with the electronics assembly in the -0018 transducers led to a design change which resulted in the -0021 configuration. In addition to the -0018 and -0021 transducers used on the LH_2 side of the MPS, -0017, -0020, and -0022 temperature transducers are used on the LO_2 side of the MPS. All 5 configurations have similar sheath-to-mandrel designs in the tip of the transducer. However, there are significant differences in the resulting sheath-to-mandrel weld penetration across the transducer configurations. All P/N ME449-0013 transducer configurations were

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	MPS cryogenic temperature transducer failure.	manufactured to the same automated weld schedule. For the sheath-to-mandrel interface, the weld schedule requires a minimum weld penetration of 0.005". RDF Corporation shop practices resulted in a weld penetration no greater than 0.010" for all transducer configurations. Variances in weld penetration percentage among the transducer configurations resulted from different sheath thickness requirements at the weld. The -0018 had a sheath thickness of 0.010", resulting in weld penetrations of 50% to 100% (0.005" to 0.010"). When the same weld schedule was applied to the -0021 configuration, with a sheath thickness of 0.040", the resulting weld penetration ranged from 12% to 25%. The added sheath thickness reduced the structural integrity of the sheath-to-mandrel weld between the -0018 and -0021 configurations. The LO_2-side transducers (-0017, -0020, and -0022) have the same sheath thickness requirement, 0.020". This results in a weld penetration of 25% to 50%. Since finding circumferential cracks on the -0021, S/N 105 transducer, a stress/fracture sensitivity analysis of all transducer configurations was performed. The analysis considered temperature change experienced by the transducer during tanking and SSME operation (ΔT = -493°F for LH_2 transducers, ΔT = -363°F for LO_2 transducers) and material thermal expansion/contraction property differences. The analysis compared thermally-induced stress to critical fracture stress to determine the design margin of the weld in all configurations. The results indicate that thermally-induced stress can exceed the stress required to induce total fracture of the 10% weld in the -0021 configuration. Thermally-induced stresses on the

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5 (Continued)	MPS cryogenic temperature transducer failure.	-0017, -0020, and -0022 units have been determined through analysis to be insufficient to lead to weld fracture, given the requirement for minimum weld penetration of 25%. The analysis results support the findings of the x-ray inspections performed on OV-102, OV-103, and spare temperature sensors. Thermal analysis did not, however, identify sufficient force to result in the circumferential cracking found in the -0021, S/N 105, necked-down area of the mandrel. Scanning Electron Microscope (SEM) inspection was performed of the cracks found in both the sheath-to-mandrel weld and the necked-down area of the mandrel. The SEM inspection revealed uniform, brittle fractures around the circumference of both cracks. No ductility was observed in the sheath-to-mandrel weld crack. The sheath-to-mandrel crack was indicative of a failure at cryogenic temperatures. Indications of ductility in the form of shear dimples were found in the mandrel crack. This finding demonstrated that the crack was initiated by a force other than thermally-induced stress at cryogenic temperatures. An impact test using a punch and hammer and a coupon of identical mandrel material was performed at room temperature to determine if failure could be manually induced. The results of the impact test duplicated the crack found in the necked-down area of the mandrel. This test demonstrated that the -0021, S/N 105 mandrel crack most likely was caused by forces outside of the operational environment. The exact cause is still under investigation. The investigation into this issue has also 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

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ORBITER		
5 (Continued)	MPS cryogenic temperature transducer failure.	the SSME throttle-down Flight Rule (5-50) would be required to allow removal of the LH_2 temperature transducers at the inlet of the LPFTP. LPFTP LH_2 inlet temperatures can be replaced with a constant temperature value determined from flight experience and updated in real-time based on the LH_2 manifold temperature transducer, the fourth LH_2-side transducer. Initial review 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. Further, detailed analysis was required to identify alternative approaches to protect the engine start box in the event that residual risk associated with installing temperature transducers near the LPOTP inlet was deemed unacceptable. For STS-43/OV-104, 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 will be used in place of the -0021 LH_2 manifold transducer to further reduce the risk. 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 will screen for structural failures by monitoring for off-scale 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-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	MPS cryogenic temperature transducer failure.	Rationale for STS-43 flight was: • All MPS temperature transducers installed on OV-104 successfully passed x-ray inspection. • 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 will not allow a launch with anomalous transducer indications. ***This risk factor was acceptable for STS-43.***
6	MPS Helium (He) tank leak. HR No. INTG-019 {AR} INTG-041 {AR} ORBI-111 {C} ORBI-129A {C}	During MPS high-pressure pneumatic decay in the Orbiter Processing Facility (OPF), a pressure drop of 140 pounds per square inch absolute (psia) in 72 hours was observed; the allowable system limit is 100 psia. The leak source was isolated to He tank #4, S/N 33, which is the 4.7 cubic-foot SSME supply, located in the aft fuselage. High-pressure decay testing at 2000 pounds per square inch (psi) indicated 1.94 psi/hour He leakage [4.9 standard cubic centimeters per second (sccs)]. The Operational Maintenance and Requirements Specifications Document (OMRSD) allowable leakage is 1.38 psi/hour for the MPS pneumatic system.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6 (Continued)	MPS He tank leak. ***No helium leakage anomalies were reported on STS-43 following removal and replacement of the leaking tank.***	A second high-pressure (2000 psi) decay test indicated lower He leakage: 0.85 psi/hour (2.15 sccs equivalent). Mass spectrometer leak checks of the pneumatic tank indicated leakage $>1x10^{-5}$ sccs; standard allowable leakage is $1x10^{-6}$ sccs for other weld/braze joints. The tank was removed and returned to the vendor (Brunswick), where the leakage condition was confirmed: 27 sccs at 4420 psi, 2 sccs at 2050 psi. The failed tank, S/N 33, had 90 recorded pressure cycles; a pressure change >500 psig is considered 1 pressure cycle and tracked and recorded for each tank. Further evaluation of the S/N 33 production data packages indicated that the welding procedures used were not the same as those originally qualified. The tank was then stripped of the Kevlar overwrap for further troubleshooting and leak isolation. Subsequent low-pressure leak tests and visual (10x) inspection were not able to pinpoint the crack location. At this point, the tank was sent to Rockwell/Downey for a failure analysis to determine the exact nature of the very tight crack causing the leak. The leak was located at a pore that developed a "tail". X-ray inspection revealed two cracks in the structural weld of the titanium liner: a 0.05"-long crack emanating from a 0.023" pore and a 0.03"-long crack connecting pores of 0.005" and 0.015". The x-rays, 36 shots at 10° intervals around the circumference, found 8 pores with "tails". This was potential evidence that porosity had grown into defects. Original build liner x-rays did not reveal cracks. Original post-proof tank x-rays were reevaluated; the original x-ray report did not identify any cracks. However, 89 pores were identified that were acceptable based on size and spacing. A high number of pores could be a manifestation of contaminated weld surfaces, filler wire, weld gas, or weld chamber. Tank S/N 33 had Material Review (MR) acceptance for weld pores during fabrication. Another of the OV-104 tanks also had an MR on weld porosity — tank #7 located in the mid-body under the payload bay liner.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	MPS He tank leak.	Hydrostatic cycle testing was also performed at Rockwell/Downey on the failed tank S/N 33; rupture occurred after 170 cycles. Test results indicated that the tank exhibits detectable leakage long before tank failure. From the initiation of leakage, crack growth is stable; a sudden failure (gross leakage or rupture) is, therefore, not expected. This test also verified that the tank met the "leak-before-burst" requirement. Build data was then reviewed for all OV-104 tanks: Acceptance Test Procedure (ATP) x-ray reports, MRs, and DRs. Two -0010 tanks, S/N 35 and S/N 37, evidenced similar weld porosity. All 10 STS-43/OV-104 MPS He tanks were checked by mass spectrometer with acceptable results. The SSME MPS pneumatic systems were isolated, and a 72-hour decay check was initiated on June 12, 1991 after replacement of the leaking tank; test results were satisfactory. In addition, review of OV-104 Nitrogen (N_2) and He tank decay data for the Environmental Control and Life Support System (ECLSS), Orbital Maneuvering System (OMS), and Reaction Control System (RCS) indicated no leakage. A review of DRs, MRs, waivers, and Corrective Action Requests (CARs) indicated no unresolved problems. An analysis of crack propagation was performed. The OMRSD leak rate limit of 1.38 psi/hour at 2000 psi converted to a theoretical crack length of a 0.070" through crack. Fracture analysis, performed using nominal 6-4 titanium properties, resulted in survival with 10 0-4500 psi pressure cycles with a scatter factor of 4; the crack

RESOLVED STS-43 SAFETY RISK FACTORS

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<u>ORBITER</u>		
6 (Continued)	MPS He tank leak.	goes unstable at 0.47" length. However, the analysis is very sensitive to material toughness variations; toughness value reduction of 10% reduces calculated life from 10 cycles to 3 cycles, with a scatter factor of 4. Rationale for STS-43 flight was: • The leaking tank (S/N 033) was replaced with a new tank (S/N 038). • System leak tests were rerun and indicated that the installed tanks were within OMRSD requirements. Mass spectrometer tests verified that the MPS He tanks did not have comparable leakage to the S/N 033 tank. • Crack growth is stable; therefore a sudden failure is unlikely. • Analysis shows that tanks which develop cracks sufficient to cause the observed leakage can sustain multiple pressure cycles prior to rupture. Tests performed on the failed tank demonstrated that it met the "leak-before-burst" requirement. • Inflight leakage will not increase. Constant system pressure decay and resultant stress levels decrease after T–0. • Flight experience indicates acceptable tank performance. The last flight of OV-104 experienced approximately 4.6 sccs He leakage on orbit; allowable leakage is 8.2 sccs. ***This risk factor was resolved for STS-43.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	STS-43/OV-104 Return-To-Launch-Site (RTLS) landing weight. HR No. ORBI-021A {AR} ORBI-179A {AR} ***STS-43/OV-104 was successfully launched and placed into orbit, and the Tracking and Data Relay Satellite (TDRS-E) payload was launched from the OV-104 payload bay. Therefore, there was no RTLS required and the OV-104 landing weight was well within specification limits.***	The STS-43/OV-104 RTLS landing weight exceeded the 240,000-pound (lb) specification limit. The RTLS down weight for STS-43 was projected to be 240,300 lb; a waiver was authorized to exceed the RTLS down weight limit for STS-43 only. Main Landing Gear (MLG) strut/tire loads were evaluated based on the projected RTLS landing weight. The MLG assembly (except tire) is acceptable for a 256,000-lb landing weight per the 6.0 loads analysis. The MLG tire is qualified for an Orbiter landing weight of 240,000 lb, initial sink speed of 5 feet per second with a 20-knot crosswind and 225 knots equivalent airspeed touchdown velocity. The loads model predicted a tire peak load of 131,000 lb for STS-43/OV-104; tire certification limit is 132,000 lb. Off-limits tire testing has demonstrated additional peak load capability: 138,000 lb at 315 pounds per square inch (psi) inflation pressure; 178,000 lb at 340-psi inflation pressure. The current OMRSD requires the tire to be inflated to a pressure of 365 psig (+5, –0 psig). Assuming a leak rate of 0.4 psig per day, the maximum allowable leak rate per the tire manufacturer's specification, STS-43/OV-104 tire pressures were expected to be approximately 349 psig for a launch date as late as July 25, 1991. OV-104 was accordingly cleared for rollout from the Orbiter Processing Facility (OPF) during the Orbiter Rollout Review on June 18, 1991. Rationale for STS-43 flight was: • Based on the current wheel load model, the expected 131,000 lb wheel load is within the current certification limit of the MLG wheel. The STS-43 predictions are well within the main gear system combined limit load capabilities.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)	STS-43/OV-104 RTLS landing weight.	• A higher minimum tire pressure of 340 psig or greater (instead of 315 psig) was used on STS-43 to increase tire load capacity. ***This risk factor was acceptable for STS-43.***
8	STS-43/OV-104 Engine Interface Unit (EIU) Power-On Reset (POR) anomaly. HR No. INTG-165 {C} ORBI-066 {AR} ***No EIU POR inflight anomalies were reported on STS-43.***	During STS-43/OV-104 EIU testing on June 28, 1991, S/N 20 experienced a single POR. The POR bit in the GPC Built-In Test Equipment (BITE) circuit was "on" when it was expected to be "off"; duration was 1.1 seconds (sec). The most probable cause was isolated to 1 of 8 internal EIU circuits (power supply/BITE). All POR reset occurrences have been an average duration of 1.2 sec on a single power supply only. Several single PORs have been experienced on other EIUs; however, this is the fourth single POR experienced with EIU S/N 20 during the life of the program. Because only sporadic, single PORs have been experienced during ground testing, the decision has been made to fly STS-43/OV-104 without replacing S/N 20. EIU PORs experienced in the past have led to the removal and replacement of the anomalous EIU. There have been no PORs experienced in flight. The concern is that a simultaneous POR-A and POR-B in the last 30 seconds (sec) prior to Main Engine Cutoff (MECO) would result in the GPCs closing prevalves on running engines, resulting in a catastrophic engine shutdown (this is the worst-case failure scenario). Failure of GPC command of the engines requires manual shutdown. Flight rules and crew procedures exist for this condition. System management alert and the main engine status light (amber) on panel F-7 will alert the crew to POR and a failure of GPC command of the engine. Crew reaction is

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ORBITER		
8 (Continued)	STS-43/OV-104 EIU POR anomaly.	required to manually shut down the main engines prior to prevalve closure to preclude catastrophic engine shutdown. The crew would require a ground call to confirm the POR indication; however, they cannot react in time to preclude catastrophic engine shutdown if this occurs during the last 30 sec prior to MECO. The recent single POR in EIU S/N 20 was characteristic of previous PORs experienced in the program. This POR was transient and self-clearing. Troubleshooting did not reproduce the problem. Rationale for STS-43 flight was: • Every EIU POR to date in the program has been during ground testing and has been a single POR (POR-A or POR-B). An EIU POR has not occurred in flight. • No single POR will fail more than 1 data command channel. Momentary POR only affects engine command capability for the duration of the POR/1 command cycle. The next subsequent command will be received by the SSME Main Engine Controller (MEC). • A single POR prior to T-0 would result in a launch hold or launch abort. • The last 30 sec prior to MECO is a very short time window. The probability for simultaneous failure of both POR-A and POR-B during this time has been calculated to be 2.5×10^{-11}. ***This risk factor was acceptable for STS-43.***

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ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
9	Generic power supply problem in the new GPCs, AP-101S. HR No. INTG-021B {AR} INTG-065C {AR} ***No GPC anomalies were reported on STS-43.***	GPC 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 provides power to logic devices within the GPC, including the main memory devices. Investigation found that 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 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 0.002-0.022" clearance below the power supply frame. The worst case of 0.002" clearance was found on the Spacelab AP-101SL

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9 (Continued)	Generic power supply problem in the new GPCs, AP-101S.	qualification unit that was subjected to full qualification thermal and vibration testing. Thinner washers, tighter bushing tolerances, and a new 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 have been detected during factory testing. There have been no incidents of this type experienced in the field on laboratory or flight units; accumulated field operating time is 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 have been through thermal and vibration testing. Two or more GPC failures during ascent, coupled with an EIU or SSME controller failure that produces more than 1 command path failure to the main engine during the last 30 seconds prior to 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. Rationale for STS-43 flight was: • Failures experienced to date have been detected during factory testing. • Thermal and vibration testing are valid screens for this failure; all flight GPCs, including those on STS-43/OV-104, passed these tests.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9 (Continued)	Generic power supply problem in the new GPCs, AP-101S.	• Loss of GPC output is Crit 1R2 (worst-case). • GPCs are redundant, and a spare GPC was available on the Orbiter for replacement of a failed unit on orbit. ***This risk factor was acceptable for STS-43.***
10	AP-101S General Purpose Computer (GPC) sleep mode anomaly. ***No GPC anomalies were reported on STS-43.***	AP-101S GPC, S/N 531, failed during factory acceptance testing at IBM in late June 1991. This GPC was 1 of the second lot of AP-101S GPCs ("14 build"), S/N 527 through 540. The S/N 531 failure mode was due to the GPC consistently going into the "sleep mode" when powered up with the mode switch positioned to "halt". The anomaly does not occur when the GPC is powered up with the mode switch positioned to either "run" or "standby". Four "14 build" units in the JSC Avionics Engineering Laboratory (JAEL) were tested; the failure did not occur on 1 of the units (S/N 529) but did occur on the other 3 (S/N 530, 532, and 533). However, this failure anomaly has not been reported on production GPCs from the first GPC lot (S/Ns through 526), or the first 2 "14 build" units (S/N 527 and 528). Measurements showed that the failure occurs when the latch flip-flop incorrectly changes to the "sleep" signal output. Preliminary investigation indicated an integrated circuit problem with inverters obtained from National Semiconductor; these inverters may be subject to a transient reset trigger at power-up that activates "sleep" operation when paired in the "sleep mode" latch application. Analysis

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ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
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10 (Continued)	AP-101S GPC sleep mode anomaly.	is continuing at IBM to verify the problem cause, and an engineering change is being considered to correct the problem. This anomalous operation has not been experienced with Texas Instrument inverters paired in the "sleep mode" latch application; this is the present configuration used in flight GPCs. Rationale for STS-43 flight was: • The GPCs installed on STS-43 all have Texas Instrument inverters installed in the "sleep mode" latch; these GPCs have never exhibited this problem. • GPC loss of output worst-case Crit is 1R2. • Present procedures leave the AP-101S GPCs powered on during flight. Therefore, the GPCs would not be subject to a power-on transient to trigger them into the "sleep mode" unless a computer needs to undergo an Initial Program Load (IPL) during flight operations. A simple workaround for reentering the IPL into the GPC has been developed if necessary; if the GPC mode switch is set to "standby" and then "halt", the IPL can then be reentered satisfactorily. ***This risk factor was resolved for STS-43.***

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<u>ORBITER</u>		
11	Right outboard elevon noise. HR No. ORBI-003 {C} ***No elevon anomalies were reported on STS-43.***	During aerosurface recycling at the pad, a "popping" noise was heard on the right outboard elevon in the area of flipper doors #14 and #15. Visual and borescope inspections were performed with no definitive results. The only significant finding was at flipper door #14 where the elevon pushrod attachment to the bell crank showed signs of the cotter pin contacting the structural beam; no damage was found, only small scratches in the Korpon surface. Flipper doors #12-#15 were raised, and the elevon was cycled successfully. Flipper door #15 was lowered, and the elevon was cycled successfully. The cycling was repeated with flipper door #14 lowered. The problem was believed to be in the "blade seal" between flipper doors #14 and #15. No noise associated with operation of flipper doors #14 and #15 was observed. However, it was found that a spacer ring that goes in the seal between flipper doors #14 and #15 had not been installed; therefore, the doors were not exactly in the flight configuration. Spacer ring installation was completed, and all 4 doors that were opened (#12 - #15) were then in the flight configuration. A small area of missing Korpon was also noted along the outer elevon wing tip behind the outboard blade seal, and a small area was found to be slightly dented under the door #15 inboard nut plate rub panel. The missing Korpon area was painted. On July 12, 1991, hydraulics was brought up again using the recirculation pumps (same as the original condition when noise was heard). No noise was heard. This issue was scheduled to go to the Level II Program Requirements Control Board (PRCB) on July 19, 1991 as an unexplained anomaly.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11 (Continued)	Right outboard elevon noise.	Rationale for STS-43 flight was: • The flipper doors were recycled and operated nominally with no noise observed. ***This risk factor was resolved for STS-43.***
12	Avionics software discrepancy provides the potential for an undetected failure of an SSME controller channel. HR No. INTG-165 {C} ORBI-066 {AR} ***No similar anomaly was experienced on STS-43.***	DR 101776 was written in 1989 against the avionics software because it was determined that the GPC was not reading all SSME controller channel fail flags. Troubleshooting determined that the GPC only looks for an SSME channel failure flag in the time frame between issuance of a command to the 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) a MCF indication is not set. The software design in both the Block I controller, currently in use, and 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.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
12 (Continued)	Avionics software discrepancy provides the potential for an undetected failure of an SSME controller channel.	DR 101776 was elevated to a Crit 1 software discrepancy due to the determination that the SSME controller software does not set a 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 sec to Solid Rocket Booster (SRB) ignition when occurrence of this discrepancy could result in 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 which 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 1 scenario results in contingency recovery of engine control through crew intervention. As in the case of dual

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
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12 (Continued)	Avionics software discrepancy provides the potential for an undetected failure of an SSME controller channel.	EIU POR, a second SSME controller channel loss in the last 30 sec prior to 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 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. Rationale for STS-43 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. ***This risk factor was acceptable for STS-43.***

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ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
13	Multiplexer-Demultiplexer (MDM), Flight Aft-3 (FA-3) power supply BITE anomaly on STS-43/OV-104. HR No. ORBI-038B {AR} ***No MDM inflight anomalies were reported on STS-43.***	During nominal launch preparation testing, MDM FA-3, S/N 43, indicated a power supply failure through a BITE indication. The MDM power supply BITE was exercised several times with FA-3 in the same configuration it was in at the time of the original incident, and the power supply failure indication was received each time. Tests of vehicle power buses were performed, individually dropping buses; no problems were experienced. If there were a problem with 1 of the 2 power supplies, the MDM would not operate properly when the good power supply was disabled. Successfully passing the power bus drop test demonstrated that both FA-3 power supplies were operating nominally and the power supply BITE circuitry was giving false failure indications. The determination that the actual failure was in the BITE circuitry led to the decision to not replace MDM FA-3, S/N 43, prior to flight. Rationale for STS-43 flight was: • Troubleshooting determined that both redundant power supplies in FA-3 were good and isolated the problem to the BITE circuitry. • Power supplies were redundant within each MDM. • MDM redundancy was adequate to protect against loss of vehicle and crew. ***This risk factor was acceptable for STS-43.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14	Contamination found in OV-103 MPS. HR No. INTG-023 {AR} ME-A1A Rev. F {AR} ***No engine anomalies attributed to MPS contamination were reported on STS-43.***	Decay checks of OV-103 MPS engine #1 determined that the Liquid Hydrogen (LH_2) feedline exceeded the 0.25 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 decision was made to remove and replace the relief valve. Testing showed that the new valve was operating properly and engine #1 decay tests were successfully completed. Post-removal inspection found that the relief valve would not properly seat due to the presence of contamination. The contamination was determined to be a plastic sliver, measuring 0.30" x 0.15". Further inspection of the entire OV-103 MPS was directed based on this finding. Inspection of engine #1 MPS components only 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 was found with several pieces of contamination; 10-to-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 found no similar contamination.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14 (Continued)	Contamination found in OV-103 MPS.	Analysis of the contamination at a 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 #2026, #2030, and #2029 were on STS-39, the last flight of OV-103. These engines were in the KSC engine shop and were to be inspected for contamination in the near-term. There was no indication of anomalous engine operation during STS-39. STS-43/OV-104 engines, #2024, #2012, and #2028, were last installed on STS-35/OV-102. No anomalous engine performance was recorded on STS-35 or on the last flight of OV-104, STS-37. Rationale for STS-43 flight was: • This was considered an isolated contamination incident relating only to the OV-103 MPS; no similar contamination was expected to be found in the OV-104 MPS. • There were no engine anomalies reported on the last flight of OV-103 or OV-104. ***This risk factor was acceptable for STS-43.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1	High-Pressure Oxidizer Turbopump (HPOTP) main housing contamination during fabrication. HR No. ME-C1 Rev. F {AR} (All Phases) ***No HPOTP anomalies were reported on STS-43.***	There is a potential for non-Liquid Oxygen (LOX) compatible material in HPOTP housings. Borescope inspection of main housing passages of new HPOTP #2029 revealed contamination at the trailing edge of diffuser vanes. The contaminant was identified as "peen-scan" coating used in the shot-peening process to increase fatigue life. Borescope inspection of additional housings found this contaminant in 6 of the 12 housings; the largest area of peen-scan was found in the HPOTP #2029 housing. Maximum time on any of the contaminated units was 51 starts/18,361 seconds. The diffuser vanes are shot-peened at the vendor during the fabrication process. Peen-scan material is applied to the diffusers to verify shot peen coverage. Peen-scan is comprised of 80% isopropyl alcohol and 20% acrylic material (non-LOX compatible). This shot-peening process has been performed since 1981; tooling, procedures, personnel, materials and chemicals used have remained unchanged. The main housing inspection process has also been consistent; flexible borescope of all accessible areas is performed. Oxygen compatibility/reactivity testing was performed. Peen-scan coated coupons were exposed to a LOX environment at an operating pressure of 5000 pounds per square inch (psi); no reaction was observed. The peen-scan coating was approximately 7 times the amount of material observed in the HPOTP #2029 housing. Impact testing was also performed at 2 kilogram-meter (kg-m) impact energy (calculated impact energy of an 800-micron particle is 0.0002 kg-m at a maximum velocity of approximately 100 feet per second); no reaction was observed on 5 samples tested to 10,000 times the calculated maximum impact energy possible in the HPOTP discharge region.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1 (Continued)	HPOTP main housing contamination during fabrication.	Rationale for STS-43 flight was: • Hot-fire test history indicated this condition was benign; maximum exposure with contaminant present was 51 starts/18,361 seconds, including 5 flights (HPOTP #9311R6). Inspection of 5 housings with contaminant exposed to hot-fire testing revealed no evidence of reactions. • The entire flight family of HPOTPs was processed with similar inspections. • HPOTP #2029 had the largest area of contamination that was found during the normal inspection process. • Material testing confirmed that peen-scan will not react when exposed to LOX at 5000 psi HPOTP operating pressure. • Material testing confirmed that peen-scan will not react at an impact energy four orders of magnitude above the calculated maximum particle impact energy in the HPOTP. ***This risk factor was resolved for STS-43.***

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
2	Undetected cold wall coolant tube leaks. HR No. ME-B7 (All Phases) {C} ***No SSME anomalies attributed to cold wall coolant tube leakage were experienced on STS-43.***	Engine #2107 was leak checked 240° post-STS-37 at Dryden; no leaks were noted during the standard soap leak check. However, when reentry insulation was removed at 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 then 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). The tubes were opened and examined. Cold wall outside diameter through corrosion was found 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, location of leaks was normally at aft manifold stubouts; the leaks were attributed to high-cycle fatigue. The leaks on Engine #2107 were adjacent to the primary component fuel drain line. Rationale for STS-43 flight was: • STS-43/OV-104 Engine #2024 was leak checked 360°; all leaks checks were satisfactory. (No tenth bay insulation is installed on this engine because it is in position #1.) • For the STS-104 engines in positions #2 and #3, Engines #2012 and #2028, insulation was removed after the last flight; no heating to the backside was found.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
2 (Continued)	Undetected cold wall coolant tube leaks.	• The leaks on Engine #2107 were unique, resulting from corrosion caused by residual etchant from the feedline during fabrication. STS-43/OV-104 engines had no feedline replacements. • The leaks on Engine #2107 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 of tube leaks on engine performance. ***This risk factor was resolved for STS-43.***
3	LPOTP #2222 turbine stator vane crack. HR No. ME-C1 (All Phases) {AR} ME-C2 (All Phases) {C} ***No LPOTP anomalies were reported on STS-43.***	A crack was discovered in a turbine stator vane of LPOTP #2222 during a routine dye-penetrant inspection. The crack, 0.125" in length, was located on 1 turbine stator trailing edge in the third row of 5 turbine stators. LPOTP #2222 is a high-time development unit with 59 starts and 24,153 seconds 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 HPOTP downstream of the LPOTP due to loss and migration of a LPOTP turbine stator vane. Cracks in turbine stator vanes were 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

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
3 (Continued)	LPOTP #2222 turbine stator vane crack.	pockets on the stator vane trailing edge root radii. Failure analysis of these cracks led to the determination that they resulted from High-Cycle Fatigue (HCF). The crack located in the #2222 turbine stator vane was the first found 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 is 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. In parallel with the analysis efforts currently underway is an effort to define a new life-limit for the turbine stators [Deviation Approval Request (DAR) 2545 submitted]. The current SSME Program approach to DARs is to set the life-limit to 25% of the operational time of the failed unit. The #2222 turbine stator had 24,253 seconds of operation when the crack was discovered. However, the last inspection of this turbine stator was at 11,724 seconds; no crack was found. Because there is no evidence to the contrary, it is assumed that the crack was initiated at 11,725 seconds; the next second of operation. Setting the DAR limit at 25% of 11,725 (2931 seconds) placed LPOTP #2027R1, installed on engine #2028 (engine #3 on STS-43/OV-104), outside of the DAR limit for flight. DAR limits are, however, also set based on the predicted time of crack initiation derived from metallurgical and fracture analyses.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
3 (Continued)	LPOTP #2222 turbine stator vane crack.	Rationale for STS-43 flight was: • Flight units had a margin >5 (postflight) on seconds compared to the failed unit total time. • Flight units had a margin >2 on seconds compared to the failed unit minimum demonstrated time with no cracks. ***This risk factor was acceptable for STS-43.***
4	Main Combustion Chamber (MCC) debond. HR No. ME-B5 Rev. F {AR} (All Phases) ***No SSME anomalies attributed to MCC debond were reported on STS-43.***	This was the first flight with a known MCC debond. Main Engine (ME)-1, #2024, had a single debond < 3/64" in diameter between nozzle tubes 125 and 126. A Level II waiver to OMRSD V41BUO.031 (that allows no debonds) was approved on February 6, 1991. Rationale for STS-43 flight was: • Analysis predicted the debond was stable; it would grow only to the extent of the marginal debond area. • Worst-case failure is leak before burst, which is protected by LCC redline.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
4 (Continued)	MCC debond.	• Hot-fire debond experience indicated no growth: MCC #2027, 3 starts, 971 sec; MCC #2007, 4 starts, 902 sec; MCC #2022, 24 starts, 11436 sec. • This debond was predicted to have no effect on engine #2024 performance. ***This risk factor was acceptable for STS-43.***
5	Helium precharge valve vent line contamination. ***No anomalies attributed to helium line contamination were reported on STS-43.***	Two line assemblies were found with Turco 4338 residue: 1 from engine #2024 after STS-35, 1 new line assembly at Canoga. Turco 4338 is a scale conditioning solution (LOX compatible) used in the descaling process. The contamination was securely adhered to the base material, less than 0.001" thick. There was no evidence of base material degradation. Worst-case analysis of loss of pneumatic shutdown capability was performed. This would require the contamination to cause jamming of the poppet in the pneumatic shutdown solenoid. If hydraulic shutdown capability is lost and the engine is unable to perform, pneumatic shutdown is the backup shutdown mode. Failure of pneumatic shutdown capability is Crit 1R2 (normal shutdown must fail first).

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
5 (Continued)	Helium precharge valve vent line contamination.	Rationale for STS-43 flight was: • There were no contamination-related anomalies in engine #2024 test history. • Corrosion, particle generation, or LOX compatibility were not issues. • Preflight checkout verified pneumatic shutdown operation. • The estimated probability of loss of pneumatic system shutdown capability was very low. ***This risk factor was resolved for STS-43.***
6	Antiflood Valve (AFV) seal leak. HR No. ME-B3 Rev. F {AR} (All Phases) ME-C3 Rev. F {AR} (All Phases) ***No anomalies attributed to AFV seal leakage were reported on STS-43.***	New build AFVs failed 2 of 5 cryogenic proof cycles (6750 psig He). The AFV RES1257 test port cap seal cracked through (150°) resulting in leakage at the test port. Failure analysis indicated crack propagation to 70% of the leg thickness, probably due to High-Cycle Fatigue (HCF) during machining at the vendor. The crack originated from the internal surface. Finish groove machining on a lathe is the only operation that cuts the internal surface. The suspected cause of the crack was tool chatter during the machining process. Final seal failure was attributed to ductile overload during the cryogenic proof test. The seal leg thickness was also found to be undersized: 0.004" vs a requirement of 0.005" or greater. Twenty seals were returned to the vendor, Sherracin/Harrison, for inspection; 4 seals were found to have undersized legs, but dye-penetrant inspection found no cracks.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
6 (Continued)	AFV seal leak.	Sherracin/Harrison lot #11-9297, the only lot currently in use, comprises a total of 176 seals; 134 seals have been used to date. Seals are installed in a valve room and tested as part of functional testing including proof testing, green run, and/or engine acceptance. AFV seals are reused following visual inspection of the teflon surfaces; no surface defects are allowed that would impair the sealing function. There were no prior problems with these RES1257 seals in the Program history; a total of 485,277 sec with no failures. RES1257 seals are used in 3 additional applications in addition to RES1257-04, the AFV test port seal. • RES1257-03, AFV cap seal: maximum test pressure 6750 ±135 psig; operating pressure 15 psia (R0019349 drain line pressure). • RES1257-01, Helium Precharge Valve (HPV) check valve Pogo side; maximum test pressure 2200 ±40 psig; operating pressure 800 psia (GOX). • RES1257-05, HPV check valve to body helium side; maximum test pressure 2200 ±40 psig; operating pressure 800 psia (helium).

RESOLVED STS-43 SAFETY RISK FACTORS

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

<u>SSME</u>

6 (Continued) — AFV seal leak.

Test history of these valves for STS-43 was:

Engine No.		2024 Starts/Seconds	2012 Starts/Seconds	2028 Starts/Seconds
Valve	AFV	5/2117	4/1368	9/3386
	HPV	19/6992	4/1368	9/3386

Rationale for STS-43 flight was:

- There were no prior failures of RES1257 seals in the Program history (total of 485,277 sec usage to date).
- STS-43 AFV seals were replaced; 6 seals recycled from current lot #11-9297.
- Minimum seal leg thickness was verified.
- Class IVc dye-penetrant inspection showed no cracks.
- STS-43 seals were recoated with teflon (RA1608-001, Class F).
- The seals were screened in use by factory functional testing and engine hot-fire test.

This risk factor was resolved for STS-43.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
7	High-Pressure Fuel Turbopump (HPFTP) thrust ball cracks. HR No. ME-D1 Rev. F {AR} (All Phases) ***No engine HPFTP anomalies were reported on STS-43.***	Four cracks were found on the HPFTP thrust ball #6008 upon disassembly of STS-38 engine #2019 at Canoga Park. The specification indicated no cracks are allowed. These cracks, approximately 0.10-0.15" 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 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 minimize wear; this is allowed by the specification. Wear galling and cracks on the balls are not permitted. The Failure Modes and Effects Analysis (FEMA) 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 was no loss of major pieces of the thrust ball in the Program history.

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
7 (Continued)	HPFTP thrust ball cracks.	Rationale for flight of STS-43 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-43/OV-104 HPFTPs were visually inspected; no cracks were noted. ***This risk factor was resolved for STS-43.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Solid Rocket Booster (SRB) Hydraulic Power Unit (HPU) fuel filter issue. HR No. B-20-21 Rev. C-DCN7 {C} *No SRB anomalies were reported on STS-43.*	An SRB HPU fuel filter, S/N 94, failed the bubble point test at United Space Boosters, Inc., (USBI), Florida Operations, on May 28, 1991. The bubble point measurement was 6.3 inches water, indicating an apparent filter mesh size of 33 microns. The requirement is 8.28 inches water, or 25 micron filter mesh, minimum. The test of S/N 94 was the result of a USBI quality review of Wiltec procedures; Wiltec is the fuel filter vendor. S/N 94 was pulled from stock and sent to Wiltec for retest, where it again failed the bubble point test at 6.4 inches water (32 micron mesh). Prior to continued bubble point testing at Wiltec, the S/N 94 fuel filter was removed from its housing, and again bubble point tested to identify the failure point. A hole in the filter mesh was identified as the cause of the bubble point failure. S/N 94 was originally bubble point tested in 1983 at Wiltec. The original bubble point test result was 12.22 inches water (15-micron mesh). S/N 94 flew on BI-013/STS-41G on October 5, 1984. After STS-41G, the filter was returned to Wiltec for normal postflight inspection and retest before it was shipped to USBI in January 1989 for another flight. Bubble point test results at Wiltec were again good; 13.8 inches (15-micron mesh). S/N 94 was vacuum baked at USBI in late 1989 prior to installation on BI-037/STS-31. After STS-31, S/N 94 was returned to Wiltec for refurbishment and bubble point tested to 13.8 inches water (15-micron mesh) before being sent to USBI in July 1990. Vacuum baking was not performed until May 1991 at USBI. After discovering the hole in the filter mesh at Wiltec, S/N 94 was returned to the Marshall Space Flight Center (MSFC) Materials and Processing (M&P) Laboratory for metallurgical analysis. Analysis determined that the screen and foil material

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1 (Continued)	SRB HPU fuel filter issue.	used to construct the filter were of proper design; 300-series stainless steel. Materials analysis identified intergranular attack at 2 places in the filter mesh adjacent to the foil weld and 1 area of intergranular attack in the foil. Based on the M&P analysis, the most probable failure cause was determined to be extended exposure to cleaning solvents and the delay between cleaning at Wiltec and vacuum baking at USBI. There is a change in work to modify the vacuum bake procedure to preclude the apparent delay between cleaning and baking. The HPU fuel filters on BI-045/STS-43 were processed using the same procedures as S/N 94. Three of the 4 HPU fuel filters were exposed to cleaning solvents for 2 months prior to vacuum baking; the fourth was exposed for 11 months. The SRB Thrust Vector Control (TVC) system, including the HPU, has 3 fuel system filters: the fuel filter with a 25 micron mesh; the Fuel Isolation Valve (FIV) screen, 200 micron mesh at the inlet; and the HPU fuel pump outlet filter with a 25 micron mesh. Particles in the fuel system greater than 200 microns will be trapped at the FIV inlet screen. Particles less than 200 microns can pass through the FIV inlet screen and cause the following to occur: • Prevent FIV closure after SRB separation if a particle lodges in the FIV. This is not considered flight critical because FIV closure occurs after SRB separation. HPU runaway would most likely result if the FIV does not close. • Prevent the SRB HPU from starting if a particle lodged in the HPU fuel pump start bypass valve. This would result in a launch abort.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1 (Continued)	SRB HPU fuel filter issue.	• There would be no adverse effect if a particle less than 200 microns would lodge in the HPU fuel pump gears. Analysis demonstrated that a particle must be greater than 1000 microns to degrade fuel pump performance. • If a particle passes through the HPU fuel pump gears, it should be trapped in the 25-micron fuel pump outlet filter, with no adverse effect. Based on this analysis, the SRB TVC fuel system has sufficient controls to tolerate contaminant particles less than 200 microns after APU start, if the 4 fuel filters similar to S/N 94 were to fail due to corrosion. Worst-case effect of a particle in the system less than 200 microns would be a launch abort. Rationale for STS-43 flight was: • The FIV inlet screen would prevent particles greater than 200 microns from entering the SRB TVC fuel system. • If the 25-micron fuel filter upstream of the FIV inlet screen were to fail, particles less than 200 microns would result in a worst-case launch abort. ***This risk factor was acceptable for STS-43.***

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
2	Right-Hand (RH) SRB forward Booster Separation Motor (BSM) failed Pyrotechnic Initiator Controller (PIC) resistance test. HR No. C-30-01 Rev. B {C} ***No SRB anomalies were reported on STS-43.***	STS-43 RH SRB forward BSM failed the PIC B resistance test during prelaunch checkout. The resistance was above the upper limit allowed for the NASA Standard Initiator (NSI) bridge wire resistance; the specified resistance is 1.25-1.75 ohms. Routine testing indicated an "off" condition; the test should have indicated "on". This is indicative of a high-resistance condition. Further fault isolation was not attempted. The faulty SRB Integrated Electronic Assembly (IEA) was removed and replaced. The removed IEA was bench tested, and the PIC resistance was found to be within the 1.25-1.75 ohm acceptable limits. The newly-installed IEA successfully passed tests (4 repetitions) on the same ordnance circuit that had previously failed in the removed IEA unit. It was observed that there had been a high amount of grease in the connector; this is believed to have been the cause of the high resistance reading in the test of the initially-installed IEA PIC circuit. Rationale for STS-43 flight was: • A new IEA was installed and tested successfully. • There was a high probability that the high resistance problem was caused by grease in the connector. This condition was rectified in the installation of the new IEA. • The PICs are redundant for each BSM cluster. ***This risk factor was resolved for STS-43.***

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Potential for higher than normal erosion and charring in the STS-43 Solid Rocket Motor (SRM) aft exit cone. HR No. BN-07 Rev. C {C} ***No SRM anomalies were reported on STS-43.***	Lower residual volatiles and higher compressive strengths have been noted with the recent (STS-31 and subsequent) carbon cloth phenolic materials relative to the earlier historical data base. Lower permeability and higher compressive strengths have been observed in restart Avtex/NARC carbon/phenolic material. The STS-39 Right-Hand (RH) cowl had the lowest residual volatiles and one of the highest compression strengths in the Shuttle data base. STS-39 RH char/erosion performance was within 2 standard deviations of nominal, and did meet Configuration End Item (CEI) requirements; however, the heat-affected depth was one of the highest that has been observed. Lower residual volatiles in combination with higher compression strengths could lead to ply lift in parallel-to-centerline wrapped parts (cowl and aft exit cone); evaluation of the cowl char and erosion data base shows increased heat-affected depths are possible with cowls having these characteristics. The STS-43 Left-Hand (LH) aft exit cone residual volatiles and compression strength are within the flight and static test data base. However, wash areas and ply lifting have been observed on several static test nozzle aft exit cones (DM-7, ETM-1A, TEM-6, QM-7, PV-1, and FSM-1); flight exit cones are not recovered. FSM-1 was the only aft exit cone with low residual volatiles and higher compression strength. All nozzles performed successfully. Exit cone performance is significantly improved in the Redesigned Solid Rocket Motor (RSRM). The RSRM aft exit cone was thickened to improve margins, and process changes were implemented to reduce liner volatiles content; however, ply lifting still was observed on some RSRM exit cones. STS-43 LH aft cone permeability and compression strengths were near the experience envelope but within the flight and static test database. However, STS-43 (360L017) exit cone performance was predicted to be within the historical data base

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1 (Continued)	Potential for higher than normal erosion and charring in the STS-43 SRM aft exit cone.	and within acceptable limits of the CEI specification. Worst-case assessment, based on STS-39 (360L015) cowl performance, resulted in a 1.35 safety factor (material heat-affected depth increased by 26%). Rationale for STS-43 flight was: • Performance of the FSM-1, TEM-7, and 6 flight exit cones demonstrated that material with low permeability and higher compression strength would perform satisfactorily. • The history base includes 108 exit cone firings without failure, • Worst-case assessment for the STS-43 exit cone, based on STS-39 cowl performance, showed safety factors greater than 1.0. ***This risk factor was acceptable for STS-43.***

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>IUS</u>		
1	Tin whisker problem on Leach relays. ***No Leach relay anomalies were reported on STS-43/IUS-15.***	Tin whiskers on Leach relay terminals and header have caused shorts. Pure tin on the header and terminals can form the whiskers. The Pyro Switching Unit (PSU) was the only Line Replaceable Unit (LRU) with Leach relays that could cause a problem. The PSU was reworked and conformally coated to prevent whisker formation. There was also a potential problem with the fusing whisker forming a plasma cloud that will sustain a short under very special conditions. Analysis limited this potential problem to 6 relays in the Power Distribution Unit (PDU). Testing was conducted to determine the conditions required for plasma formation. This has a potential Inertial Upper Stage (IUS) mission success impact; safety requirements are not affected. Rationale for STS-43 flight was: • The pyro switching unit in IUS-15, which is the only LRU with Leach relays that could cause a problem, was reworked and conformally coated to prevent whisker formation. • Analysis limited a short due to plasma cloud formation by a fusing whisker to mission success impact; no safety requirements were affected by the potential plasma problem. ***This risk factor had a potential impact on mission success for STS-43.***

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>IUS</u>		
2	Failed solder joints in the Redundant Inertial Measurement Unit (RIMU). ***No RIMU anomalies were reported on STS-43/IUS-15.***	During test/checkout at Boeing, RIMU S/N 006 failed. The failure was investigated for possible generic impact. Cracked solder joints were determined to be the cause of the failure. RIMU S/N 006 is a Full-Scale Development (FSD) unit, one of the original production units for the IUS. Several solder joint failures have occurred in FSD units; these failures have been attributed to thermal cycling, the lead forming tool, and the conformal coating process. However, no trends were identified. The lead forming tool and the conformal coating process were changed for the first production unit and subsequent units. One FSD card each has been determined to be in RIMU S/N 14 (prime for IUS-15) and RIMU S/N 24 (backup for IUS-15). Rationale for STS-43 flight was: • The likelihood of an RIMU failure occurrence was determined to be very small. • Failure of an FSD board could cause loss of only 1 gyro, resulting in slight degradation in injection accuracy but well within specification. The 4 other gyros would allow for conduct of a successful mission (only 3 gyros are required).

RESOLVED STS-43 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>IUS</u>		
2 (Continued)	Failed solder joints in the RIMU.	• RIMU S/N 14 had continued to function successfully in the current configuration for 690 hours. This unit had no anomaly history through prelaunch processing. • RIMU performance is continuously monitored from 17 hours prior to liftoff. • Safety requirements were not affected. ***This risk factor had a potential impact on mission success for STS-43.***

SECTION 5

STS-40 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-40/OV-102 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.

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STS-40 INFLIGHT ANOMALIES

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Inertial Measurement Unit (IMU) #2, Serial Number (S/N) 023, failed preflight calibration. IFA No. STS-40-V-01 HR No. ORBI-051 {C} ***No IMU anomalies were reported on STS-43.***	During the second launch attempt of STS-40/OV-102, IMU #2, S/N 023, indicated shifts in accelerometer output data during repeated preflight calibration runs. The data from the first calibration run indicated that the Y-axis accelerometer experienced a 5.54σ excursion before it stabilized. The second calibration run data indicated excursions less than 1σ; within Operational Maintenance Requirements and Specifications Document (OMRSD) limits. Data from a third calibration run indicated similar excursions as the first. OMRSD requirements state that a recalibration will be performed in the event of an accelerometer measurement between 2 and 5σ; over 5σ the IMU is required to be replaced. Launch Commit Criteria (LCC) require all 3 IMUs to be calibrated and functioning properly prior to launch. Because the cause of the 5σ excursions could not be determined, IMU #2, S/N 023, was declared failed, and the launch attempt was scrubbed. IMU #2 was removed and replaced prior to the successful launch of STS-40/OV-102. IMU S/N 023 had a good history of operation, with no similar accelerometer problems. Initial failure analysis at the Johnson Space Center (JSC) Inertial Systems Laboratory (ISL) could not repeat the stability problem encountered during prelaunch calibration runs. Troubleshooting and testing continue. There have been 7 previous inflight IMU anomalies recorded to date. There is no indication that this is a generic IMU problem.

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	IMU #2, S/N 023, failed preflight calibration.	IMUs are Crit 1R2 for erroneous output and Crit 1R3 for loss of output. During flight, Redundancy Management (RM) monitors the 3 IMUs for erroneous accelerometer data and deselects the accelerometer output when limits are exceeded. Rationale for STS-43 flight was: • The IMU system has triple redundancy. • LCC requires 3 good, calibrated IMUs [prior to T–31 seconds (sec)] for launch. • There are flight rules in place to deal with single or double loss of redundancy. The first failure will cause minimum duration of flight. The second failure puts the mission in the next primary landing site posture. ***This risk factor/anomaly was resolved for STS-43.***
2	1307 bulkhead environmental seal and thermal blanket anomalies. IFA No. STS-40-V-02 HR No. ORBI-305A {C} ***No similar anomalies were reported on STS-43.***	Upon opening the Payload Bay Doors (PLBDs) on orbit, a section of the port-side 1307 bulkhead PLBD environmental seal was discovered separated from its retainer and protruding into the Payload Bay (PLB). Three thermal blankets on the 1307 bulkhead were also found partially unfastened, but not completely loose. These anomalies were indicative of air intrusion into the PLB during ascent. Further inspection of the protruding seal determined that it had separated at a repair splice, located at the Y_o = -33" position on the bulkhead. There were 2 protruding pieces; the inboard piece was approximately 22" in long, the outboard

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	1307 bulkhead environmental seal and thermal blanket anomalies.	piece was approximately 8" long. The separated environmental seal is constructed from an Inconel wire spring, with a teflon sheath, approximately 0.75" in diameter, and 64" in length. The seal is attached at the centerline and at the Y_o = -64" position. The concern was that the protruding environmental seal could prevent the port PLBD from fully closing. Extensive analysis, undertaken during the STS-40 mission to determine the potential effects of the protruding seal during PLBD closing, determined that the 8" outboard seal protrusion would not be a problem. The 22" inboard seal, however, was believed to be likely to interfere with the PLBD latch hook and roller assemblies. This would result in the latch not fully locking. In either case, analysis demonstrated sufficient margin in the PLBD drive motor to overcome friction caused by the seal jamming in the interface between the port PLBD and the 1307 bulkhead. Because of the analysis findings, and to test techniques in the event that an Extravehicular Activity (EVA) contingency was required, a demonstration simulating the protruding environmental seal was performed at the Kennedy Space Center (KSC) on OV-103. An astronaut, wearing EVA gloves, performed several tests which included replacing the seal in its retainer and cutting the protruding portions of the seal. These tests were successful in proving that options existed in the event that a contingency EVA was required. Tests were also made to prove analysis results concerning the potential for the seal to impede PLBD closure and

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	1307 bulkhead environmental seal and thermal blanket anomalies.	locking. These test cases demonstrated that the 22" inboard seal protrusion could migrate into the latch and roller assemblies during PLBD closure. However, in all cases the latch assembly was able to overcome the seal to attain the locked position. Analyses were also performed to determine the thermal and structural effects of the 1307 bulkhead seal not being in the proper configuration. Thermal analysis assumed that 63" of the 64" seal was missing during reentry. Results of this analysis predicted bulkhead temperatures resulting from air intrusion and aeroheating to be within allowable limits. Venting analysis performed assuming the entire 64" seal was missing demonstrated positive margins for all associated structural elements. Based on the analyses and the demonstrations performed on OV-103, the decision was made to close the STS-40/OV-102 PLBDs on the nominal timeline for reentry. The PLBDs were successfully closed and locked for reentry, with no indication that the protruding seal interfered with closure. Postflight quick-look inspection indicated that the environmental seal had separated at the inboard side of the repair splice. There was no indication of heat damage in the area of the separation. Further inspection and removal of the seal was performed upon removal of SpaceLab from the PLB. Hands-on inspections of the STS-43/OV-104 1307 bulkhead environmental seals were performed. The OV-104 seals were found to be properly installed in the retainers and were found not to have repair splices similar to those on OV-102 and

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)	1307 bulkhead environmental seal and thermal blanket anomalies.	OV-103. Additionally, tests were performed to determine if the 1307 environmental seals were compressed when the PLBDs were closed. Pictorial evidence and mold impressions demonstrated that the centerline to Y_o = ±64" environmental seals were properly compressed after PLBD closure. Similar tests on OV-102 and OV-103 found that the environmental seals were not compressed. Additionally, there were no indications of air flow (i.e., displaced thermal blankets) on STS-38 and STS-37, the last flights of OV-104. Rationale for STS-43 flight was: • OV-104 1307-bulkhead seals were inspected and found to be properly installed in the retainers. Inspection included visual as well as "hand on" check of installation integrity. Seal compression was verified through mold impressions and photographs. • OV-104 1307-bulkhead seals did not have repair splices similar to those on OV-102 and OV-103. There was no indication of air intrusion on the last 2 OV-104 flights. • Air intrusion into the PLB at the 1307 bulkhead was experienced on several previous flights; loosening of the thermal blankets is not considered to be a problem. ***This risk factor/anomaly was resolved for STS-43.***

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3	Heat erosion of STS-40/OV-102 Right-Hand (RH) External Tank (ET) umbilical door centerline latch. IFA No. STS-40-V-11 HR No. ORBI-302A {AR} ***No ET umbilical door anomalies were reported on STS-43.***	Postflight inspection of STS-40/OV-102 identified significant heat effects (melting) of the RH ET umbilical door centerline latch (forward, outboard). The forward end was found severely discolored and eroded; the erosion had a measured maximum width of 2" that was tapered from forward to aft. The measured depth of the erosion across the latch was 0.1". A 0.180" step was also observed at the mold line between the latch fitting and flow restrictor fingers; this gap (0.10") should be a butt fit. The Thermal Protection System (TPS) tile near the forward and aft latch fitting was also found with signs of erosion around the edges. Further evaluation of these findings determined that the umbilical door pressure seal was not breached. However, the hot gas (air) flow path, determined by heat effects on the structure, entered near the forward outboard corner of the RH ET umbilical door, traversed the full length of the door in the cavity between the thermal barrier and the pressure seal, and exited through a small opening at the aft outboard side of the RH ET umbilical door. The environmental pressure seal exhibited localized surface overtemperature due to hot-gas impingement (approx. ±3/4" from the initial impingement zone). A 0.025-0.030" gap was also found between the sill fitting and structure (into the aft fuselage) on both the RH and Left-Hand (LH) door cavities; the gaps should have been sealed with Room-Temperature Vulcanizing (RTV) by design. The concern associated with this anomaly is the potential for structural damage resulting from excessive thermal effects during reentry and the potential for hot gas to get past the pressure seal into the aft fuselage. However, the door latch fitting appeared to have worked as a heat sink, minimizing thermal effects and resulting in no evident structural damage or degradation. A similar anomaly occurred on STS-1/OV-102 on the same door latch; it was attributed to step-and-gap problems with the TPS.

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3 (Continued)	Heat erosion of STS-40/OV-102 RH ET umbilical door centerline latch.	There are two potential causes for this anomaly: improper step-and-gap due to lack of RTV or damage to the tile/latch fitting as a result of a strike by the centerline latch. There was no clear evidence to isolate this anomaly to one of these potential causes. Review of the processing records indicated that the RTV was properly applied to the latch fitting. The step-and-gap in this area was also recorded as being correct. The melted latch fitting and surrounding tile were to be replaced. Inspection of STS-43/OV-104 ET umbilical doors for improper operation or step-and-gap was performed because of the STS-40/OV-102 discovery. This inspection identified similar, corresponding flow-exit gaps (pinpoint size) on OV-104. Additional Room Temperature Vulcanizing (RTV) was applied to eliminate these gaps. The OV-104 thermal barriers, newly installed during the STS-43 flow, were found to be intact and functioning properly when ET umbilical doors were cycled closed. X-ray examination found no subsurface cracks in the OV-104 latch fittings. Rationale for STS-43 flight was: • STS-43/OV-104 ET umbilical door step-and-gap was inspected and repaired as required. • Mylar pull data was reviewed and met OMRSD requirements. • New thermal barriers were installed on STS-43/OV-104 that meet the design requirements. ***This risk factor/anomaly was resolved for STS-43.***

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4	Auxiliary Power Unit (APU) #1 fuel test line temperature above Fault Detection and Annunciation (FDA) alarm limits. IFA No. STS-40-V-12 HR No. ORBI-104A {C} *No similar anomalies were reported on STS-43.*	During reentry, APU #1 fuel test line temperature reached 99°F, above the 95°F FDA limit. Because of previous APU fuel line heater anomalies, the crew was instructed to turn off the APU #1 fuel line heater. A decrease in temperature was witnessed, however, prior to the heater being commanded off by the crew. The redundant temperature sensor indicated a rise in the fuel line temperature, but did not indicate temperatures above the FDA limit. A similar anomaly was experienced on STS-28, APU #1 fuel test line. This is a Crit 2R3 measurement; however, problems with APU fuel line heaters have increased the sensitivity to exceeding the FDA limit. The FDA limit was increased in early 1990 from 90°F to the current 95°F. Analysis has not yet determined that a further increase is warranted. Rationale for STS-43 flight was: • Redundant temperature sensors are in place on all APUs to verify heater integrity. • APU heaters are redundant and are monitored by onboard FDA and the Mission Control Center during operation. *This risk factor/anomaly was resolved for STS-43.*

STS-40 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5	Cylindrical object debris seen at External Tank (ET) separation. IFA No. STS-40-V-16 HR No. ORBI-302A {AR} ***No debris was reported at ET separation on STS-43.***	OV-102 is the only Orbiter equipped with cameras in the umbilical cavity. During review of ET separation film, a cylindrical object was seen floating away from the Orbiter LH umbilical cavity. Review of separation films determined that the object was an umbilical guide-pin bushing. Two guide pins are used during the ET/Orbiter umbilical mate process. The Inconel 718 bushings are interference-fit into the 2219 aluminum body of the umbilical. The concern was that debris of this type could impede the ET umbilical door from fully closing. Previous analysis of the potential for escaping debris fragments preventing the ET umbilical door from closing determined that the probability is very small. At ET separation, the Orbiter performs a maneuver away from the ET and escaping debris prior to the ET umbilical door closure. This was clearly demonstrated in the STS-40 film of the escaping bushing. Rationale for STS-43 flight was: • Preflight inspection of STS-43/OV-104 umbilicals determined final acceptability for flight. • The probability for debris jamming the ET umbilical door is remote. • ET umbilical doors may be recycled in flight if there is an indication that closing or latching is impeded. • The ET separation burn moves the Orbiter away from escaping debris. ***This risk factor/anomaly was acceptable for STS-43.***

SECTION 6

STS-37 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-37/OV-104 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-37 INFLIGHT ANOMALIES

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Reaction Control System (RCS) primary thruster R1U failed off during External Tank (ET) separation maneuver. IFA No. STS-37-V-01 HR No. ORBI-056 {AR} ***No thruster anomalies were reported on STS-43.***	RCS primary thruster R1U failed off during the STS-37 ET separation maneuver. Recorded pressure data indicated that the thruster Chamber Pressure (P_c) was 10 pounds per square inch absolute (psia); nominal P_c is 150 psia at thruster firing. Thruster injector temperatures indicated some oxidizer and fuel flow in R1U when firing was attempted. R1U was deselected by Redundancy Management (RM) and remained off for the remainder of the mission. There were several similar thruster anomalies of this type on previous missions. The pressure traces from the STS-37 R1U failure were similar to those seen on R3D and R4R on STS-36. Iron nitrate contamination of the oxidizer valve poppet was determined to be the cause of both STS-36 thruster failures. Iron nitrate contamination has been found to form when the oxidizer, Nitrogen Tetroxide (N_2O_4), is allowed to contact moisture in ambient air. This contamination prevented the oxidizer valves from opening in the time allotted to achieve proper P_c. It is believed that iron nitrate contamination may also have caused R1U to fail. Thruster R1U was removed when OV-104 returned to Kennedy Space Center (KSC) and was sent to White Sands Test Facility (WSTF) for further analysis.

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1 (Continued)	RCS primary thruster R1U failed off during ET separation maneuver.	Rationale for STS-43 flight was: • STS-43 RCS thrusters passed all Operational Maintenance Requirements and Specifications Document (OMRSD) tests. • RCS primary thrusters are Crit 1R2 during a Return-To-Launch Site (RTLS) abort and Crit 1R3 during normal flight. ***This risk factor/anomaly was acceptable for STS-43.***

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2	Water Spray Boiler (WSB) #2 did not cool Auxiliary Power Unit (APU) lube oil while under operation of controller "A". IFA No. STS-37-V-02A HR No. ORBI-036 {AR} ***During STS-43 ascent, WSB #2, S/N 018, gave no indication that it was cooling lube oil from APU #2, S/N 208 (IFA No. STS-43-V-02). Lube oil return temperature and gear box forward bearing temperature exceeded the specification limits; APU #2 was shut down immediately after MECO. (See Section 7, Orbiter 1 for more details.)***	WSB #2 failed to cool APU #2 lube oil after the end of the pool boiling period during ascent. WSB #2 was under the operation of controller "A". The crew was directed to switch to controller "B" when lube oil temperature reached 280° Fahrenheit (F); nominal cooling begins at 250°F. Lube oil temperatures had begun to drop just prior to the crew action. The same anomaly occurred during STS-38, the previous flight of OV-104. Both controller "A" and WSB #2 were removed and replaced after STS-38. The most probable cause of this problem was freezing of the spray bar due to wax buildup in the WSB. Wax, in the form of Pentaerythritol, is formed when hydrazine, the APU fuel, is allowed to mix with lube oil. Pentaerythritol will begin to melt when lube oil temperature exceeds 200°F. There were no preflight indications of wax buildup in the APU #2 lube oil or in WSB #2. Research of WSB cooling problems indicated an emerging trend when WSBs are paired with APU Serial Number (S/N) 208. Further investigation into this and other APU/WSB combinations is in work. APU #2/WSB #2 were hot-oil flushed during the STS-43 flow with an improved closed-loop technique. The closed-loop hot-oil flush was used on APU systems for STS-39/OV-103 and STS-40/OV-102. No similar APU/WSB inflight anomalies were reported on either flight.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2 (Continued)	WSB #2 did not cool APU lube oil while under operation of controller "A".	Rationale for STS-43 flight was: • For nominal missions, analysis indicates adequate cooling capability to prevent pre-Main Engine Cutoff (MECO) APU shutdown due to high APU lube oil temperatures. • APU #2/WSB #2 were hot-oil flushed during the STS-43 flow using the improved closed-loop technique. ***This risk factor/anomaly was resolved for STS-43.***

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Power Reactant Supply and Distribution (PRSD) Oxygen (O_2) manifold valve #2 failed to close when commanded. IFA No. STS-37-V-03 HR No. ORBI-094 {AR} ***No PRSD O_2 manifold valve anomalies were reported on STS-43/OV-104. However, PRSD Hydrogen (H_2) manifold #1 isolation valve failed to close when commanded on STS-43 flight day #7 (IFA No. STS-43-V-09); this H_2 isolation valve was the same model as the O_2 isolation valve that failed on STS-37. (See Section 7, Orbiter 3 for more details).***	The PRSD O_2 manifold valve #2 failed to close when commanded the first 2 times on orbit. The valve, S/N 28, finally closed on the third command and was left closed for the remainder of the STS-37 mission. O_2 manifold valve #2 is 1 of 2 redundant valves used to isolate the manifold or PRSD supply tank from a system leak. Failure of both manifold valves to close would result in depletion of fuel cell O_2 reactants and potential loss of the 3 fuel cell power plants. O_2 manifold valve #2, S/N 28, previously failed to close when first commanded on OV-104 during STS-34. Postflight troubleshooting found no problems with the valve operation, and the anomaly was closed as unexplained. S/N 28 worked properly on the 2 OV-104 flights between STS-34 and STS-37: STS-36 and STS-38. Troubleshooting and historical data review determined that there are problems with a type of bulkhead feedthrough connector associated with the O_2 manifold valve #2 operation. Commands to O_2 manifold valve #2 were rewired to another bulkhead connector. If this problem recurs, cryogenics management and O_2 manifold valve #1 provide protection against leakage.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3 (Continued)	PRSD O_2 manifold valve #2 failed to close when commanded.	Rationale for STS-43 flight was: • Commands to O_2 manifold valve #2 were rewired to an alternate connector. • Redundancy and cryogenics management protect against leakage. ***This risk factor/anomaly was resolved for STS-43.***
4	Indications of low P_c on primary thrusters L1U and L1L during interconnect operations. IFA No. STS-37-V-08 HR No. ORBI-056 {C} ***No thruster anomalies were reported on STS-43.***	During STS-38/OV-104, low P_c was experienced on 4 primary thrusters: R1U, R3D, RF3L, and R4U. Postflight troubleshooting did not reveal any thruster leaks or other anomalies which might lead to low thruster P_c. Recent analysis determined that R1U, R3D, RF3L, and R4U all indicated low P_c during interconnect operations; the right pod thruster manifold was interconnected to the left Orbital Maneuvering System (OMS) propellant tanks. When the RP03 propellant source was switched back to the straight feed configuration from the right OMS propellant tanks, thruster P_c in R1U, R3D, RF3L, and R4U returned to nominal. This finding led to the decision to perform thruster firings on STS-37/OV-104 in the interconnect configuration. When this was performed, thrusters L1U and L1L showed degraded P_c, approximately 130 psia instead of 150 psia nominal. P_c in L1U and L1L returned to nominal after reconfiguration to straight feed. It was believed that there might be contamination in the oxidizer interconnect line. No further action was taken at KSC.

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4 (Continued)	Indications of low P_c on primary thrusters L1U and L1L during interconnect operations.	Rationale for STS-43 flight was: • All thrusters on STS-43/OV-104 passed OMRSD checkouts. • The low thruster P_c problem, if related to the interconnect issue and experienced on STS-43/OV-104, is a mission impact and not a safety concern. • RCS thrusters are Crit 1R2 for RTLS abort and Crit 1R3 for other mission phases. ***This risk factor/anomaly was resolved for STS-43.***
5	Backup Flight Software (BFS) navigation initialization anomaly. IFA No. STS-37-V-09 HR No. ORBI-066 {AR} ***This anomaly did not recur on STS-43.***	Postlaunch data analysis of the BFS telemetry indicated that, from the prelaunch BFS OPS-1 transition until the T-8 second (sec) BFS reinitialization, the Z-component (altitude) of the BFS state vector was increasing at a rate of approximately 1 foot per second (ft/sec) to approximately 7700 feet (ft). The BFS navigation errors were cleared at the T-8 sec point in the launch countdown when the BFS was reinitialized to the pad B position. Ascent telemetry review indicated that both the BFS and the Primary Avionics Software System (PASS) performed nominally.

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	BFS navigation initialization anomaly.	Data review found that the Z-component was approximately 250 ft at the OPS-1 transition plus 3-minute (min) point; previous flight data indicated the Z-component to be 6 to 30 ft during the same time in the launch countdown. During previous launch countdowns, Z-component errors of 1500 to 3000 ft were observed up to the T-8 sec period. Recent pre-STS-37 tests conducted in the Shuttle Avionics Integration Laboratory (SAIL) with the new General Purpose Computers (GPCs) demonstrated errors of 4000 to 4500 ft. Errors are believed to be caused by gravity feedback and this eventually leads to error growth in the Z-component. The investigation included a code audit of the BFS and PASS. This audit determined that the problem was only in the first initialization of the navigation function in the BFS. The PASS navigation function did not have the same problems. Testing of the BFS at SAIL repeated the STS-37 prelaunch anomaly on each attempt and demonstrated that the problem was not from the Inertial Measurement Unit (IMU) input. It is believed that this anomaly was caused by the increased processing time of the new GPCs combined with the way the BFS sequences initialization of the navigation function. This software problem has been determined to be limited to the prelaunch initialization of the BFS. Reinitialization of the BFS at T–8 sec clears any problems created during the first BFS initialization. Discrepancy Report (DR) 106197 was generated to identify this BFS navigation initialization problem for resolution. This resolution will not be implemented prior to STS-43 and may potentially recur. Recurrence is not a safety concern.

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	BFS navigation initialization anomaly.	Rationale for STS-43 flight was: • The anomaly was isolated to prelaunch initialization of the BFS navigation function only and is cleared at the T-8 sec BFS reinitialization. • Analysis and SAIL testing demonstrated no similar anomalies with the PASS. ***This risk factor/anomaly was resolved for STS-43.***
6	WSB #3 overcool during entry. IFA No. STS-37-V-12 HR No. ORBI-036 {AR} ***No WSB overcooling was experienced on STS-43. However, there was a failure to cool APU #2 lube oil during STS-43 ascent (IFA No. STS-43-V-02). (See Section 7, Orbiter 1 for more details.)***	System #3 post-Main Engine Cutoff (MECO) APU/WSB lube oil temperature ran cooler than the other two systems; lube oil return temperature was 231°F minimum. System #3 saw a second overcool during entry; lube oil return temperature was 211°F minimum. System #3 APU/WSB showed evidence of contamination. High gearbox pressure was observed during the pad APU confidence run. The filter and lube oil were changed out prior to flight. System #3 APU/WSB were hot-oil flushed with the closed-loop technique during the STS-43 flow. The closed-loop hot-oil flush was also used on APU systems for STS-39/OV-103 and STS-40/OV-102. There were no similar APU/WSB anomalies reported during either mission.

STS-37 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6 (Continued)	WSB #3 overcool during entry.	Rationale for STS-43 flight was: • For nominal missions, analysis indicated adequate cooling capability to prevent pre-MECO APU shutdown due to high APU lube oil temperatures. • APU #3/WSB #3 were hot-oil flushed using the improved closed-loop technique ***This risk factor/anomaly was resolved for STS-43.***

SECTION 7

STS-43 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-43/OV-104 mission. 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.

SECTION 7 INDEX

STS-43 INFLIGHT ANOMALIES

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	No cooling on Water Spray Boiler (WSB) #2 on ascent. IFA No. STS-43-V-02 HR No. ORBI-121 {AR}	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 was confirmed failed, the decision was made to invoke Flight Rule 10-23, allowing the 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 hydraulic spray valves were removed and replaced after troubleshooting at Kennedy Space Center (KSC). APU/WSB system #2 will be hot-oil flushed prior to the next flight of OV-104, STS-44.

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1 (Continued)	No cooling on WSB #2 on ascent.	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 to be 279°F. The Operational Maintenance Requirements and Specifications Document (OMRSD) requires a hot oil flush of the APU/WSB if the 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.
2	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}	PRSD H_2 tank #1, heater "B" failed off on Flight Day (FD) #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 operation 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" to adversely impact consumable margins. No further PRSD heater problems were experienced during the remainder of the flight. Troubleshooting efforts at KSC found 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" operated nominally. Laboratory analysis indicated that the fuse was defective; this was not considered to be a generic failure problem.

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3	Power Reactant Supply and Distribution (PRSD) Hydrogen (H_2) manifold #1 isolation valve failed open. IFA No. STS-43-V-09 HR No. ORBI-286A {AR}	PRSD H_2 manifold #1 isolation valve failed to close when commanded on FD #7. This 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 KSC troubleshooting. Instrumentation of the valve command circuit was approved for the next flight of OV-104.

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4	APIJ #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}	Postflight evaluation of APU #1, Serial Number (S/N) 305 data identified the potential for Pulse Control Valve (PCV) leakage for a period of 35 sec during reentry. This was demonstrated by the GG chamber pressure traces returning to a non-zero level after each pulse. Data review indicated that 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 would have required crew action. A likely cause of this anomaly was initially thought to be transient contamination that prevented the PCV from completely closing. The concern with this anomaly is 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. Sundstrand found that the PCV and Shutoff Valve (SOV) were leaking beyond specification due to a known age/cycle related failure mode of the valve seats.

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5	Main Propulsion System (MPS) Liquid Hydrogen (LH_2) 4-inch disconnect portion of seal stuck in flapper. IFA No. STS-43-V-13 HR No. INTG-041 {AR} ORBI-035A {AR}	Postlanding inspection of STS-43/OV-104 identified an audible leak emanating from the LH_2-side 4-inch disconnect. On closer inspection, 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 with 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 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

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
5 (Continued)	MPS LH_2 4-inch disconnect portion of seal stuck in flapper.	3 pounds-mass (lbm) of hydrogen in the system when the ET umbilical doors close [MECO + 1 minute (min)]. 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, indicates that the aft compartment helium purge is not adequate to eliminate the risk of a hydrogen fire. However, 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 second (sec); Major Mode 602 (MM602) + 30 sec]. Since RTLS dump flow paths 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 entry, presents the potential for a flammability hazard. A similar blowing leak through the 4-inch flapper valve was experienced on STS-26/OV-103. Troubleshooting determined that this problem was attributed to improper flapper valve shimming. It was later determined that 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.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	MPS LH_2 4-inch disconnect portion of seal stuck in flapper.	Several options to mitigate the residual risk for the RTLS contingency have been considered. One option is 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 the 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 is some concern with installing the software patches required to implement this option. A second option requires the crew to manually set the 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 has been estimated to be less than 3 lbm. Analysis of this option led to the determination that it 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

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)	MPS LH_2 4-inch disconnect portion of seal stuck in flapper.	to MM602 and MM602 + 20 sec. This action returns LO_2 dump to normal software control. The proposed single-step and the 2-step manual procedures have been validated in the Shuttle Mission Simulator (SMS). 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 in time for the STS-44/OV-103 mission. Analysis of potential adverse affects 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 entry. At 56,000 feet (ft) altitude, residual hydrogen in the manifold would yield 10-11% hydrogen concentration 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 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 decision to incorporate the second procedural step is pending.

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
6	Right-Hand (RH) outboard brake pressure bias. IFA No. STS-43-V-14	Postflight data review indicated that RH outboard brake pressure #4 appeared biased 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 pressure anomaly did not adversely affect the Orbiter rollout distance. A similar anomaly occurred on the previous flight of OV-104, STS-37, when the RH outboard brake was approximately 100 psia low. Post-STS-37 troubleshooting could not duplicate the problem. The OV-104 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 servo-valve (LV24). No problems were experienced during the previous flight of OV-103, 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 servo valve. The OV-104 brake servo valve module was removed and replaced, and will undergo failure analysis.

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}	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 valve, it was discovered that both tangs at the end of the valve springs were missing. Visual inspection 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 bleed/check valve has been 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 would be ingested by the engine with no adverse consequence. The only potential for a Crit 1 effect on the SSMEs is in the event that an engine 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 cause concern.

STS-43 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
1	SSME #3 controller Digital Computer Unit (DCU) failure on channel "A". IFA No. STS-43-E-01 HR No. INTG-165B {AR}	Engine #2028, controller F21, experienced a DCU channel "A" hold without a power loss prior to the start of 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 A halt. The decision was made to replace the discrepant controller with a spare (F18), and the launch was scrubbed. 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 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 MCE. 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.

APPENDIX A

LIST OF ACRONYMS

AFV	Antiflood Valve
AMOS	Air Force Maui Optical Site
AOA	Abort-Once-Around
APE	Auroral Photography Experiment
APU	Auxiliary Power Unit
AR	Accepted Risk
ARC	Ames Research Center
ASE	Airborne Support Equipment
ATO	Abort-To-Orbit
ATP	Acceptance Test Procedure
BFS	Backup Flight Software
BIMDA	Bioserve Instrumentation Technology Associates Materials Dispersion Apparatus
BITE	Built-In Test Equipment
BSM	Booster Separation Motor
Btu	British Thermal Unit
C	Controlled
CAR	Corrective Action Request
CEI	Configuration End Item
CG	Center of Gravity
CPU	Central Processing Unit
DAR	Deviation Approval Request
DC	Direct Current
DCU	Digital Computer Unit
DR	Discrepancy Report
DTO	Detailed Test Objective
EAFB	Edwards Air Force Base
ECLSS	Environmental Control and Life Support System
ECP	Engineering Change Proposal
EDM	Electrical Discharge Machining
EDT	Eastern Daylight Time

APPENDIX A

LIST OF ACRONYMS - CONTINUED

EI	Entry Interface
EIU	Engine Interface Unit
EOM	End-Of-Mission
ET	External Tank
EVA	Extravehicular Activity
F	Fahrenheit
FA	Flight Aft
FASCOS	Flight Acceleration Safety Cutoff System
FCOD	Flight Crew Operations Directorate
FCS	Flight Control System
FCV	Flow Control Valve
FD	Flight Day
FDA	Fault Detection and Annunciation
FID	Failure Identification
FIV	Fuel Isolation Valve
FMEA	Failure Modes and Effects Analysis
FMEA/CIL	Failure Modes and Effects Analysis/Critical Items List
FR	Flight Rule
FRR	Flight Readiness Review
FRSI	Felt Reusable Surface Insulation
FSD	Full-Scale Development
ft	Feet
ft-lb	Foot-Pounds
ft/sec	Feet Per Second
g	Gravitational Acceleration
GAS	Get-Away-Special
GG	Gas Generator
GOX	Gaseous Oxygen
GPC	General Purpose Computer
H_2	Hydrogen
HCF	High-Cycle Fatigue
He	Helium
HPFTP	High-Pressure Fuel Turbopump
HPICS	Heat Pipe Instrumentation and Control System
HPOTP	High-Pressure Oxidizer Turbopump
HPRS	Heat Pipe Radiator System
HPU	Hydraulic Power Unit

APPENDIX A

LIST OF ACRONYMS - CONTINUED

HPV	Helium Precharge Valve
HR	Hazard Report
I/O	Input/Output
IBM	International Business Machines
IEA	Integrated Electronic Assembly
IFA	Inflight Anomaly
IMU	Inertial Measurement Unit
INTG	Integration
IPL	Initial Program Load
IPMP	Investigation into Polymer Membranes Processing
ISL	Inertial Systems Laboratory
ITA	Instrumentation Technology Associates
IUS	Inertial Upper Stage
JAEL	Johnson Space Center Avionics Engineering Laboratory
JSC	Johnson Space Center
kg-m	Kilogram-Meter
KSC	Kennedy Space Center
LACE	Low-Power Atmospheric Compensation Experiment
LaRC	Langley Research Center
lb	Pound
lbm	Pounds-Mass
LCC	Launch Commit Criteria
LH	Left-Hand
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LOX	Liquid Oxygen
LPFTP	Low-Pressure Fuel Turbopump
LPOTP	Low-Pressure Oxidizer Turbopump
LRU	Line Replaceable Unit
LSFR	Launch Site Flow Review
L-2	Launch Minus 2 Day
M&P	Materials and Processing
MCC	Main Combustion Chamber
MCF	Major Component Failure
MDA	Materials Dispersion Apparatus

APPENDIX A

LIST OF ACRONYMS - CONTINUED

MDF	Minimum Duration Flight
MDM	Multiplexer-Demultiplexer
ME	Main Engine
MEC	Main Engine Controller
MECO	Main Engine Cutoff
min	Minute
MLG	Main Landing Gear
MM602	Major Mode 602
MOD	Mission Operations Directorate
MPS	Main Propulsion System
MR	Material Review
ms	Millisecond
MSBLS	Microwave Scanning Beam Landing System
MSE	Mission Safety Evaluation
MSFC	Marshall Space Flight Center
N_2	Nitrogen
N_2O_4	Nitrogen Tetroxide
NASA	National Aeronautics and Space Administration
NM	Nautical Mile
NOAA	National Oceanic and Atmospheric Administration
NSI	NASA Standard Initiator
NSRS	NASA Safety Reporting System
NWS	Nose Wheel Steering
O_2	Oxygen
OCTW	Optical Communication Through the Shuttle Window
OMI	Operations and Maintenance Instructions
OMRSD	Operational Maintenance Requirements and Specifications Document
OMS	Orbital Maneuvering System
OPF	Orbiter Processing Facility
OPS	Operations Sequence
ORBI	Orbiter
OSMQ	Office of Safety and Mission Quality
OV	Orbiter Vehicle
P/N	Part Number
PAR	Prelaunch Assessment Review
PASS	Primary Avionics Software System
P_c	Chamber Pressure

LIST OF ACRONYMS - CONTINUED

PCF	Protein Crystal Facility
PCG	Protein Crystal Growth
PCV	Pulse Control Valve
PDU	Power Distribution Unit
PFS	Primary Flight System
PIC	Pyrotechnic Initiator Controller
PLB	Payload Bay
PLBD	Payload Bay Door
POR	Power-On Reset
PR	Problem Report
PRCB	Program Requirements Control Board
PRSD	Power Reactant Supply and Distribution
psi	Pounds Per Square Inch
psia	Pounds Per Square Inch Absolute
psig	Pounds Per Square Inch Gage
PSIG	Propulsion Systems Integration Group
PSU	Pyro Switching Unit
R/IM	Refrigerator/Incubator Module
RCS	Reaction Control System
RH	Right-Hand
RHC	Rotational-Hand Controller
RIMU	Redundant Inertial Measurement Unit
RM	Redundancy Management
RMS	Remote Manipulation System
RPC	Remote Power Controller
RSRM	Redesigned Solid Rocket Motor
RTLS	Return-To-Launch-Site
RTV	Room-Temperature Vulcanizing
S/N	Serial Number
SAA	Single Access Antenna
SAIL	Shuttle Avionics Integration Laboratory
SAMS	Space Acceleration Measurement System
SBUV	Solar Backscatter Ultraviolet
sccs	Standard Cubic Centimeters Per Second
scfm	Standard Cubic Feet Per Minute
scim	Standard Cubic Inches Per Minute
SDIO	Strategic Defense Initiative Organization
sec	Second

APPENDIX A

LIST OF ACRONYMS - CONTINUED

SEM	Scanning Electron Microscope
SGL	Space to Ground Line
SHARE	Space Station Heat Pipe Advanced Radiator Element
SLF	Shuttle Landing Facility
SMS	Shuttle Mission Simulator
SOV	Shutoff Valve
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSBUV	Shuttle Solar Backscatter Ultraviolet Experiment
SSCE	Solid Surface Combustion Experiment
SSME	Space Shuttle Main Engine
SSRP	System Safety Review Panel
TAEM	Terminal Area Energy Management
TAL	Transatlantic Abort Landing
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TIG	Time of Ignition
TPCE	Tank Pressure Control Experiment
TPS	Thermal Protection System
TVC	Thrust Vector Control
USBI	United Space Boosters, Inc.
UVPI	Ultraviolet Plume Instrument
VDT	Vehicle Data Table
WPAFB	Wright Patterson Air Force Base
WSB	Water Spray Boiler
WSTF	White Sands Test Facility