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MISSION SAFETY EVALUATION REPORT FOR STS-33

Postflight Edition: March 15, 1990

Safety Division

Office of Safety, Reliability, Maintainability, and Quality Assurance

National Aeronautics and Space Administration

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MISSION SAFETY EVALUATION

REPORT FOR STS-33

Postflight Edition: March 15, 1990

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EXECUTIVE SUMMARY

Discovery, with a Department of Defense (DoD) payload, was successfully launched on November 22, 1989, at 7:23 p.m. Eastern Standard Time (EST) from Kennedy Space Center (KSC), Florida. This was the first post-Challenger night launch. The Solid Rocket Booster separation could be clearly seen nearly 30 miles over the Atlantic, and the bright pinpoints of light from the Orbiter's main engines were visible by long-range tracking cameras almost to the time of engine cutoff.

The landing was delayed one day because of strong wind gusts across the dry lakebed runway at Edwards Air Force Base (EAFB), California. The next day, *Discovery's* deorbit burn was again bumped back, by 90 minutes, to allow the winds to subside to within acceptable limits. The engines were then fired to begin the 1-hour 20-minute deorbit to Earth; this was longer than usual due to the high orbital altitude on this flight. Because the crosswinds on the lakebed were still beyond acceptable limits, even though the strong gusts had subsided, *Discovery* was ordered en route to use one of the EAFB concrete runways. *Discovery* landed at 7:31 p.m. EST on Monday, November 27, 1989.

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, Reliability, Maintainability, and Quality Assurance (SRM&QA) 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. Unresolved safety risk factors impacting STS-33 flight were also documented prior to the STS-33 Flight Readiness Review (FRR) (FRR Edition) and prior to the STS-33 Launch Minus Two Day (L–2) Review (L–2 Edition). This final postflight edition evaluates performance against safety risk factors identified in 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, Reliability, Maintainability, and Quality Assurance (SRM&QA) 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-33 safety risk factors that represent a change from previous flights, factors from previous flights that have 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 brief mission description, including launch data, crew size, mission duration, launch and landing sites, and other mission-related information.

Section 3 - Contains a list of safety risk factors/issues, considered resolved or not a safety concern prior to STS-33 launch, that were impacted or repeated by anomalies reported for the STS-33 flight.

Section 4 - Contains a list of safety risk factors that were considered resolved for STS-33.

Section 5 - Contains a list of Inflight Anomalies (IFAs) that developed during the STS-34 mission.

Section 6 - Contains a list of IFAs that developed during the STS-29 mission.

Section 7 - Contains a list of IFAs that developed during the STS-33 mission. Those STS-33 IFAs which are considered to represent safety risks will be addressed in the MSE for the next Space Shuttle flight.

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

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

SECTION 2

STS-33 MISSION SUMMARY

2.1 Summary Description of STS-33 Mission.

Space Shuttle *Discovery*, with a Department of Defence (DoD) payload, was successfully launched at 7:23 p.m. Eastern Standard Time (EST) on November 22, 1989, from Pad 39B at Kennedy Space Center (KSC). STS-33 was the first post-Challenger night launch. The Solid Rocket Booster (SRB) separation could be clearly seen over the Atlantic, and the bright pinpoints of light from *Discovery's* main engines were visible to the naked eye nearly 600 miles down range and almost to Main Engine Cutoff (MECO) by long-range tracking cameras.

Ascent performance was nominal. Anomalies reported on STS-33 were the least of any post-Challenger era flights. Of the Orbiter anomalies reported, a cabin pressure leak through the Waste Collection System (WCS) was the most critical. A Gaseous Nitrogen (GN_2) leak in Water Spray Boiler (WSB) #1 and high temperature oscillations in Flash Evaporator System (FES) "B" were anomalies experienced on previous *Discovery* missions.

The STS-33 launch film showed stud hang-up at Holddown Post (HDP) #3 similar to that experienced during the STS-34 launch. Broaching and loss of epon shim material were also experienced at HDP #3. Thread impressions were visible on the forward side of the same hole. One of the two pyrotechnic charges used on each frangible nut did not appear to explode properly, as indicated by ductile separation of the frangible nut. The potential problem experienced with skewed firing of the frangible nut pyrotechnic charge identified the need for further analysis relative to the influence and contribution of bolt hang-up and liftoff performance degradation. Marshall Space Flight Center (MSFC) and Rockwell International (RI) analyses indicate that all 8 HDP bolts could hang-up with no deleterious liftoff performance effects, provided that all frangible nuts are released.

Postflight inspection of STS-33 SRBs found connectors that were improperly torqued and lockwired. These anomalies resulted in no adverse or safety-of-flight conditions.
Corrective procedures and awareness training are now in place at KSC to correct problems with connector/fastener installations.

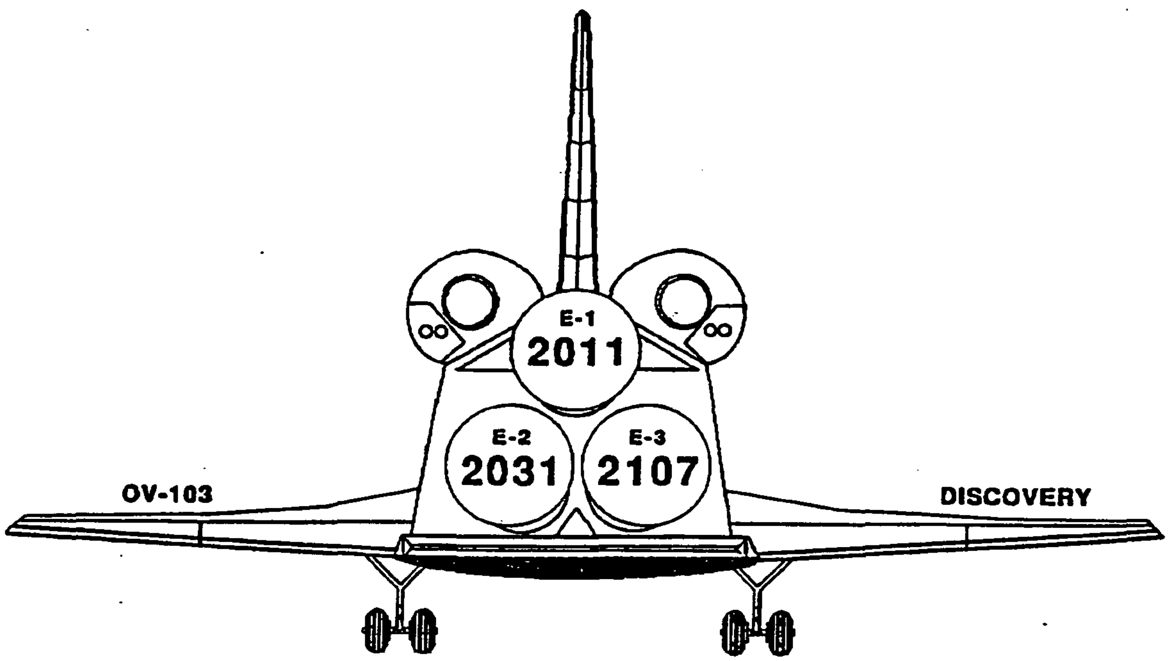
Postflight inspection of *Discovery's* main engines revealed two anomalous conditions. First, inspection at Dryden revealed damage to the engine heat shield blankets. Engines #2011 and #2107 sustained typical flight damage, with torn blankets and some pillow damage. Engine #2031 damage was considered unique in that the 1" bumper strip that attaches the heat shield blanket broke loose. The detached blanket dropped down on top of the nozzle. The flexible seal beneath the blanket was exposed; however, further inspection revealed that the seal was only slightly discolored and that it was still flexible. Postflight inspection of engine #2107 also found that the nozzle was discolored, or "blued", on the front face of the aft manifold. Discoloration in this uninsulated nozzle region has not been experienced on previous flights. No discoloration was seen on engine #2031 nozzle. It was also noted that this was a new nozzle, which had never flown.

There were no Solid Rocket Motor (SRM) and no External Tank (ET) anomalies reported on STS-33.

The landing was delayed one day because of strong wind gusts across the dry lakebed runway at Edwards Air Force Base (EAFB), California. The next day, November 27, 1989, *Discovery's* deorbit burn was again delayed, by one orbit, to allow the winds to subside to within acceptable limits. Reaction Control System (RCS) thrusters and engines were fired to begin the 1-hour 20-minute deorbit to Earth. Because the crosswinds on the lakebed runway were still beyond acceptable limits, *Discovery* was ordered en route to land on a concrete runway at EAFB. Landing occurred at 7:31 p.m. EST, after 77 orbits.

2.2 Flight/Vehicle Data

- Launch Date: November 22, 1989
- Launch Time: 7:23 p.m. EST
- Launch Site: KSC Pad 39B
- RTLS: Kennedy Space Center, Runway 33
- TAL Site: Ben Guerir
- Alternate TAL Site: None
- Landing Site: Edwards AFB, CA, Concrete Runway
- Landing Date: November 27, 1989
- Landing Time: 7:31 p.m. EST
- Mission Duration: 5 Days, 8 Minutes
- Crew Size: 5
- Inclination: Classified
- Altitude: Classified
- Orbiter: OV-103 (9) *Discovery*
- SSMEs: (1) #2011, (2) #2031, (3) #2107
- ET: ET-038
- SRBs: BI-034
- SRMs: RSRM Flight Set #7



ENGINE	#2011	#2031	#2107
POWERHEAD	#2016	#2019	#2014
MCC*	#4005	#2019	#4002
NOZZLE	#4016	#4017	#4019
CONTROLLER	F24	F27	F11
FASCOS*	#17	#19	#29
HPFTP*	#4012	#6102	#4011
LPFTP*	#82207	#2120	#2117
HPOTP*	#4205	#6402	#2422
LPOTP*	#2126	#2120	#4105

* Acronyms can be found in Appendix A.

2.2 Payload Data

The payload was classified.

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

SECTION 3

SAFETY RISK FACTORS/ISSUES IMPACTED BY STS-33 ANOMALIES

This section lists safety risk factors/issues, considered resolved (or not a safety concern) for STS-33 prior to launch (see Sections 4, 5, 6, and 7), that were repeated or related to anomalies that occurred during the STS-33 flight. The list indicates the section of this Mission Safety Evaluation (MSE) Report in which the item is addressed, the item designation (Element/Number) within that section, a description of the item, and brief comments concerning the anomalous condition that was reported.

ITEM		COMMENT
Section 4: Resolved Safety Risk Factors		
Orbiter 2	OV-104 Star Tracker Door stalled during closure.	The Y-Star Tracker Door thermal blanket was found totally detached from the door and lying loose on the bottom of the Star Tracker cavity during STS-33/OV-103 postflight inspection (IFA No. STS-33-11). A small tear on the top of the blanket indicated that it was detached when the door closed. It did not, however, interfere with the door operation. Rockwell International (RI) analysis indicated that the thermal blankets in the Star Tracker cavity are not necessary. RI recommended the removal of these blankets because of the redefined thermal environment. The Level II Program Requirements Control Board (PRCB) directed removal of the thermal blankets from all vehicles.
Section 5: STS-34 Inflight Anomalies		
Orbiter 8	Auxiliary Power Unit (APU) #2 fuel pump heater "B" cycling high.	A similar anomaly (IFA No. STS-33-16) was reported on APU #1 and APU #3 bypass line "A". Temperature sensors on the bypass line of both APUs demonstrated erratic behavior. It is believed that these lines experienced vibration which led to loosening of the temperature sensor mounts. Temperature sensors on APU #1 bypass lines "A" and "B" were replaced. Replacement of APU #3 temperature sensors was deferred until after STS-31, when APU #3 will be replaced.

ITEM		COMMENT
Section 5: STS-34 Inflight Anomalies		
SRB 1	Right Solid Rocket Booster (SRB) Holddown Post (HDP) #2 broached and shoe lifted from Mobile Launch Platform (MLP) during liftoff.	The STS-33 launch film showed the stud at HDP #3 hung-up, similar to the occurrence on STS-34 (IFA No. STS-33-B-01). Broaching also occurred at HDP #3 (IFA No. STS-33-B-02). Thread impressions were also visible on the forward side of the same hole. One of the two pyrotechnic charges used on each frangible nut did not appear to explode properly, as indicated by ductile separation of the frangible nut. The potential problem experienced with skewed firing of the frangible nut pyrotechnic charge identified the need for further analysis relative to the influence and contribution of the bolt hang-up at HDP #3 and liftoff performance degradation. Marshall Space Flight Center (MSFC) and RI analysis indicated that all 8 HDP bolts could hang-up with no deleterious liftoff performance effects, provided that all frangible nuts are released.
KSC 1	Connectors on SRBs improperly torqued and lockwired.	During postflight inspection, 2 cable connectors were found incorrectly installed, and 2 ground straps were found loose due to omitted washers (IFA Nos. STS-33-K-01, STS-33-K-02, and STS-33-K-03). The Right-Hand (RH) forward skirt Range Safety System Ground Support Equipment cable was found not fully seated on its mating connector. This cable is not used in flight, but is used during range safety ground checkout. The left-hand upper strut separation ordnance connector was found finger-loose, but correctly lockwired. Slack in the lockwire indicated that the connector had not

	ITEM	COMMENT
<u>Section 5: STS-34 Inflight Anomalies</u>		
KSC 1 (Continued)		been properly torqued prior to lockwire installation. Two ground strap fasteners were found bottomed out due to the omission of washers on the RH SRB aft Integrated Electronics Assembly bracket. Efforts continue to ensure proper connector/fastener installation. No adverse conditions resulted from these anomalies.
<u>Section 6: STS-29 Inflight Anomalies.</u>		
Orbiter 6	Water Spray Boiler (WSB) #1 exceeded specification leak rate.	During on-orbit operations, the WSB for hydraulic system #2 demonstrated excessive Gaseous Nitrogen (GN_2) leakage (IFA No. STS-33-17). This was similar to the GN_2 leak experienced on OV-103/STS-29. The specified leak rate was exceeded on STS-33 by 0.06 pounds per square inch/hour (psi/hr).
Orbiter 9	Flash Evaporator System (FES) primary controller "B" outlet oscillation.	During deorbit preparations, FES "B" was shut down because it was above specified temperature limits (STS-33-13). This was similar to the occurrence on STS-29. Prior to STS-33, the midpoint temperature sensors were repacked due to a lag that existed between the midpoint temperature sensor and the actual Freon Coolant Loop temperatures. This should have rectified the problem. After the first occurrence of this anomaly on STS-33, FES "B" was recycled, which brought the temperature into the specified limit. The STS-33 anomaly is believed to have been caused by a tolerance build-up in the lead/lag times of Controller "B" and its 3 temperature sensors.

SECTION 4

RESOLVED STS-33 SAFETY RISK FACTORS

This section contains a listing of the safety risk factors that were considered resolved for STS-33. These items were reviewed by the NASA safety community. A description and information regarding problem resolution are provided for each safety risk factor. The safety position with respect to resolution is based on findings resulting from System Safety Review Panel (SSRP) and Program Requirements Control Board (PRCB) reviews (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.

SECTION 4 INDEX

INTEGRATION

1	Space Ordnance Systems NASA Standard Initiator lot test failures.
2	Potential procurement of incorrect sheet metal for Space Shuttle Program Element subassemblies.
3	Auxiliary Power Unit fuel pump failed during fuel pump calibration at Sundstrand.
4	Unitrode diodes alert.
5	Integrity of loctite and primer installed on External Tank and Solid Rocket Booster threaded fasteners was suspect.

ORBITER

1	OV-103 has one Multiplexer-Demultiplexer that may contain Erie capacitors which are prone to failure.
2	OV-104 Star Tracker Door stalled during closure.
3	Potential tire failure due to undetected delaminations after preroll.
4	Orbital Maneuvering System fuel tank weld crack in communication screen.
5	Excessive body flap deflection during ascent.
6	Reaction Control System left helium valve "A" failed.
7	Fuel cell separator plate plating defects.
8	17" disconnect main actuator drive link bearing cracks.
9	Kemet capacitor.
10	Loose connector backshells.
11	2" Gaseous Oxygen disconnect bolt problem.
12	Safety wire on Vernier Thruster Valve found missing during vendor inspection.
13	Display Electronics Unit #2 failure.
14	Multiplexer-Demultiplexer Flight Forward #1 failed Built-In-Test-Equipment test.

SECTION 4 INDEX - (Continued)

SECTION 4 INDEX - (Continued)

ET

1 Liquid Hydrogen feedline vacuum jacketed bellows subassembly failed preacceptance leak tests performed during fabrication.

GFE

1 Crew parachute pyrotechnic cable cutter bent pins.

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Space Ordnance Systems (SOS) NASA Standard Initiator (NSI) lot test failures. HR No. INTG-052B C-30-12 Rev. A *No NSI anomalies were reported on STS-33.*	One-hundred test units from SOS NSI lot MDE were subjected to 20°F temperature cycles and random vibration at -260°F as part of lot acceptance testing. Subsequent to environmental testing, a bridgewire resistance check was performed. One unit had a bridgewire resistance of 1.74 ohms; specification is 1.05 ± 0.10 ohms. A second unit showed an open circuit during the bridgewire resistance check. As a result of these anomalies, the lot certification effort was postponed. The failed units were returned to SOS for failure analysis. Representatives from Johnson Space Center (JSC) Quality Assurance, Reliability, and Safety Office (QARSO), Range Safety Office, and Pyro Engineering organizations witnessed the failure investigation at SOS. To inspect the bridgewire welding, the end closures of the anomalous units were machined off and the propellant was soaked out. No apparent cause for the anomalies could be determined from this inspection. However, a review of the equipment used, in particular the electrode, revealed that the electrode configurations were not to drawing. The electrode tips had been sharpened to a point; the print calls for a flat tip. SOS is in the process of welding bridgewire samples in the discrepant configuration for further investigation. SOS is also investigating how the nonconforming electrodes passed receiving inspection. The SOS NSIs installed on STS-33 were not from lot MDE. There were no anomalies resulting from the lot tests performed on the lot of NSIs used on STS-33. *This risk factor was resolved for STS-33.*
2	Potential procurement of incorrect sheet metal for Space Shuttle Program Element subassemblies. *No anomaly was reported on STS-33 related to incorrect material.*	The Space Shuttle Main Engine (SSME) Project at Marshall Space Flight Center (MSFC) identified the possibility that incorrect material was procured and used in the High-Pressure Fuel Turbopump (HPFTP) support sheet metal. The identified sheet metal was procured from McGregor Supply Company. Because of the potential that other Space Shuttle Program Elements procured sheet metal from McGregor, all Space Shuttle Program Elements were directed to investigate

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
2 (Continued)		procurement and material usage records to determine the impact of this problem on the Program. The following is a summary of the investigation results. <u>SSME Project Investigation</u>: The SSME Project identified the potential for using incorrect sheet metal, Hastelloy X instead of the required Haynes 188, in the HPFTP support structure. The sheet metal is used to protect the turbine support structure from direct hot gas flow. Hastelloy X was found to be used in newly-procured sheet metal subassemblies. Eddy current tests were performed on 30 delivered units to verify material used. Of the 30 tested, 5 were found to be made of Hastelloy X material. These units were found in the HPFTP assembly process at Rocketdyne and had not yet flown. Material for the 5 discrepant units was traced to 1 purchase order #104040 and 1 material heat lot #5L7241R. Ten additional, undelivered units were identified at McGregor to contain Hastelloy X. These were also from purchase order #104040 and heat lot #5L7241R. An investigation of the sheet metal source found that heat lot #5L7241R originated at the Cyclops Mill. Cyclops stated that they performed chemical and mechanical analysis on all heat lots. Cyclops shipped 55 sheets from heat lot #5L7241R, identified as Haynes 188, to Fry Steel, a material distributor, on August 17, 1984. Records at Fry indicated receipt of 55 sheets from Cyclops in August 1984 that were identified as Haynes 188 from heat lot #5L7241R. Fry did not perform material verification on receipt, accepting the lot by certification from Cyclops. Two of the 55 sheets from heat lot #5L7241R were sold to McGregor in July 1987. McGregor also accepted the material by certification. In January/February 1989, Rocketdyne received 5 turbine support subassemblies from McGregor on purchase order #104040. Rocketdyne does not require material certification on supplied parts, but accepts the material from the supplier with certification of raw materials.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)		Eddy current inspection of the 5 sheet metal support subassemblies showed that all delivered subassemblies with discrepant material had some shield details with correct material, Haynes 188. This implied that only 1 of the 2 sheets used by McGregor to make the subassemblies, identified as being from heat lot #5L7241R, was actually Hastelloy X. Further investigation found that 1 sheet identified as being from heat lot #5L7241R, located at another material distributor, was confirmed to be Haynes 188. This supported a theory that there may have only been one sheet of incorrect material identified as part of heat lot #5L7241R. Investigation was continued to locate the remaining 52 sheets of heat lot #5L7241R sold to Fry Steel. The subassemblies from purchase order #104040/heat lot #5L7241R were impounded and will not be used on flight units. A review of procurement and build records determined that subassemblies on SSMEs currently installed/assigned to the Orbiter fleet were all procured on purchase orders other than #104040, with raw materials coming from heat lots other than #5L7241R. During the investigation process, the SSME Project compared the material properties of Haynes 188 and Hastelloy X. This effort was performed to determine the acceptability of sheet metal subassemblies made of Hastelloy X in the event some were found on flight HPFTPs. Hastelloy X was found to be a good high-temperature material, with comparable strength and low-cycle fatigue characteristics to Haynes 188. It has a slightly lower projected life cycle (less than 1000 cycles), when subjected to 1600°F, the maximum turbine inlet temperature. Relative to the SSMEs, the rationale for flight of STS-33 was based on the fact that no flight units were fabricated after August 1984, were not from heat lot #5L7241R, nor were the STS-33 units procured on purchase order #104040. No flight units are known to have been manufactured with discrepant material.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION 2 (Continued)		Orbiter Project Investigation: Review of Rockwell International (RI) procurement records revealed no direct procurements from McGregor Supply Company. Screening of supplied assemblies/subassemblies found that three suppliers, Sundstrand, Marquardt, and Aerojet, use either Haynes 188 or Hastelloy X in manufacturing. All reported that they do not procure materials from McGregor. There are no materials on the Orbiter vehicles that have been procured from McGregor. External Tank (ET) Project Investigation: The ET Project reported that Martin Marietta Manned Space Systems has no record of subcontracts with McGregor. Martin Marietta has procured materials from Fry Steel; however, all procurements were for use in tooling or facilities. An investigation of Martin subcontractors who use steel found that none have subcontracts with McGregor. Six Martin subcontractors have purchased bar stock from Fry, but none have reported material discrepancy problems. One of the 6 Martin subcontractors buys directly from Cyclops. None of these Martin subcontractors use Hastelloy X or Haynes 188 on hardware manufactured for the ET. Solid Rocket Motor (SRM) Project Investigation: The SRM Project reported that there were no procurements from McGregor Supply Company.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
2 (Continued)		Solid Rocket Booster (SRB) Project Investigation: The SRB Project reported that United Space Booster, Inc. (USBI) certified that SRBs have no components or flight hardware containing material purchased from McGregor Supply Company. USBI also certified that Sundstrand, the SRB Auxiliary Power Unit (APU) supplier, has not procured components from McGregor Supply Company for the past 20 years. ***This risk factor was resolved for STS-33.***
3	APU fuel pump failed during fuel pump calibration at Sundstrand. HR No. ORBI-100 INTG-149 ***No APU fuel pump anomalies were reported on STS-33.***	Hydraulic Power Unit (HPU) fuel pump Serial Number (S/N) R88B003 failed catastrophically during pump calibration at the vendor, Sundstrand. After 14 minutes of total test run time, the test operator noticed leakage into the seal cavity drain and shut down the test. Following shutdown, rapid hydrazine decomposition destroyed the pump and severely damaged the Ground Support Equipment (GSE) on which the pump was installed for the test (no HPU involvement in the test). The HPU pumps are very similar to Orbiter APU fuel pumps, and the concern was that detonation of either an APU or HPU pump could lead to a catastrophic Space Shuttle Vehicle (SSV) event during flight operations. This fuel pump was flown on STS-27 and was refurbished for future use. During the test, a sudden decrease in discharge pressure and fuel flow were observed. High torque was experienced, and there was hydrazine leakage through the fuel pump shaft seal. The test was discontinued. Detonation of the pump occurred after shutdown.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
3 (Continued)		Post-failure inspection found the following: • Pump drive gear bearing was cracked. • Pump gear journal was discolored – reddish stains and circumferential markings. • Shaft seal was discolored – violet tint. • Pump cover was deformed outward. • Drive shaft was discolored on both spline ends. • Drive shaft carbon seal was damaged. Failure of the fuel pump was attributed to improper lubrication and/or contamination. The pump was filled with hydrazine and stored for 29 days prior to test. Normal soak time is 10 hours (hr) to 1 week. It is believed that excessively long storage allowed the hydrazine to leak or evaporate through the shaft seal (normal leakage route) and/or GSE valves. A companion pump, stored under similar conditions, was essentially dry when opened; it had only 2 drops of hydrazine left. Contamination was found in the form of aluminum hydroxides and oxides and iron deposits. Iron oxide is a catalyst that will ignite in hydrazine at room temperature. Other failure modes were evaluated – electrical arc, bearing seizure, test stand failure, mechanical spark – but investigation results led to discounting these failure modes. The most probable failure cause was contamination. The abnormally long soak (29 days) allowed hydrazine to escape through the shaft seal and/or GSE valves. (The GSE valves will not leak if installed properly.) In S/N R88B003, hydrazine reacted with introduced air and produced the contaminants. The sister pump was stored

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
3 (Continued)		under similar conditions, and similar contaminants were found. Introduction of contaminants between the bearing and journal caused local breakdown of the hydrodynamic lubrication film. During pump operation, this was manifested in an indication of bearing stress at the end of the 10-minute initial calibration run during 1500 pounds per square inch gage (psig) steady-state conditions. Thirty-minute shutdown for orifice changeout allowed heat to be retained in the localized journal and bearing region. Restart of the calibration run was at low discharge pressure (<1200 psig), and the final 1500 psig steady-state condition generated sufficient friction heat to cause rapid decomposition of hydrazine. Pump calibration requires continuous loads for extended periods of up to 10 seconds. Normal flight usage is 1-3 second pulses. Drive torque increase coupled with reduction in flow just prior to the end of the initial 10-minute test run is a critical indication of bearing distress. When the final test was resumed, temperatures in the low-flow area (shaft seal area) were then raised to decomposition level within 3 minutes due to frictional heating from the bearing. Rapid hydrazine decomposition dislodged the shaft seal resulting in gross hydrazine leakage. There have been a total of 4 bearing-related APU/HPU fuel pump failures; all of these failures occurred at fuel pump level testing. Two of the 4 were SRB/HPU fuel pump failures attributed to contamination. The other 2 were Orbiter/APU fuel pump failures; the first was an unexplained anomaly, and the second was caused by contamination. All of these failures were preceded by abnormal torque increase, with flow and pressure decrease; this is an indication of bearing distress. The fuel pump Acceptance Test Procedure (ATP) serves as a screening test for bearing distress. ATP pump tests exposed component problems in the previous bearing-related fuel pump failures. No bearing-related fuel pump failures have

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
3 (Continued)		been experienced subsequent to a successful fuel pump ATP; this is the screen used to show that the hardware is healthy. Orbiter APU processing at Sundstrand included teardown of fuel pumps to the parts level. O-rings were replaced, all surfaces were visually inspected, and the unit was reassembled in a clean room environment. The fuel pump was calibrated on a test stand, orifice adjustment was made if required, and flush and decontamination were performed prior to final assembly and installation in the APU. An APU-level pre-ATP run was performed, followed by an APU-level ATP. The APU was flushed and decontaminated, and was stored in a container under a Gaseous Nitrogen (GN_2) blanket. Orbiter APU fuel pumps have an extensive data base: approximately 45 pumps and 1705 hr total run time (test stand and APU). They have accumulated many hours of run time without incident. On the Orbiter, a leak occurring through the seals is directed to a drain line that runs to a 500 cubic centimeter (cc) catch bottle for each APU. If the catch bottle becomes overfilled, it would be relieve overboard at approximately 28 pounds per square inch absolute (psia) through a drain port. The flight crew can monitor the catch bottle pressure on a Cathode Ray Tube (CRT) display. In addition, the crew is alerted when there is a pressure change in the catch bottle. Corrective action can be taken by the flight crew to shut down an Orbiter APU as required. STS-33 fuel pumps showed no torque, flow, or pressure abnormalities. However, it was decided to hot fire all 4 SRB HPUs on the pad on November 11, 1989. Normal on-pad hot fire is 20 seconds (sec). The nozzle was gimballed to add actuator loading on the 1 HPU that had been changed out; the other 3 HPUs saw normal system load. Inspection for leakage was performed after firing. Hot fire was preceded by a minimum soak of 9 hours on all 4 systems. On-pad hot fire showed no anomalies; performance was nominal.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
3 (Continued)		Rationale for flight was: • SRB HPU fuel pumps were screened by ATP. No pump that has successfully passed ATP screening has ever failed. • All SRB HPU fuel pumps performed normally during the 20-sec hot fire on November 11, 1989. No leakage or other anomalies were experienced. • Orbiter APU fuel pump catch bottle pressure can be monitored by the crew, and corrective action can be taken to shut down an APU if required. • There have been no failures in a total of 1670 APU/HPU operational runs. ***This risk factor was resolved for STS-33.***
4	Unitrode diodes alert. HR No. ORBI-038 INTG-113 INTG-145A B-50-26 Rev. C-DCN2 ***No anomalies were attributed to Unitrode diodes on STS-33.***	MSFC Investigation: Unitrode diodes were investigated against several alerts and evaluated for Space Shuttle applications. Generic failures of Unitrode diodes with DO-35, soft glass, silver button construction were identified. Recent evaluations by MSFC, Department of Defense (DoD), and the aerospace industry determined that this type of construction cannot be upgraded for use in space. Recent SRB Multiplexer-Demultiplexer (MDM) failures have occurred that were traced to Unitrode diode failures. Diodes 1N3600, with Lot Date Code (LDC) 8217, were the subject of the latest Unitrode diode alert (MSFC Alert No. 5668). The Unitrode diode failure mode is caused by poor metallurgical bonds between the silicon die and dumet slug, resulting in a degraded conductive path.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
4 (Continued)		There has been 1 confirmed Unitrode diode failure from LDC 8217 in the SRB Program. A companion diode failed due to probable overstress related to the confirmed failure. Two other failures were recorded and attributed to probable overstress during troubleshooting. Maximum stress for critical diodes occurs during MDM bench testing and not during countdown or flight. MSFC investigation has determined that there were 12 suspect diodes in the SRB aft MDM S/N 056 of STS-33/BI-034. Two were found in a Criticality 1R circuit (only 1 side of the redundancy affected); the other 10 were in 3 noncritical circuits. During launch countdown, a Unitrode diode failure in the aft MDM would result in violation of Launch Commit Criteria (LCC) and a scrub. During flight, failure would result in shutdown of 1 APU. Prelaunch testing and checkout verified operation and redundancy through the launch countdown. Rationale for flight of STS-33/BI-034 with Unitrode diodes in the SRB aft MDM is based on only 1 confirmed failure out of 775 LDC 8217 diodes in the SRB MDMs. Prelaunch testing and checkout also verified diode integrity through launch. <u>JSC/RI Investigation</u>: There have been 3 similar Unitrode diode failures in Orbiter MDMs – 2 at the vendor and 1 on the Orbiter during Palmdale checkout. JSC determined that 18 Orbiter MDMs have 702 LDC 8217 Unitrode diodes (excluding Multiplexer Interface Adapters (MIAs)). The diodes are used in core and Input/Output Module (IOM) power supplies. Single failure could result in loss of 1 core or loss of some Input/Output (I/O) channels.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
4 (Continued)		Review of Criticality 1 and 1R2 Line Replaceable Units (LRUs) showed usage of diodes 1N3600 in the following units: S-MIA, D-MIA, DBIA, IMU, MTU, ADTA, and Ku-Band. Sample data review showed no Unitrode LDC 8217 diode usage or reported failures in these devices. Rationale for flight was based on the low failure rate (4-out-of-1478 total for Orbiter and SRB), no failure history other than MDMs, and MDM redundancy (port moding provides redundancy in the core power supply, and redundant MDMs are used for the IOM power supply function). *This risk factor was resolved for STS-33.*
5	Integrity of loctite and primer installed on ET and SRB threaded fasteners was suspect. HR No. C-00-04 Rev. B-DCN2 S.11 *No anomalies were reported on STS-33.*	During routine shelf-life extension testing of primer used on ET and SRB threaded fasteners, it was found that the applied primer and loctite did not meet minimum torque breakaway requirements when tested at USBI. Breakaway torque was found in the range of 4 to 10 inch-pound (in-lb); the requirement is 20 to 50 in-lb. The shelf life of the primer was suspect because it had been approved previously for an extended shelf-life period. Further testing at USBI, using primer identified as within shelf life, also failed to meet the minimum requirements. The loctite used in these tests had already expired its identified shelf life. The results of these tests raised the potential that both the loctite and the primer had become defective due to an unknown cause. Loctite and primer used on the ET and SRBs are procured by MSFC and sent to Kennedy Space Center (KSC) for use. Loctite and primer used on Orbiter fasteners are procured by RI and sent to KSC for use. Loctite and primer used on the Orbiter are not in question.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
5 (Continued)		Plans prior to launch called for acquiring new lots of loctite and primer for testing in different combinations of old and new in an attempt to isolate the problem. Results of these tests were presented at the Launch Minus Two Day (L-2) Review. Tests of loctite and primer at all other facilities passed. Because tests of loctite lots only failed at USBI, the test setup and controls at USBI are in question. ***This risk factor was resolved for STS-33.***

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	OV-103 has one MDM that may contain Erie capacitors which are prone to failure. HR No. ORBI-038 *No MDM anomalies were reported on STS-33.*	In 1981, Erie capacitors were found to be failure prone due to a low-resistance short. The Orbiter Program Office (OPO) directed that the capacitors be purged from increment III MDM builds and directed that replacement of the capacitors on the Orbiters would be by attrition or when the MDMs were returned to the vendor. Presentations made at the STS-28 Flight Readiness Review (FRR) Action Item Closeout Meeting indicated that OV-103 had 1 MDM installed with Erie capacitors. The configuration of the OV-103 MDM was as follows: S/N 3 OA2 Operational Instrumentation – Aft MDM #2 (256 Erie capacitors). OA2 is a Criticality 1R/2 unit. Rationale for flight was based on the low probability of failure, and that there have been no Erie capacitor-related flight failures to date. ***This risk factor was acceptable for STS-33.***
2	OV-104 Star Tracker Door stalled during closure. HR No. ORBI-011B *The Y Star Tracker Door thermal blanket detached during the STS-33 mission (IFA No. STS-33-11). It did not, however, interfere with the door operation.*	The Z Star Tracker Door stalled during closure as OV-104 was readied for rollout. Investigation found that 6 of 7 fasteners on the Z-door insulation blanket were located incorrectly. The wrong reference line was used to locate the fasteners causing the insulation blanket to protrude into the door track. The Y-door was also cycled to verify that the blanket was installed correctly. It was found that the blankets on the Y-door and the Z-door interfered with the Star Tracker bright object sensor. Interference with the Star Tracker door blanket has most likely existed for some time.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2 (Continued)		Corrective actions included a redesign of the blankets to improve fastener location and a cutout in the blanket for the bright object sensor. The modified blankets were installed on OV-103. Blanket installations on other door and moving part areas were assessed. Modifications were directed and implemented to tape cotter pins on the Payload Bay Door torque tubes to alleviate possible blanket interference. Design of blanket installations in all other areas was found to be acceptable. Proposals were made to remove the Star Tracker Door thermal blankets. Recent analysis indicated that the thermal environment in the Star Tracker cavity is not as severe as previously believed. Temperature sensors are to be installed in the cavity for a future flight to collect actual temperatures. ***This risk factor was resolved for STS-33.***
3	Potential tire failure due to undetected delaminations after preroll. HR No. ORBI-021 ORBI-185 *No tire failures were reported on STS-33.*	Tires are now inspected for possible delamination when new and after preroll in accordance with a new requirement. Tire preroll will be performed every 15 months. Previously built tires have had no Nondestructive Inspection (NDI) after preroll as do new tires. Preroll has been determined to be benign relative to stressing the tires. Only 60,000-pound (lb) load at 5 knots is imposed on a tire during preroll. Vendor NDI of a test tire found no irregularities and no indication of delamination. These inspections included special multiview x-rays of the tires. There are plans for destructive testing of an STS-34 tire at the vendor after the flight. The tire will be sliced, and pull tests for adhesive capabilities will be performed on the plies.

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3 (Continued)		Rationale for flight included: • NDIs showed no delamination on the test tire. • No history of tire delaminations was found on any flight vehicles. *This risk factor was resolved for STS-33.*
4	Orbital Maneuvering System (OMS) fuel tank weld crack in communication screen. HR NO. ORBI-054 ORBI-166 *No OMS fuel tank anomalies were reported on STS-33.*	Several weld cracks were observed during acceptance testing of 3 communication screen panels on an OMS fuel tank. This tank was returned to White Sands Test Facility (WSTF) for failure analysis due to temperature rise experienced during probe replacement at KSC. The communication screen is part of the OMS propellant acquisition system, which provides bubble-free propellants to the OMS engines. In addition to the communication screen, 2 other propellant acquisition system parts, the collector manifold and the galley assembly, remove bubbles from the propellant. These parts are downstream from the communication screen panels. Bubbles in the propellant result in reduced thrust, hard starts, and transient combustion instability. Hard starts and combustion instability can result in engine explosion leading to possible loss of crew and vehicle. The particular OMS fuel tank in question also had weld cracks in the communication panels in 1983. Weld cracks, found during tank acceptance testing, were due to a lack of fusion in the weld. The cracks were repaired, and the tank was installed in LP04 on OV-103. Only 5 communication panels have been found with weld cracks: 3 on the LP04 fuel tank, 1 on an oxidizer tank on LP04, and 1 that was scrapped.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
4 (Continued)		All tanks on OV-103/STS-33 passed the bubble-point test before delivery. To date, flight data shows no gas ingestion. The cracks found on the communication screens were 1/2" to 3/4" in length. Detailed fracture analysis by the vendor indicated that the cracks will not grow to greater than 3/4". Cracks of this size, or less, will not cause bulkhead structural failure or leakage in flight under the most adverse maneuvers. Rationale for flight was based on the following: • All OV-103 OMS fuel tanks passed the acceptance test program with either repaired cracked welds on the communication screen panels or no indication of cracks. • Bubbles will not pass through weld cracks of 3/4" or less under the most adverse flight conditions. • Two additional means exist downstream of the communication screen panels for removing bubbles from the propellant. ***This risk factor was acceptable for STS-33.***
5	Excessive body flap deflection during ascent. IFA No. STS-28-ORB-24 HR No. ORBI-025 ***No excessive body flap deflection was observed during the review of STS-33 ascent films.***	Excessive body flap deflection was believed to be observed by the film analysis team from the E-207 tracking camera at approximately 46-second Mission Elapsed Time (MET) during STS-28 ascent. The camera was turned on at T-0 versus T+50 seconds on prior flights. Body flap deflection was witnessed on the film at Max Q for about 10 sec. Initial measurements taken from the film were assessed to show a deflection of up to 9 ±4" at a natural frequency of 8 Hertz (Hz). This amplitude measurement was suspect due to dynamics of the vehicle/camera, plume effects, and variable lighting and was later revised to 6.1 ±3.0". Camera photographs from previous flights did not provide the view angle needed to observe flap movement.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)		Deflection of approximately 2" was witnessed during qualification testing prior to STS-1. Acoustical qualification testing resulted in deflections at a natural frequency of 12.4 Hz. Also during acoustical testing, similar deflections to those recorded on STS-28 were seen in the area of rotary actuator attachment; however, investigation found that a bearing in the rotary actuator was walling out. The OPO developed and implemented a plan for testing OV-102 (STS-28) in the Orbiter Processing Facility (OPF) using a shaker to verify the natural frequency of the body flap and inspection of the inner body flap and actuator mechanisms. Modal vibration tests were conducted on OV-102 and OV-103 body flaps. Static deflection tests were also conducted on each of the body flaps to determine free play; the free play was within the allowable range for all 3 vehicles. The body flap on OV-102 was removed, and the fittings, attach points, etc. were inspected and measured; no significant problem was detected. The internal cavities of the flap were borescope inspected. Some evidence of heating (discoloration) and metal filings indicative of wear were seen, but no significant problems were found. One actuator was returned to Sundstrand for evaluation; it tested 3% less efficient than when new, which is an excellent result for an actuator with an equivalent amount of time in service. Since no significant problems were found during all of the testing and inspection of the OV-102 body flap, the flap was reinstalled on OV-102. Three new actuators and the retested actuator were installed. Body flap test and inspection results for OV-104 (STS-34) were satisfactory. No problem was found during the tests on the other 2 vehicles that would affect OV-104. Review of Configuration Verification Accounting System (CVAS) documentation verified that the OV-102 hardware was installed per design requirements; there were insignificant differences from vehicle to vehicle. OV-102 actuator attachments were within design requirements, and the actuators passed the ATP. However,

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER 5 (Continued)		additional analysis was assigned as an action item at the STS-34 FRR to determine the following: • Calculate estimates of potential visual amplification and distortion associated with views through space shuttle engine plume gases. Estimate bounds of measurement accuracy, including the end-to-end photo/video system performance capabilities. • For each of the following peak-to-peak body flap deflections (4", 6", 9", 13"), determine area and type of predicted damage, and if no predicted damage, the margin. <u>Results of the STS-34 FRR Action Item</u>: Estimates of potential visual amplification and distortion associated with camera views through the SSME plume were determined to have little to no effect (approximately ±0.2"). Readability errors were calculated to be ±2.2" based on comparison with other, non-moving areas on the Orbiter. Summary of more recent analysis of STS-28 film, considering the effects of the plume and readability errors, led to the conclusion that there was body flap motion of 6.1 ±3.0" peak-to-peak on STS-28 compared to 9 ±4.0" originally measured. Analysis of predicted structural/component damage resulting from various peak-to-peak deflections found that no damage would result with deflections up to 6" peak-to-peak. Tile damage would begin to occur at around 6.5". Structural damage would occur at higher levels of deflection, beginning with a bearing failure of the actuator rib upper lug at 7.5" peak-to-peak, and a tension failure of the actuator rib upper lug at 8.7" peak-to-peak. Based on the predicted tile and structural damage,

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)		coupled with the latest prediction of the peak-to-peak deflection seen on STS-28, additional modal testing may be performed on OV-102 prior to STS-32. It was reported during the STS-34 Safety, Reliability, Maintainability and Quality Assurance (SRM&QA) Prelaunch Assessment Review (PAR) on October 10, 1989, that no significant body flap tile damage has occurred on any flight which could be attributed to excessive body flap deflections. Modal vibration tests and static tests were conducted on OV-103. OV-103 remained constant at 8.23 Hz with a constant effective stiffness. The OV-102 body flap exhibited a load thumping noise during testing; OV-103 was much quieter. Only the port outboard actuator on OV-103 thumped. Free-play tests performed on OV-103 resulted in exceeding the test criteria. This result was deemed inconclusive because it was later determined that the free-play test setup on OV-103 was not correct. Rerun of the OV-103 free-play tests was not possible due to the unavailability of the vehicle; however, tests and analyses performed on other Orbiters indicated that a body flap deflection problem does not exist on any vehicle. STS-34/OV-104 ascent film review found some evidence of body flap deflection; however, it was not considered as great as that seen on STS-28/OV-102. No damage to tile or body flap mechanisms was reported during postflight inspection of STS-34/OV-104. ***This risk factor was resolved for STS-33.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	Reaction Control System (RCS) left helium valve "A" failed. HR No. ORBI-129A *No RCS helium valve failures were reported on STS-33.*	The RCS left helium oxidizer valve "A" closed when the "B" valve was opened during the STS-28 regulator reconfiguration. The failure was attributed to shock induced by the pressure surge following opening of the "B" valve. When the valve is opened, high pressure upstream of the valve causes helium to surge into the low-pressure area between the valve and the regulator. This surge and shock occur with high-delta pressure across the valve and may cause the valve in the parallel circuit (in this case the "A" valve) to close. These valves are held open magnetically and close by spring tension. The surge and the mechanical shock may cause the valve to close by combining with the spring to overpower the magnetic latch. Only one of the 6 helium valves demonstrated this anomaly on STS-28. The Flight Rules were changed for STS-33 to require the crew to place the "A" valve in the manual "open" position prior to opening the "B" valves. This change prevented the "A" valve from closing due to shock from the pressure surge caused by opening the "B" valves. ***This risk factor was resolved for STS-33.***
7	Fuel cell separator plate plating defects. HR No. ORBI-282A *No fuel cell anomalies relating to separator plate defects were reported on STS-33.*	During teardown of fuel cells S/N 104 and S/N 115 for operational improvements, plating blisters were found on 46 separator plates (Oxygen (O_2)-to-Hydrogen (H_2), and H_2-to-coolant). Fuel cell operating times were approximately 1000 hr; they had 3 flights each (STS-9, STS-61C, and STS-26). All of the plates were from a single lot of 255 manufactured from December 1982 through December 1983. The blister failure was separation of the gold and nickel layers from the magnesium base material. No corrosion was observed through to the magnesium. Potassium hydroxide, used as an electrolyte with water, passivates the bare magnesium, and no corrosion occurs. Corrosion pits may develop if material impurities are present at the blister site. Explosive mixing of H_2 and O_2 through the separator plate could

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)		lead to a catastrophic event; indication of H_2 and O_2 mixing requires immediate fuel cell shutdown and safing. Mixing of H_2 and coolant is more benign, resulting in slow degradation in fuel cell performance. Two plates from the suspect lot were the subject of fuel cell leakage failure investigations. The first plate was removed from a fuel cell due to low performance of 2 adjacent cells. Blisters were identified, and the plate was subjected to extensive test and evaluation. A 100 pounds per square inch differential (psid) helium pressure test in water revealed a minuscule through-plate leak of approximately 1 bubble/minute. The leak was determined not to be at the blister. Further investigation determined that the leak was the result of base metal porosity. This was the only case of base metal porosity discovered in a plate which had passed the normal acceptance test program. The second plate was removed to locate a coolant leak. The coolant leakage was found to be due to a seal debond; again not related to the blister problems. Separator plates from this suspect lot are in the current flight or qualification fuel cells. An accounting of these plates follows: OV-103 2 in Fuel Cell #1, H_2-to-O_2 plate, with 411 hr of operation. 1 in Fuel Cell #2, H_2-to-coolant plate, with 519 hr of operation. 45 in Fuel Cell #3, 2 in H_2-to-O_2 plate, 43 in the H_2-to-coolant plate, with 854 hr of operation.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7 (Continued)		The Orbiter Program Management has directed that Fuel Cells #1 and #3 on OV-103 will be changed out after STS-33. The rationale for flying STS-33 with these suspect fuel cells was as follows: • Leakage in the H_2-to-O_2 separator plate can be detected by the cell performance monitor. If leakage is detected, procedures call for the crew to shut down the indicated fuel cell. Loss of 1 fuel cell results in a minimum-duration flight; loss of a second requires emergency powerdown and landing at the next primary landing site. 94% of the suspect plates on OV-103 are in one fuel cell: #3. • Qualification fuel cells operated for 2000 hr with no problems. Fuel cell history showed successful operation with blistered plates for up to 3500 hr. S/N 104 and S/N 115 fuel cells, where blisters were initially found, have over 1000 hr of operating time. The fuel cells on OV-103/STS-33 had less than 854 hr of total operation. The other 2 have much less. • Turnaround testing (Nitrogen (N_2) diagnostics and coolant leak checks) verified fuel cell integrity. For OV-103/STS-33, these tests were successfully passed. *This risk factor was resolved for STS-33.*
8	17" disconnect main actuator drive link bearing cracks. HR No. ORBI-302A *No 17" disconnect anomalies were reported on STS-33.*	During Operational and Maintenance Requirements and Specifications Document (OMRSD) borescope inspection of OV-104, main actuator drive link bearings in the 17" disconnect were found to be cracked. The bearings are made of Vespel (a plastic material). In the Liquid Hydrogen (LH_2) actuator, 1 of 4 bearings was cracked; 2 of 4 Liquid Oxygen (LO_2) actuator bearings also had cracks. Inspection of OV-102 found 2 of 4 LO_2 actuator drive link bearings cracked. No cracks were found in the LH_2 side of OV-102. During the 1981-1982 Qualification Test

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		Program, 4 drive link bearings were reported cracked. This was probably caused by thermal excursions. Stresses are relieved after crack formation. These cracks were found not to impair the function of the 17" disconnect and were accepted by Space Shuttle Program Management. OV-103 drive link bearings could not be inspected in the current mated configuration. Of the 3 Orbiters, OV-103 actuators are the newest in the fleet with only 2 flights. Actuators on OV-102 have experienced 3 flights, OV-104 actuators have been used for 9 flights. One snap ring was missing when the OV-105 LH_2 disconnect was delivered to RI. Analysis of this anomaly found that interference between the ring and groove is possible. As part of the OV-103 turnaround for STS-33, the 17" disconnect actuators were inspected to determine if the snap rings were in place. The snap rings hold the bearings on the actuator pins. The drive bearings were not, however, inspected on OV-103. No requirement currently exists to do such an inspection. The OPO is currently developing OMRSD requirements to inspect the bearings during turnaround activities. Test and flight experience have demonstrated no significant effect on critical 17" disconnect actuator functions with cracked bearings. Valve closure timing, flapper angle, and tip load functions have all been checked for proper operation. Worst-case results would be binding of the actuator, considered unlikely due to the lubrication used on the bearings. In the event that a cracked bearing fails completely (i.e., breaks-up), the drive link bearings are physically contained. Latches in the disconnect provide backup against inadvertent closure of the flapper during ascent. Additionally, the flapper closing function is backed up by mechanical closure systems.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8 (Continued)		Cracks in the actuator drive link bearings, if they exist on OV-103, were not considered a constraint to flight. Rationale for flight was based on previous experience which demonstrated that cracked link bearings do not impair actuator performance. ***This risk factor was resolved for STS-33.***
9	Kemet capacitor. HR No. ORBI-334 ***No anomalies associated with Kemet capacitor failures were reported on STS-33.***	Input/Output Processor (IOP) S/N 35 failed on OV-104 on August 8, 1989. Failure analysis isolated the failure cause to a Kemet ceramic capacitor located on the IOP timing page (307). This particular capacitor is used as a power supply decoupler; failure results in a short circuit which loads down the power supply. Further investigation revealed that this was the third IOP failure attributed to these capacitors. These failures occurrences were: • AC6416, August 1, 1983, 500 hr, during Contractor Acceptance Test (CAT). • AD3786, December 7, 1987, 3500 hr, at +120°F. • KB0994, August 8, 1989, 10703 hr, at KSC. All the failed Kemet capacitors came from a single lot, LDC 8132 (M11015/89). Dendrite growth (electromigration) resulted in leakage and shorting. This is similar to the previous failure concerns with Erie and USCC capacitors. IOP 38 (slot 2) and IOP 36 (slot 5) on OV-103 have up to 25 each of these capacitors.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9 (Continued)		The worst-case effects are: • Loss of IOP (Crit 1R2), Critical Item List (CIL) 05-5-B02-1-1. • Loss of Primary Avionics Software System (PASS) set plus loss of Backup Flight System (BFS) General Purpose Computer (GPC). • Command path failure to the SSME in the last 30 seconds prior to Main Engine Cutoff (MECO). • Loss of data from 1 Inertial Measurement Unit (IMU) plus faulty data from another. • Command path failure to Aerosurface Amplifier (ASA) or Ascent Thrust Vector Control (ATVC) plus undetected Flight Control System (FCS) failure. • Command path failure to Helium (He) blowdown during Return To Launch Site (RTLS) or Transatlantic Landing (TAL) abort. The plan was to fly the IOP in slot 2 as is. Parts in IOP slot 2 on OV-103 had around 7400 hr. There was a lower probability of failure than for Erie and USCC capacitors. All scenarios involved low probability of additional failures.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9 (Continued)		Corrective action was taken. The Program Requirements Control Board (PRCB) directed that the BFS GPC would not contain Kemet capacitors. IOP S/N 36 in slot 5 was replaced with IOP S/N 20. Current flight rules state that if the BFS (GPC #5) fails on ascent, procedures are to identify GPC #2 as the BFS. Since GPC #2, IOP S/N 38, contained suspect Kemit capacitors, it was not desirable to use GPC #2 as the BFS for STS-33. Safety, Reliability, and Quality Assurance (SR&QA) notified the Mission Operations Director (MOD) of the needed BFS GPC Fail Recovery procedure modification. Based on this notification, MOD implemented a change to the flight rules which stated that if the BFS (S/N 20 in slot 5) fails during ascent, MOD will make a call to change to an alternate GPC besides GPC #2. ***This risk factor was resolved for STS-33.***
10	Loose connector backshells. ***No loose Orbiter connector backshells were found following STS-33 flight.***	Inspection of OV-103 aft fuselage connectors revealed 4 connectors with loose backshells. The problem was not limited to a specific connector type. The concern was that the loose backshells could result in connector disassembly. All accessible OV-103 aft fuselage connectors were inspected for loose backshells. A total of 532 connectors were inspected; 4 were found with loose backshells. These connectors were documented by Problem Reports (PRs) in the KSC Problem Reporting and Corrective Action (PRACA) system. Three of the 4 connectors (without shield terminations) were retorqued to the applicable specification. The fourth connector contained shield terminations which required loctite application on threads prior to torquing; it was inspected by engineering and accepted for STS-33 flight by Material Review Board (MRB) disposition.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
10 (Continued)		Each PR disposition included a step to verify that the spot-tie holding the harness to the connector strain-relief tang was in good condition and correctly secured. This was essential, since the rationale for acceptance of loose backshells for flight is based in part on confidence that this tie will prevent any further connector disassembly. A Requirements Change Notice (RCN) has now been implemented to require inspection to verify electrical connectors are properly mated and backshells are not loose or broken. ***This risk factor was resolved for STS-33.***
11	2" Gaseous Oxygen (GO_2) disconnect bolt problem. HR No. INTG-150A ***No 2" GO_2 disconnect anomalies were reported on STS-33.***	As part of the modification kit for the new Belleville spring assembly, a thicker retainer cover and longer MP 35-N bolts were provided. However, old "short" Inconel 718 bolts were installed at KSC. This resulted in a lower margin of safety due to weaker material and possible incomplete thread engagement in the insert. If changeout is necessary, rollback and demate is required. This is a Crit 1/1 situation. Preload would be lost on the disconnect mating surfaces. This would result in loss of pressure to the LO_2 tank and loss of ullage from that tank resulting in SSME explosion due to pump overspeed from loss of Net Positive Static Pressure (NPSP). The disconnect could also slam shut causing rupture of the lines due to waterhammer effect, and the LO_2 tank would collapse due to loss of positive pressurization. Bolts on OV-102 were changed out. STS-34/OV-104 had flown with this short-bolt condition. Rockwell analysis indicated that there was a positive margin of safety (+0.041 above an Factor of Safety (FOS) of 1.4) and sufficient thread engagement even with the short bolts; the insert was locked in place. The bolts will be changed out on OV-103 after this flight. ***This risk factor was resolved for STS-33.***

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
12	Safety wire on Vernier Thruster Valve found missing during vendor inspection. *No anomalies were reported on STS-33.*	The safety wire on the Vernier Thruster valve electrical conduit fitting attach screw was found missing during the vendor's inspection. This condition was found on a Vernier Thruster removed from OMS Pod RP04 which was removed for evaluation of external corrosion after 3 flights. The corrosion inspection was not related to the missing safety wire. Fourteen additional thrusters were inspected at the vendor; all 14 had the same safety wire missing. All conduit attach screws were inspected and found to be within torque requirements. The attach screw holds the electrical conduit fitting to the valve body and provides no potential for thruster propellant leakage. The worst-case effect of the attach screw falling out is that the screw may become loose debris. There were no recorded occurrences of loose or missing conduit attach screws. If the screw does fall out, the conduit will remain in place because it is captured. The potential for structural damage is remote because the screw is only 1.4 grams in mass. Jamming of thruster gimbal actuators is also extremely remote. *This risk factor was resolved for STS-33.*
13	Display Electronics Unit (DEU) #2 failure. *No DEU failures were reported on STS-33.*	During prelaunch preparation for STS-33, DEU #2 S/N 27 failed with a memory parity error after GPC #2 was powered-on. The memory parity error was indicated by a Built-In-Test Equipment (BITE) flag when communications were established. Cycling recovered DEU #2 to nominal operation. Troubleshooting could not repeat the problem. A decision was made to remove and replace S/N 27 prior to flight. Subsequent tests were performed satisfactorily. S/N 27 was returned to the vendor for troubleshooting. *This risk factor was resolved for STS-33.*

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
14	MDM Flight Forward #1 (FF1) failed BITE test. HR No. ORBI-038 *No MDM anomalies were reported on STS-33.*	During preflight testing, MDM FF1 S/N 30 failed BITE-4 test on both ports. Troubleshooting isolated the problem to card 8, channel 11. Card 8 drives the speed brake/rudder/elevon surface position indicators, a Crit 3 function. Surface positions are also available on the Guidance, Navigation, and Control (GN&C) System Summary #1 display. Channel 11 drives the Left-Hand (LH) outboard elevon surface position indicator. Further troubleshooting isolated the problem to the BITE circuitry. Rationale for flight was: • The problem was isolated to 1 channel only; the rest of the MDM FF1 functions were not affected. • Surface position of the speed brake, rudder, and elevons can be monitored on the GN&C System Summary #1 display. *This risk factor was resolved for STS-33.*

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	Nozzle tube bulge on engine #2011. HR No. ME-B7A ME-B7C ME-B7M ME-B7S *No SSME anomalies were reported on STS-33.*	A bulge in the nozzle tube on engine #2027, STS-34 engine #1, was found during inspection. Protrusion measurements indicated that the bulge was present for at least 2 flights. Measurements made after STS-30 indicated that the bulge was not growing. Leak checks were performed which verified that there were no leaks. The bulges were the likely result of tube stacking during fabrication, with no impact on operation or performance. Engine #2011 has similar nozzle tube bulges. Analysis concluded that the bulges resulted from pressure buildup between the jacket tubes caused by a hydrogen leak. These bulges were found through leak checks. Certain tubes that had previously bulged finally cracked, providing a small leak path. Subsequent test firings found no degradation in engine performance due to the leaks and associated bulges. The rationale for flight was: • Leak checks performed after STS-29, the last flight of OV-103 with engine #2011, verified that there were no leaks. • Special leak checks performed on engine #2011 confirmed the absence of leaks. The pressure applied during this special test was held for an extended period of time with no leak measured. • Feel checks on engines #2031 and #2107, the other 2 engines on OV-103, indicated no nozzle tube bulge anomalies. • The protrusion measurements showed dimensions present since manufacturing, with no change in protrusion noted. *This risk factor was resolved for STS-33.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
2	Main Combustion Chamber (MCC) #4007 bond line leak. HR No. ME-B5A ME-B5C ME-B5M ME-B5S *No SSME anomalies were reported on STS-33.*	Leak checks and borescope inspections after STS-29 found a class III leak in the aft bond line of MCC #4007. Statistical analysis of hot-fire histories indicated that this was a random failure condition; it was classified as an infant mortality failure rather than structural fatigue. This finding was consistent with the structural analysis performed. There has been a demonstrated higher probability of a defect initiating after the first hot fire due to yielding of the Narloy-Z material during the hot fire. Subsequent hot firings are less severe on the bond line than the first one. Inspection of the fleet after STS-29 found no other debonds. The MCCs on the STS-33 engines had accumulated hot fire time prior to ultrasonic inspections. The engines and MCCs on STS-33 successfully passed all leak checks. No fabrication discrepancies or proof test disbonds, indicative of marginal bonds, were recorded for the STS-33 engines. ***This risk factor was resolved for STS-33.***

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
3	Engine #0213 MCC liner cavity burst diaphragm ruptured at higher than design pressure. HR No. ME-B5A ME-B5M ***No SSME anomalies were reported on STS-33.***	During test 904-052 at Stennis Space Center (SSC), engine #0213 prematurely shut down at T+5.6 seconds. The shutdown was due to the MCC liner cavity pressure exceeding 165 psig, the maximum redline pressure established for this test. Nominal redline pressure for flight is 135 psig. The redline was set at 165 psig for this test to ensure that no subsequent structural chamber damage would occur. The MCC liner cavity pressure is normally about 20 to 25 psig during acceptance tests of engines that are not leakers. Engine #0213 MCC was a known leaker, causing the burst diaphragm to rupture on 4 previous tests between 120 and 135 psig. The burst diaphragm ruptured at approximately 195 psig on this test, which was a cause for concern. Investigation was undertaken to determine the reason that the burst diaphragm did not rupture until 195 psig. Preliminary indications were that this diaphragm gave way gradually instead of bursting, causing pressure to continue to build. All diaphragm geometric and material characteristics were found to be nominal, except for the thickness of Room Temperature Vulcanizate (RTV) compound applied over the diaphragm. During a subsequent hot-fire test with a different burst diaphragm, the diaphragm ruptured at 165 psid. Hot-fire test data showed uniform pressure within the liner cavity and uniform ambient temperature downstream of the diaphragm. Laboratory tests confirmed that rupture pressure is influenced by RTV thickness applied over the diaphragm and cryogenic gas pressure. Rupture pressure experience from 7 hot fire samples with an MCC liner leak have demonstrated the following FOS levels:

Liner Cavity Pressure		Factor of Safety		
Basis	Pressure psia	Mainstage 104%	65%	Cutoff
MAX	195	2.1	1.8	1.4
90/95	213	1.9	1.6	1.2
99/95	255	1.6	1.4	1.06

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
3 (Continued)		Rationale for flight was: • Engines #2011, #2031, and #2107 had no indications of MCC leakage. Green runs of these engines were successfully performed prior to shipment, with no premature shutdown or MCC leakage noted. Engine #0213 was a known leaker prior to the recent test. • The cavity has a pressure containment capacity of 300 pounds per square inch (psi), considerably above even the delay rupture of the anomalous burst diaphragm. Worst-case liner cavity pressure will not buckle the MCC liner: – FOS is 1.8 at mainstage, 1.4 at cutoff with maximum measured pressure. – FOS is 1.4 at mainstage and greater than 1.0 at cutoff with 99/95 pressure based on 7 hot-fire tests of known leakers. ***This risk factor was resolved for STS-33.***
4	Severity 2 SSME Controller software problem fix implemented for STS-33 (generic). HR No. INTG-165 *No SSME anomalies were reported on STS-33.*	A severity 2 software problem was identified in the SSME Controller software. An Unsatisfactory Condition Report (UCR) was written to document a coding error in the SSME Controller software in the sample-and-hold circuit monitoring subroutine. The function of this subroutine is to determine if sample-and-hold feedback signals are within tolerance (within 2% of full-scale value of the sample-and-hold digital command). If not, a retest is performed. This retest capability allows for differentiation between channel noise and actual hardware failures. The nature of the coding error was such that the retest was not performed. Instead, the first failure was reread, resulting in disqualification of all Channel A actuators. The potential impact of this error is that good Channel A actuators would be

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
4 (Continued)		disqualified if the first failure indication is a soft failure due to noise. The software response to this failure would be a launch scrub or abort before liftoff, and loss of actuator redundancy after liftoff. This software fix was incorporated into the SSME Controller software delivered for STS-33 and subsequent flights. ***This risk factor was resolved for STS-33.***
5	Engine #2027 MCC coolant outlet temperature sensor failed high on STS-34. HR No. ME-B5A ME-B5C ME-B5M ***No SSME anomalies were reported on STS-33.***	MCC coolant outlet temperature failed high at 85 sec into flight on engine #2027/Main Engine (ME) #1 on STS-34. The sensor (-22 configuration) was found to have an element wire fracture due to defective fabrication. The plasma spray ceramic shattered during element insertion. A dislodged ceramic chip (with encrusted wire segment) broke free under vibration. The condition was first discovered on -71 and early -81 configured hot gas sensors. The manufacturing process specification was revised to control element insertion for new builds after October 1985. This sensor monitors a maintenance parameter; the failure mode results in loss of data only (Crit 3). The sensor was removed at Dryden. The sensor tip was undamaged. Continuity test indicated an open circuit. The -22 configuration sensor was returned to Rosemont for failure analysis. The recommendation was to continue to use the -22 configuration sensor. The probability of failure is low. This was the first program failure due to a -22 sensor defect. The Mean-Time-Between Failure (MTBF) is 175,547 sec (1 in 337 flights). ***Not a safety concern for STS-33.***

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
6	Uralite in G-15 joint on engine #2031. HR No. ME-D3C ME-D3M *No SSME anomalies were reported on STS-33.*	Evaluation of a Uralite sample used in the G-15 joint revealed that the material had not been properly applied and had not set up properly. Uralite is used to fill minor surface imperfections in MCC and nozzle interfaces to the G-15 seal. The joint is mated with Uralite in the liquid phase. Uralite flows onto bellows surfaces; the quantity is insufficient to fill the cavity. The nozzle on engine #2031 was lowered and Uralite was reapplied. The G-15 seal and the Flow Recirculation Inhibitor (FRI) were replaced, and acceptability was verified by testing. Analysis and evaluation of the Uralite indicated compliance with the bellows. Uralite does not preclude proper seal installation and will not restrict bellows seal movement (in the cured state). Uralite will compress/extrude under bellows loading. ***This risk factor was resolved for STS-33.***
7	STS-34 ME #2030 Controller, Unit Number (U/N) F19, checkout anomaly. HR No. INTG-165 *No SSME anomalies were reported on STS-33.*	ME #2 Controller on OV-104, U/N F19, failed 2 of 64 Low-Pressure Fuel Pump (LPFP) discharge pressure 80% R-Cal checks during T-27 hr sensor checkout for STS-34. During Direct Memory Access (DMA) data load, the Channel (CH)-B DMA failed to load data at least twice. Both Digital Computer Units (DCUs) verified memory parity error circuitry each major cycle. If DCU-A attempts a DMA write request to DCU-B memory during the parity checker test, the DMA write is blocked, and the memory location retains the previous data. The parity checker blocks DMA write 6 out of 20,000 microseconds (μsec) (major cycle). The DMA write request (1 μsec each) occurs 58 times per major cycle. The problem is limited to data input into the standby DCU only; it can occur in all phases of operation. The concern is that DCU-B data is not updated when DCU-B takeover occurs. The data includes sensor calibration data, Oxidizer Preburner Oxidizer Valve (OPOV) command limit setting at +5.5 seconds, and HPFTP discharge

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
7 (Continued)		temperature drift monitor mainstage lower limit. Sensor calibration is satisfactory if there are no Failure Identifications (FIDs) in the DCU-B buffer; OPOV command limit too low would result in engine shutdown upon DCU-B takeover; HPFTP drift monitor too low could delay failed sensor disqualification after DCU-B takeover. Failure investigation was performed at Honeywell. The problem was assessed to be produced by either intermittent DMA address failure or by asynchronous DCU software timing interference. No controller hardware problem has occurred to date. Asynchronous DCU software timing was the most probable cause. DCU-B parity check interference was demonstrated; DMA write was blocked 6 out of 20,000 μsec. F-19 failure occurred after 21,000 checks. The potential software fix affects standby DCU only – in sensor checkout, deactivate the parity checker; in all other operating modes, deactivate the parity checker in alternate major cycles. This assures data read into the standby DCU is never more than 20 μsec old. The near-term recommendation was to make no changes, and to continue to verify the sensor checkout in DCU-B and rerun the check until satisfactory if necessary. The long-term solution was to modify the software and verify no adverse results from the fix (target is for STS-35). Rationale for flight was: • Problem was isolated to the Controller. – CH-B DMA addressing hardware. – Asynchronous DCU software timing related.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SSME		
7 (Continued)		• T-27 hr checks validated the Controller. – All hardware functions were verified. – Sensor calibration was correct. • Low probability of operation with incorrect DCU-B OPOV limit or HPFTP drift monitor. – Controller has demonstrated MTBF = 6055 flights per failure. – Channel switchover would be required. This has never happened in the history of the flight program. – F-19 DMA failure was the only one of its type in 41,438 hours. ***This risk factor was acceptable for STS-33.***
8	Crack found on fuel preburner diffuser lip (multiple SSMEs). ***No SSME anomalies were reported on STS-33.***	A crack was known to exist on the lip of the fuel preburner diffuser lip of engine #2011. There have been 7 instances of similar cracking found during the life of the program. Investigation found that 6 of the 7 previous instances of cracks were attributed to diffusers or housings that had dimensions that exceeded drawing specifications and had excessive interference fits. Interstage seal wear was also typical on units with cracked diffusers. The rationale for the flight was: • Diffuser cracks are acceptable for continued flight usage. This is a Crit 3 failure mode.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
8 (Continued)		• Engine #2011 crack is typical of prior experience. The crack faces will remain open after failure and there was no history of ignition, melting, or fretting due to these cracks. • Prior flight experience with cracked diffusers resulted in benign conditions and no detrimental results. There were 7 known cracked units. A total of 49,344 sec of test time during 137 tests were logged with cracked diffusers. The fleet leader of these 7 units had undergone 64 tests, for a total of 28,699 sec of operation. • Engine #2011 fuel preburner diffuser passed final acceptance test with this cracked condition. ***This risk factor was resolved for STS-33.***
9	Engine #2107 High-Pressure Oxidizer Turbopump (HPOTP) start redline assessment. HR No. ME-C1S ***No SSME anomalies were reported on STS-33.***	STS-33 engine #2107 predicted HPOTP turbine Channel B discharge temperature was 1525° Rankine (R) from engine start plus 2.3 to 5.8 sec versus the start redline of 1560°R maximum. This left only a 35°R margin to the redline. HPOTP turbine discharge temperature Mainstage redline was established at 1760°R for STS-1 and SUBS to protect the oxidizer heat exchanger. The start transient redline was set at 1560°R for STS-6 and SUBS to assure the Mainstage limit of 1760°R was not exceeded at 109%. For the engine #2107 HPOTP, with powerhead U/N 2014, discharge pressure was $+5.3\sigma$ high compared to the Phase II data base. The Phase II data base, however, was based only on 3 engines. Discharge pressure of engine #2107 HPOTP was only $+2.8\sigma$ high when compared to the total SSME data base. The high discharge pressure was indicative of high flow resistance in the main injector oxidizer dome, which results in increased HPOTP turbine discharge temperature. Borescope inspection of the main injector oxidizer dome prior to and after the HPOTP green

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
9 (Continued)		run found no discrepancies. The cause of the high resistance associated with powerhead U/N 2014 was not readily apparent; however, powerhead U/N 2014 had been consistently demonstrated to be high since manufacture when used on any HPOTP. A recommendation was made and approved by Space Shuttle Program Management, to increase the HPOTP Start redline value from 1560°R to 1610°R for STS-33 engine #2107 to compensate for the higher predicted HPOTP turbine discharge temperature. *This risk factor was resolved for STS-33.*
10	A dent was found in feedline #2 on engine #2031. *No SSME anomalies were reported on STS-33.*	Inspection of engine #2031 found a dent in feedline #2. This is the 1 1/2" feedline that runs outside the full length of the nozzle. X-ray inspection of the feedline found no wall thinning in the area of dent. No adverse effects were expected as a result of this dent. This condition was approved through the MRB and approved prior to STS-33 launch. *This risk factor was resolved for STS-33.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1	Cracks found in an APU containment housing. HR No. A-20-16 Rev. C A-20-24 Rev. B *No APU containment housing anomalies were reported on STS-33.*	During disassembly of APU S/N 163 at Sundstrand, a crack was noted in the notch machined into the APU housing for a seawater vent. The crack was in the same location as the crack found on APU S/N 165 prior to STS-27. Both S/N 163 and S/N 165 were manufactured by Gentz. While APU S/N 163 has never flown, it had 8 hot starts. A crack in the APU containment housing weakens the structure and could allow hot-gas impingement on other components or fail to contain the shrapnel from an internal failure of the APU. One or a combination of both of these conditions could cause catastrophic failure of the SRB and loss of the Orbiter and crew. Investigation of APU S/N 163 found that the crack was 1.25" long, running radially through the containment wall; nearly the identical signature of the crack found in APU S/N 165. Helium leak decay checks were performed on S/N 163. A decay of 6.5 psi/10 minutes was recorded, greater than the specification limit of 1.5 psi/10 minutes. Examination of the fuel pump housing showed no visible signs of hot-gas impingement. There is a difference in the way APUs were manufactured by the two vendors, Gentz and D'Velco. This difference is in interpretation of the requirement for a seawater vent. Gentz notched the seawater vent into the cast material, D'Velco did not. Analysis has demonstrated that the derived stresses equal or exceed material capability in a notched housing, thus leading to a propensity to crack. Metallurgical results support the stress analysis failure mode that induces an overload condition of a notched housing at the seawater vent. The Right-Hand (RH) STS-33 SRB was found to have Gentz APUs from the same lot as those found with cracked housings (S/Ns 163 and 165). It was directed that the Gentz APU be changed out before the flight of STS-33. S/N 148 was installed and successfully passed an on-pad hot-fire test. *This risk factor was resolved for STS-33.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
2	Water entry into forward skirt on STS-34 during recovery. *No water was found in the forward skirt of either STS-33 SRB.*	Approximately 500 gallons of water entered the SRB forward skirt during STS-34 recovery. The water entered through a 1/4" diameter threaded hole used to attach safety wire for Development Flight Instrumentation (DFI) cable connectors. This instrumentation was not installed on STS-34, and the bolt used to plug the hole was inadvertently not installed. The STS-33 SRBs were inspected, and the installation of both plug bolts was verified to be acceptable. *This risk factor was resolved for STS-33.*
3	Integrated Electronic Assembly (IEA) test failures. HR No. B-50-26 Rev. C-DCN2 *No IEA anomalies were reported on STS-33.*	While performing normal receiving inspection electrical test at USBI, aft IEA S/N 55 failed the power bus isolation test due to a hard short. Power bus "A" return to chassis read shorted; the resistance to chassis should be 160,000 ohms minimum. The unit had tested "good" in the vendor's ATP. The IEA was sent back to Bendix where the cover was removed. It was found that one bundle of 49 wires was wedged, or pinched, between a standoff bracket and another wire bundle. The wire bundle in question was tied to the standoff bracket without a mylar guard around the bundle, which the original design required. This caused the power cable to rub against the standoff, pinching certain wires, and a short to chassis ground resulted. Three wires in the bundle of 49 carry Crit 1 functions; the other 46 have no flight-critical effect. Eleven other aft IEAs were inspected, and 5 additional instances of rubbing were found. It was determined that this configuration was allowable in the applicable drawing and was not just the result of technician error. Both STS-33 aft IEAs were removed on the pad at KSC. The IEAs were taken to the USBI Acceptance Refurbishment Facility (ARF), where Bendix personnel removed the covers and performed a visual inspection supported by government Quality Assurance (QA) personnel. Inspection of the LH IEA found no pinching of wires against the standoff and adequate visible clearance between the bundle and

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
3 (Continued)		the standoff. Inspection of RH IEA S/N 35 also found no wire pinching; however, there was no visible clearance between the bundle and the standoff. The decision was made to reinstall the LH IEA and to replace the RH IEA with S/N 11 from STS-32. Previous inspection of STS-32 IEAs found that S/N 11 had no pinching and clearance was in excess of 1/2". Reinstallation/installation of the IEAs was completed on STS-33 and followed by a successful hot fire of the APUs on November 18, 1989 to assure system integrity. ***This risk factor was resolved for STS-33.***
4	External Tank Attachment (ETA) ring cover fastener on BI-036/STS-36 RH assembly broke in half during installation. HR No. S.11 ***No ETA fastener failures were reported on STS-33.***	During installation on BI-036 at the Rotation, Processing, and Surge Facility (RPSF), an ETA ring cover fastener broke at the second thread from the shank during 60% torquing. This was the first of 468 1/4" NAS 1954C4 fasteners to be torqued to the 60% design value on the ETA ring. The fasteners are torqued in a 3-step process: (1) to running torque, (2) to 60% design torque, and (3) to the design torque value. Installation operations were halted. The broken fastener and 50 other fasteners from the ETA ring were sent to MSFC for evaluation. Calibration of the torque wrench in use when the fastener failed was verified to be correct both before and after the incident. The technician and the quality control inspectors witnessing the installation all stated that the torquing operation was properly performed. The failed fastener was 1 of 6000 purchased from Emanon Aircraft Corporation in May 1985. These fasteners were produced by Screw Corporation, a division of Voi-Shan. All 6000 fasteners were delivered in 3 lots over a 4-month period in 1986. Lot identification was not imprinted on the fasteners, and lots were mixed in the stock bin. Therefore, traceability to the specific lot from which the failed bolt was produced was impossible. There were 679 of these bolts remaining in stock. Material certifications and test data from Voi-Shan were reviewed, with no anomalies found.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
4 (Continued)		The evaluation program at MSFC included the following: • Dimensional and visual checks of the failed bolt and the 50 sample bolts were performed with no anomalies noted. • Scanning Electron Microscope examinations of the failed bolt fracture surface indicated that the break was the result of torsion tension failure. • Fifteen micro-hardness tests were performed on 2 of the sample bolts, resulting in the determination that the bolts were within specification. • Atomic absorption analysis verified correct chemical composition. • Tests performed on 5 sample bolts verified minimum tensile strength of 195 ksi. • Micro-hardness tests performed on the failed bolt verified that it was within specification. • Torque tests performed on 5 bolts found a minimum torque limit of 360 in-lb versus 220-260 in-lb predicted. Fail safe analysis indicated that a loss of 30% of the bolts randomly spaced (i.e., not all in a row) could be experienced before the FOS is reduced to 1.0 in the ETA ring covers. This was considered very unlikely. Failure of 1 of the 468 was considered insignificant relative to the FOS.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
4 (Continued)		Rationale for flight was based on the successful test results and the fact that the ETA could lose up to 30% randomly spaced bolts before the FOS is reduced to 1.0. The failed bolt on BI-036 appeared to be an outlier. Installation torque is the maximum load condition experienced by these bolts. Additionally, all bolt heads are contained in Thermal Protection System (TPS) material and cannot become a debris source. ***This risk factor was resolved for STS-33.***
5	Hydrazine leaks in RH aft skirt. HR No. A-20-24 Rev. B-DCN3 ***No hydrazine leaks were reported on either STS-33 SRB.***	Three minor hydrazine leaks were found in the RH aft skirt during inspection of SRB HPU fuel pumps. The leaks were determined to originate at the following locations: Leak #1 RH tilt fitting, upstream of the Fuel Isolation Valve (FIV) at the joint between the flex hose and the valve connection. Leak #2 RH tilt fitting, downstream of the FIV at the joint between the 90° elbow and the valve connection. Leak #3 RH rock fitting, downstream of the FIV at the joint between the 90° elbow and the tee connection. On November 15, 1989, technicians retorqued the fittings to 540 in-lb in an attempt to stop the leakage. Two of the 3 leaks, leaks #1 and #2, stopped after this operation. Leak #3 still showed signs of a very small amount of leakage after retorquing the fitting. Potential leakage at leak #3 was found using an ammonia dräger, which measured the presence of 0.25 parts per million (ppm) of hydrazine near the fitting. Similar leaks have been experienced on the SRB and the Orbiter APUs. An Orbiter leak, with a measured concentration of 1 ppm of hydrazine, was waivered to fly on a previous mission.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
5 (Continued)		Technicians at KSC inspected the RH rock fitting after the hot-fire test of the HPUs for signs of increased hydrazine leakage. Dräger measurements performed after the HPU hot fire found hydrazine concentrations at leak #3 to be 0.8 ppm. No drops were seen; however, there was enough visible hydrazine at the leak site to nearly form a drop. This amount is below the specified limit. A waiver for this condition was prepared and approved prior to launch. Rationale for flight was: • No additional hazards exist for this condition. A hazard would exist in the case of dripping hydrazine. • Hot fire on November 18, 1989, did not increase hydrazine leakage. • Hydrazine fittings were lockwired and would not come completely loose. ***This risk factor was resolved for STS-33.***

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
1	RH aft center segment on STS-28 showed ply separation. HR No. BC-10 Rev. B ***No SRM anomalies were reported on STS-33.***	Postflight inspection of the STS-28 RH SRM aft center segment revealed a ply separation in the internal insulation. This was the first time that this condition had been witnessed. Ply separation has been determined to be a Crit 1 failure (worst case). No heat effects were detected at exposed surfaces in the area of the separation. Because there were no heat effects detected, the separation most probably occurred after motor operation, possibly at water impact. The ply was found to be open from the forward direction in a straight line centered at approximately 290°, and ran 42" from the tang end of the segment. The open ply was 11.5" in circumferential length, 1.1" in longitudinal length at the center of the separation, and 0.5" in longitudinal length at the end of the separation. Ply separation was 0.05" in radial depth. There have been no other reports of ply unbonds similar to that seen on the STS-28 segment. With the unbonded ply removed, the FOS was calculated to be 1.61. Analysis at Thiokol showed that, due to contamination, the insulation plies were not knitted properly for 3 to 5" in length intermittently around the circumference. The contamination was found to be adhesive residue from yellow vinyl tape used in a contingency process at Thiokol. Review of the build paper found that this contingency operation, called the vacuum bag process, was performed. Upon removal of the tape, the process requires that residue be removed by double wiping the affected area with methyl chloroform. Analysis of the residue on the aft center segment found that residue removal was adequately performed. The vacuum bag process was used on all aft segment insulation installation processes. The difference between normal operation and contingency operation is that, at the end of the residue removal step on the aft segment, the area is 100% visually inspected using a black light prior to installation of the lining. The vacuum bag process was also used on the forward segment, 25 to 48" forward of the field joint.

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1 (Continued)		Review of build papers for the STS-33/BI-034 segments found no record of need for or use of the vacuum bag process. ***This risk factor was resolved for STS-33.***
2	Insulation voids on forward dome. HR No. BC-10 Rev. B ***No SRM anomalies were reported on STS-33.***	Insulation at station 215, approximately 50" forward of the factory joint on the STS-31 LH forward segment, was found to have a below-specification safety factor: 1.43 instead of 1.50. The safety factor is a margin above the case temperature of 200°F. In the insulation multi-ply layup, variations in ply thickness can result in an out-of-tolerance condition after cure. The dome was x-rayed to determine if any anomalies existed; 6 voids were found. The segment was washed out, and the dome area was dissected to evaluate the voids. Additional voids at the insulation-to-case interface, not detected by x-ray, were found during visual inspection. The voids were all in the thickest area of the insulation. All voids were considered acceptable based on the thickness of the insulation. The insulation thickness in this area was greater than that required to meet thermal/erosion criteria. Voids in the forward dome are most probably due to forces incurred during autoclave cure and flow of the rubber. Voids are repeatable and will occur only in the thick boot area. It was found that the manufacturing process was changed prior to the preparation of STS-8 SRMs. It is very likely that voids have been present in this area of the forward dome since that process change. The manufacturing process, however, assures that insulation thickness exceeds the minimum drawing requirement by 0.3 to 0.4".

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
2 (Continued)		Based on past experience, there were insulation voids in the forward dome of STS-33. However, it was determined that occurrence of these voids does not reduce the safety factor below the required 1.43. *This risk factor was acceptable for STS-33.*
3	Forward dome thin insulation FOS. HR No. BC-10 Rev. B *No SRM anomalies were reported on STS-33.*	During investigation of insulation voids in the forward dome of the STS-31 LH SRM, existence of thin-insulation areas was identified. Thin-insulation areas are induced by the forward dome insulation process. Layers of calendered asbestos-filled Nitrite Butadiene Rubber (NBR) are layed-up against the forward dome surface. Thickness is controlled by the number and shape of the NBR pieces used; minimum layup thickness is 0.530". The contour (radius) regions are formed by extruded strips of NBR. Patterning cloth is installed on the insulation surface to form a textured surface that enhances the liner bonding. Bleeder cloth is installed to ensure a vacuum is drawn over the entire insulation surface. The entire internal segment is vacuum bagged with a one-piece bag, and a vacuum is then applied. The insulation is autoclave cured at 100 psi and 290°F (7 hr of cure time and 4 hr cooldown). Tooling and fabrication were modified beginning with the STS-8 SRM flight set, and have not changed. The 57 motors processed using this method have either flown or been used in static tests. The thinned insulation in the forward dome was determined to be derived from tooling problems. Two causes were identified: (1) excess rubber in the igniter boss region is forced outboard by the floating mold ring; (2) the pressure transfer ring bridges and creates higher-pressure pinch points towards the edges of the ring and a resulting lower-pressure zone toward the center of the ring. In addition to creating folds, voids, and bulges, thin-insulation areas develop outboard of the transition region. The problems induced by the insulation process are generic. To date,

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
3 (Continued)		6 SRM forward domes were examined; all have similar voids, folds, bulges, and thin-insulation areas. The forward dome insulation is designed to meet 2 conditions, whichever is greater: • Median + 3σ erosion times a 1.5 safety margin factor. • Median + 3σ erosion plus thermal protection thickness to maintain the case/insulation bond line at less than 200°F. The forward dome insulation is exposed to motor chamber gases for 100 sec. Assuming specification insulation thickness at motor start and nominal erosion, 0.335" of insulation would remain after motor burn (by design). Insulation thickness of 0.335" provides 190 sec of additional exposure time beyond the 100-sec exposure to motor chamber gases. Assuming increased erosion of median + 3σ, 0.186" of insulation would remain after motor burn (by design). At 0.186" of insulation, the dome could withstand an additional 58 sec of exposure time. The STS-28 RH motor had 0.323" of insulation remaining in the thinned area of the dome after motor burn. The motor could have withstood an additional exposure time of 81 seconds, an insulation erosion safety factor of 1.81, and a case structural safety factor of 4.20. Local intermittent thin spots are formed around the circumference in a band approximately 5 to 6" from the igniter boss opening. These are caused by the higher pressure toward the outer edge of the pressure transfer ring. On the STS-31 RH forward segment, the minimum local condition found was 0.396", versus 0.503" required by the design drawing. With 0.503" of insulation, the nominal insulation

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
3 (Continued)		erosion safety factor is 1.59, and the case structural safety factor is 4.20. Based on the worst-case thinning found on the 6 segments examined, 0.396", the median + 3σ insulation erosion safety factor was calculated to be 1.29, with a case structural safety factor of 4.15. This lower insulation erosion safety factor results in a case/insulation interface temperature of 157°F, 37°F above the 120°F ambient. Analysis was also undertaken to determine the effect of a thin-insulation condition worse than the worst case seen to date (0.396"). The results of this analysis found that an insulation thickness of 0.350" would result in a case/insulation interface temperature of 200°F, the design limit, by the end of motor burn. The resulting insulation erosion safety factor is 1.20, an additional 19 seconds of exposure time, and a case structural safety factor of 4.10. Thiokol Corporation claims that the case structural integrity is maintained at temperatures up to 1050°F. Their analysis indicated a case/insulation interface temperature of 600°F at the end of motor burn even if the initial insulation thickness is 0.265". This results in an insulation erosion safety factor of 1.00, no additional exposure time, and a case structural safety factor of 4.10. Information pertaining to the risk associated with thinned insulation, described above, was presented to the Space Shuttle Program Management and was accepted. Safety understood the meaning of the various "safety factors" and concurred that appropriate safety margin existed. ***This risk factor was resolved for STS-33.***

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
4	Fretted aft field joint segments. HR No. BC-01 Rev. B BC-08 *No SRM anomalies were reported on STS-33.*	During postflight inspection of STS-26 Redesigned Solid Rocket Motor (RSRM) cases, a segment in the aft field joint was found to be fretted . Frets in the sealing surface area were blended in accordance with Thiokol Specification STW7-2744, and frets outside the seal area did not require it and were not blended. This procedure requires that the pit be blended along an axial length 30 times the depth plus the width (30D+W). Magnetic particle inspection was conducted after proof test to detect flaws of 0.1" in length or greater. No such flaws were found or reported. Fracture critical flaw size is 0.3" long by 0.15" deep, which is significantly larger than the fret size in Flight Set #7 and is also significantly larger than the detection capability of magnetic particle inspection. The strength of the case in the area where the fretting occurred was not significantly reduced; therefore, there was no strength concern from fretting on Flight Set #7. The effect of the blending on the capture feature O-ring squeeze and O-ring footprint (assuming worst-case conditions) is as follows: the squeeze is reduced from 20.6% to 19.1%, and the footprint is reduced from 0.147" to 0.137". Therefore, blending will have no significant effect on the sealing capability of the capture feature O-ring. There was only 1 segment on STS-33/RSRM Flight Set #7 that was fretted. This segment is not joined with another fretted segment. The fretted attach segment functioned as a normal attachment. There was no effect on strength and sealing. ***This risk factor was resolved for STS-33.***

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
5	Concern with proposal to leave the igniter heaters on after Pyro Initiator Controller (PIC) resistance tests. HR No. FI-01 Rev. B *No SRM anomalies were reported on STS-33.*	A proposal was made to leave the igniter heaters on up to the T-9 minute point in the countdown. Currently, the igniter heaters are turned off at the T-4 hr mark. Analysis indicated that, under nominal Florida winter conditions, the igniter temperature could go below the LCC minimum temperature if the heaters are turned off at the T-4 hr point. Concerns over continuing heater cable failures were raised considering the location of heaters and the heater cables relative to PIC cabling. It was postulated that a failure of the heater or a cable short could cause the PICs to fail through either electromagnetic effects or thermal effects. Since testing of the PICs is performed with a minimal amount of personnel on the pad due to inherent hazards, testing of the PICs after a heater or heater cable failure could not be accomplished during the last few hours prior to liftoff. Investigation into routing of the PIC cabling relative to the heaters found that there was a point where the PIC cables cross directly over the igniter heaters at a 90° angle. Analysis of the electromagnetic field at this point found that induced voltage is in the millivolt range and would not result in detrimental effects. Cork insulation is placed between the heater and the PIC cables at the crossover point to reduce the threat of thermal effects. Routing of the igniter heater cables relative to the PIC cables was also investigated. It was noted that, in the aft systems tunnel, the booster separation motor/NSI cables and the field joint heater cables were adjacent and parallel. Electromagnetic induction between cabling is maximized when cables are parallel. For this reason, and the proximity of the cables in the tunnel, a waiver was generated and approved. Relative to the PIC and igniter heater cables, an installation requirement exists that these cables should not be installed less than 2" apart from one another. Review of STS-33 installation closeout photographs found that the cables were installed more than 2" apart. Electromagnetic and thermal effects were therefore not considered a concern. *This risk factor was resolved for STS-33.*

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
6	Adhesive failure of polysulfide on STS-34 RH SRM nozzle. HR No. BC-04 Rev. B *No SRM anomalies were reported on STS-33.*	Polysulfide in the nozzle-to-case joint exhibited 100% adhesive failure, 360° around the interface. This failure could degrade the sealing capability of the polysulfide thermal protection material. Occurrence of this failure on STS-34 was the first of its kind experienced. This anomaly was found during disassembly of the SRMs. All of the polysulfide remained on the nozzle side of the nozzle-to-case interface. The polysulfide surface was found to be very smooth, indicating that no adhesive properties were present. There was no indication of hot-gas leaks into the polysulfide material. Removal of the nozzle was considerably easier than exhibited on the LH SRM, due to the lack of adhesion. The polysulfide material is not required to provide an insulation function. ***This risk factor was resolved for STS-33.***
7	LH Safe and Arm (S&A) device did not rotate in sufficient time to meet requirement. HR No. BI-03 Rev. B *No SRM anomalies were reported on STS-33.*	During prelaunch testing of SRM S&As, the LH S&A device did not rotate from "safe-to-arm" quickly enough to meet the specified requirement. This S&A took 2.6 sec to rotate versus the 2.0-sec requirement. The LH S&A device was cycled 12 times after this initial rotation, with an average rotation time of 0.8 sec. A review of inspection and test records found that the S&A devices on STS-33 had not been rotated for 2 months. At the last rotation, the LH S&A took 1.4 sec to rotate. S&As tighten up if not periodically cycled a number of times. Use of Krytox lubrication will be implemented on S&As which support STS-31. Rationale for flight was based on experience. Exercising the S&A devices, as was done after the first slow rotation on the LH S&A, reduces friction and rotation time for several weeks. *This risk factor was resolved for STS-33.*

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
ET		
1	LH_2 feedline vacuum-jacketed bellows subassembly failed preacceptance leak tests performed during fabrication. HR No. S.06 *No ET anomalies were reported on STS-33.*	A leak was detected during preacceptance testing of an LH_2 feedline bellows. Investigation determined that the leak was through a stress corrosion crack, 0.0025" x 0.018", extending nearly through the 3 plys of the bellows. This was the only time that this failure mode was experienced on the LH_2 feedline bellows. Investigation found that the crack originated during fabrication. Investigation also verified that only the Armco 21-6-9 corrosion resistant steel used in the bellows was affected. Five bellows fabrication sequences are used. Two of the sequences provide potential for water contamination and the environment necessary to originate stress corrosion cracking. During resistance welding, tap water was used for cooling and was thought to have become trapped between the plys (tap water has been replaced with demineralized water). Examination of the residue in the stress corrosion crack found the presence of chlorine, which was considered to be the corrosion source. Plastic strain, induced later in the fabrication process, opens any existing cracks. Martin Marietta analysis showed that the crack tip is blunted during the fabrication sizing step and that cracks do not grow after sizing. RI metallurgists concurred with this finding. Leak checks are performed after sizing. This is the step where the stress corrosion crack was found. All assemblies are later proof tested during the acceptance test process. Successful leak tests performed after sizing and during acceptance testing assure bellows integrity. Evaluation of the potential for crack initiation and growth after acceptance leak checks was performed. It was determined that a stress corrosion crack cannot be initiated due to the lack of aqueous solution in the bellows and lack of sufficient temperature to enhance corrosion, assuming there was aqueous solution left in the crack. Bellows are cured for 58 hr at 155°F during fabrication, resulting in the

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ET</u>		
1 (Continued)		baking out of any residual aqueous solution. The highest operating temperature seen by any of the bellows is 615°F for 8.5 minutes during flight; this is not considered sufficient time for further crack growth. Propagation of existing cracks at ambient temperatures is extremely slow or nonexistent. A fracture mechanics analysis was performed assuming a longitudinal crack twice as long as the largest measured, or 2.0" x 0.075". This worst-case crack would be through the outer ply. The results of this analysis found that the critical flaw size in flight is larger than 1.0". Mission life with critical flaws exceeds 1,000 missions. The conclusion was that there is no chance of a catastrophic failure of the bellows from stress corrosion cracks. Analysis also determined that the FOS for all bellows is greater than 10, with a maximum of almost 58 for jacketed bellows. Investigation was conducted into the effect of a crack, and resulting air leak, in the outer jacket of the bellows. A test was performed allowing air to replace argon in the bellows jacket. Cryogenics were then pumped through the bellows, and a fiber optic camera was inserted to record results. Cooling of the air was sufficient to form a slush; however, the air did not freeze hard. Air freezing in the jacket would reduce the flex capability of the bellows, which could result in failure. The rationale for flight was based on the following: • In-process checks identified the leak resulting from the stress corrosion crack. • The cause of the leak was determined to be stress corrosion cracking of the bellows due to chloride contamination.

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ET</u>		
1 (Continued)		• Bellows on STS-33/ET-034 successfully passed proof, acceptance, and leak tests. • Bellows have a high FOS; greater than 10. • Fracture analysis showed a large critical flaw size (greater than 1") and a high mission life capability (greater than 1,000 missions). • Cryopumping of air would not impair the flex capability of the bellows. ***This risk factor was resolved for STS-33.***

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>GFE</u>		
1	Crew parachute pyrotechnic cable cutter bent pins. HR No. PPA-0015 *No GFE anomalies were reported on STS-33.*	During parachute packing for STS-33 and STS-32, bent pins were observed on the 2 crew parachute pyrotechnic cable cutters that release the drogue cover flap. Release of this flap deploys a spring-loaded pilot parachute which in turn deploys the drogue chute. The failure was observed on November 3, 1989. Discrepancy Report (DR) 1W930016 was initiated; Failure Investigation Action Report (FIAR) JSC-EC-0411 was initiated. This failure is Crit 1R/3 for failure to operate, and Crit 1/1 for inadvertent operation. It is very unlikely that the pins can be bent by the crew during normal activity in any seat position. Ninety minutes in the trainer deliberately trying to bend the pin did not result in a bent pin. The pin will operate even if bent. There is no evidence that bending retracts the pin. The force required to bend the pin is a lot less than the force required to pull out the pin (8 lb versus 15 lb). The Naval Surface Weapons Center evaluation of the problem resulted in the following recommendations: • Provide protection for the pin similar to that provided on the manual D-ring backup drogue chute deployment. • Provide a protective sleeve over the pin. • Tie Kevlar directly to the pin. Rework was accomplished by Ellington Air Force Base personnel. Certification is by similarity. The safety requirement for an addendum to the hazard reports to reflect the configuration was completed.

RESOLVED STS-33 SAFETY RISK FACTORS

ELEMENT/ SEQ. NO.	RISK FACTOR	COMMENTS/RISK ACCEPTANCE RATIONALE
GFE		
1 (Continued)		Existing parachutes have been used on every flight from STS-26 to STS-34. Parachutes are inspected and repacked 6 times; parachutes must be repacked and inspected every 120 days. Parachute repacking and inspection schedule time must not exceed launch plus 30 days (accounts for launch delay). ***This risk factor was resolved for STS-33.***

SECTION 5

STS-34 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-34 mission. Each anomaly is briefly described, and risk acceptance information and rationale are provided.

SECTION 5 INDEX

ORBITER

1	Engine Interface Unit #3 momentary 60-kilobit data stream loss.
2	Auxiliary Power Unit #1 fault to high speed.
3	Multiplexer-Demultiplexer Flight-Critical Aft #1 Input/Output errors.
4	Auxiliary Power Unit #2 Gas Generator/Fuel Pump heater "A" inoperative.
5	Flash Evaporator System hi-load inboard duct temperature low.
6	Auxiliary Power Unit #3 seal leak into drain bottle.
7	Right Orbital Maneuvering System engine cover heater system "B" failed off.
8	Auxiliary Power Unit #2 fuel pump heater "B" cycling high.
9	Cryogenic Oxygen manifold #2 isolation valve did not close.
10	Right vent door #3 motor #1 operating on 2 phases.
11	External Tank/Orbiter Liquid Oxygen aft separation hole plugger failed.
12	Right-hand stop bolt was bent on centering ring of forward External Tank attach separation assembly.

SRB

1	Right Solid Rocket Booster Holddown Post #2 broached and shoe lifted from Mobile Launch Platform during liftoff.
2	Right Solid Rocket Booster forward segment missing Thermal Protection System from forward section of systems tunnel cover.

SRM

1	Left Solid Rocket Motor rock actuator bracket damage.
2	Left Solid Rocket Motor factory joint weather seal forward edge unbonds.
3	Putty on right Solid Rocket Motor igniter outer gasket and left Solid Rocket Motor igniter gasket retainer.
4	Left Solid Rocket Motor center field joint aft side unbond of K5NA closeout.
5	Left and right Solid Rocket Motor aft dome Ethylene Propylene Diene Monomer blisters.

KSC

1	Connectors on Solid Rocket Boosters improperly torqued and lockwired.

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Engine Interface Unit (EIU) #3 momentary 60-kilobit (Kbit) data stream loss. IFA No. STS-34-02 HR No. INTG-021A INTG-065 INTG-072 INTG-165 *No anomaly was reported on STS-33.*	EIU #3 Built-In Test Equipment (BITE) bit #13 set and the 60-Kbit data stream was lost, both momentarily. Recurrence would have resulted in loss of 60-Kbit data. The problem did not recur during this mission. Redundancy exists in the EIU for critical command and data paths; however, the BITE is not flight critical. The worst-case effect is loss of Space Shuttle Main Engine (SSME) performance data to the Operational Instrumentation (OI) recorder; the data is not mission essential. The 60-Kbit data is used to confirm critical Launch Commit Criteria (LCC) for Liquid Oxygen (LO_2) dome temperature and ice detection. The most probable cause was BITE circuitry failure. All 3 EIUs were scheduled to be replaced with modified EIUs during the STS-36 flow. *Not a safety concern for STS-33.*
2	Auxiliary Power Unit (APU) #1 fault to high speed. IFA No. STS-34-04 HR No. ORBI-031 *No anomaly was reported on STS-33.*	APU #1 experienced an uncommanded speed shift to the high-speed band at L+2.5 minutes during ascent. This speed shift was intermittent over a 4-5 second (sec) period and was permanent thereafter. The crew commanded the APU to high speed 15 sec after the uncommanded shift to avoid alarms. The APU operated satisfactorily at high speed for the remainder of the ascent. Troubleshooting found that the speed shift was due to the pulse control valve failing open. This valve varies fuel flow to the APU to regulate speed. After failure of the pulse control valve, the shutoff valve was used to control APU speed. APU #1 was not turned on until Mach 10 and was turned off at postlanding wheel stop. APU #1 Gas Generator Valve Module (GGVM) was sniffed checked, and insulation connectors were visually inspected; there was no indication of propellant leakage and no obvious external GGVM deformity.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
2 (Continued)		The APU controller was sent to Sundstrand for troubleshooting. During testing in an ambient temperature environment of 110°F, Sundstrand was able to duplicate this anomaly twice. The controller was left at ambient room temperature overnight and then returned to a temperature of 110°F. At the higher temperature, Sundstrand test conductors repeated the anomalous condition 5 additional times. The pulse code card in this controller was found to be susceptible to high thermal environments. The problem was isolated to an intermittent open 2N2222 transistor, Lot Date Code (LDC) 8131, in the pulse control circuit within APU Controller Serial Number (S/N) 311. This was the first Orbiter use of this Controller. The Line Replaceable Unit (LRU) criticality is 1R2. Analysis of the failed transistor indicated circuit failure to open was caused by depletion (thinning) of the gold wire at the transistor base junction due to "purple plague". Cross-sectional analysis showed extensive voiding between the bond wire and the pad. Five transistors on the same circuit board, with the same Part Number (P/N) and LDC as the failed part, passed electrical tests and were subsequently destructively analyzed. All passed nondestructive bond pull tests; 2 parts were destructively pull tested with good results. Two transistors were cross-sectioned and showed excellent bond-to-pad junctions. Analysis of 4 additional transistors from the Solid Rocket Booster (SRB) program, with the same P/N and LDC as the failed part, resulted in all findings being normal. Rationale for flight was based on an affected APU being able to operate in high speed while the other APUs operate normally. Increased risk due to APU overspeed was acceptable. There was no evidence to indicate that this was a generic failure problem. ***Not a safety concern for STS-33.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
3	Multiplexer-Demultiplexer (MDM) Flight-Critical Aft (FA) #1 Input/Output (I/O) errors. IFA No. STS-34-05 HR No. ORBI-038 *No MDM anomalies were reported on STS-33.*	At L+35 minutes, MDM FA #1 failure was detected by both the Primary Avionics Software System (PASS) and Backup Flight System (BFS) just prior to Orbital Maneuvering System (OMS) #2 burn. There was no response from the MDM primary port for 2 consecutive return word commands. I/O reset alone did not restore communications with MDM FA #1. Power cycling and I/O reset temporarily restored communications. The problem recurred and was restored by cycling power a few times. The crew was able to recover FA #1 operation for the remainder of the mission by moding to the backup port. Loss of the MDM FA primary port was acceptable during flight due to redundancy. Postflight troubleshooting at Dryden prior to MDM powerdown confirmed that port #1 was not communicating. The MDM was removed and replaced at Kennedy Space Center (KSC). While there have been 33 instances of loss of 1 MDM port, MDMs have 2 ports and re-porting can be accomplished. Both ports must fail in order to lose an MDM. Loss of 1 port prior to launch is acceptable. Certain critical MDMs are themselves redundant; therefore, loss of an MDM FA results in a minimum duration flight. Loss of a flight-critical MDM is an exception to the single-fault tolerant rule; loss of a single MDM reduces redundancy in multiple systems. This increases the risk that a single Line Replaceable Unit (LRU) failure in any 1 of several systems could put the Orbiter at a zero-fault tolerance level. Loss of 2 FA MDMs results in a next Primary Landing Site (PLS) mission termination. Rationale for flight of STS-33 was based on the acceptability of a loss of any single MDM FA port (internal redundancy) and the loss of 1 FA MDM (external redundancy and minimum duration flight). ***This anomaly was resolved for STS-33.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4	APU #2 Gas Generator (GG)/Fuel Pump (FP) heater "A" inoperative. IFA No. STS-34-06 HR No. ORBI-250 *No anomalies were reported on STS-33.*	APU #2 GG/FP system "A" heaters did not respond when selected. System "B" heaters were selected and operated acceptably; the "B" heater was cycling high. (See Orbiter 8 below.). Postflight testing at Dryden indicated that "A" heaters were operating properly. Thermostat S27A was removed and replaced; retest was successful. The thermostat worked properly during vacuum testing at Johnson Space Center (JSC). It was sent to Sundstrand, and testing has shown no problem. APU #2 heater system was thoroughly examined. It was totally rewired from forward panel A12 to the APU in the aft. Retest was performed per Operations and Maintenance Instruction (OMI) V1019. The same problem occurred on STS-27 and STS-30 missions. This is an OV-104 unique problem. *Not a safety concern for STS-33.*
5	Flash Evaporator System (FES) hi-load inboard duct temperature low. IFA No. STS-34-07 HR No. ORBI-276B *FES B outlet temperature oscillation occurred on STS-33 (IFA No. STS-33-13). This was not similar to the anomaly on STS-34.*	During ascent, post-Main Engine Cutoff (MECO), the hi-load inboard duct temperature was observed to be lower than expected. The flash evaporator is the primary heat sink during ascent, initialized at approximately 140,000 feet (ft) by avionics software. Both heaters were enabled on the hi-load duct. Approximately 3 minutes later, the crew shut down FES primary A and switched to secondary; the temperature continued to decrease. The system stabilized under radiator flow. Heaters were left on for bakeout. On flight day 2, the topping FES functioned properly on primary A and primary B controllers.

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
5 (Continued)		Data analysis indicated that the lower than expected duct temperature was created by high heat load and transients induced by the Radioisotope Thermoelectric Generator (RTG) cooling loop. RTGs will not fly again until STS-41. *Not a safety concern for STS-33.*
6	APU #3 seal leak into drain bottle. IFA No. STS-34-08 HR No. ORBI-100 *No anomaly was reported on STS-33.*	APU #3 cavity seal drain line pressure increased and fuel pump inlet pressure decreased. The possible cause of this anomaly was a leak in the static seal. The drain bottle was drained and checked at KSC to determine if the seal leak had degraded. The seal had not degraded; the catch bottle quantity was within acceptable limits. The same anomaly occurred on STS-30. The STS-30 drain bottle contained 30 cubic centimeter (cc) of propellant. This anomaly is unique for OV-104, APU #3. No leaks have been experienced an OV-103 or OV-102. The recent APU fuel pump detonation at Sundstrand highlights the concern with seal leaks. (See Section 4, Integration 3.) This leak should be fixed prior to STS-36, the next flight of OV-104. *Not a safety concern for STS-33.*

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
7	Right OMS engine cover heater system "B" failed off. IFA No. STS-34-09 HR No. ORBI-120 *No anomaly was reported on STS-33.*	During heater configuration to "B" heaters, the Aft Propulsion System (APS) right pod (RP03) "B" heaters failed to activate. The pod was removed. Investigation found a recessed pin in the heater connector. The connector was replaced. *Not a safety concern for STS-33.*
8	APU #2 fuel pump heater "B" cycling high. IFA No. STS-34-10 HR No. ORBI-250 *A similar anomaly (STS-33-16) was reported on APU #1 and APU #3 bypass line "A". Thermostats on APU #1 were removed and replaced. Changeout of APU #3 thermostats was deferred until after STS-31.*	APU fuel pump heater "B" cycled erratically toward higher temperatures. This anomaly was possibly related to the failure of APU #2 fuel pump heater "A". Thermostat S27B was removed and replaced; retest was successful. The thermostat was returned to Sundstrand for failure analysis. There was no indication of improper APU heater operation on OV-103 or OV-102 prior to STS-34. *Not a safety concern for STS-33.*

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	Cryogenic Oxygen (O_2) manifold #2 isolation valve did not close. IFA No. STS-34-12 HR No. ORBI-303A *No similar anomalies were reported on STS-33. There was, however, a sticky cryogenic O_2 check valve reported (STS-33-08).*	The crew attempted to close the cryogenic O_2 manifold tank #2 valve on panel R-1 per the sleep configuration. The crew reported that they held the switch for 5 seconds. There was no talkback. No switch discrete was received. The valve closed properly on the first troubleshooting step while on-orbit. Postflight, the valve opened properly. There is redundancy in the valves. Review of the valve design found that it will lose the closed indication when in the relief mode. Concern would be raised if the closed indications were not received during Main Propulsion System (MPS) dump or during reentry. *Not a safety concern for STS-33.*
10	Right vent door #3 motor #1 operating on 2 phases. IFA No. STS-34-19 HR No. ORBI-178A *No vent door anomalies were reported on STS-33.*	During prelaunch and landing configuration of vent doors, the right vent door #3 motor #1 operated on 2 phases. This occurred 3 times in flight. Phase B was lost when the door opened, and phase C was lost when the door closed. This anomaly also occurred with the same door motor during the STS-30 turnaround flow. During that flow, the problem occurred twice in 50 cycles. The motor control assembly was removed and replaced. Troubleshooting at Rockwell International (RI)/Downey could not duplicate any suspect relay failure during 900 relay cycles. Postflight troubleshooting at KSC repeated the anomaly; 1 phase of the Power Drive Unit (PDU) was found open. The vent door PDU was replaced and retested satisfactorily. Vent door motors are redundant and will operate on 2 phases, as demonstrated during STS-34. *Not a safety concern for STS-33.*

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11	External Tank (ET)/Orbiter Liquid Oxygen (LOX) aft separation hole plugger failed. IFA No. STS-34-20 HR No. ORBI-302A *No anomaly was reported on STS-33.*	The ET/Orbiter LOX aft separation hole plugger failed to extend fully by approximately 2" at ET separation. Postflight inspection found jamming caused by a detonator booster and detonator. A crushed backshell from the right aft connector was found on the runway after the ET umbilical door was opened. There was concern that loose debris could block the ET umbilical door from closing, resulting in the possible loss of the vehicle during reentry. Rationale for flight of subsequent missions was based on the probability being remote that escaping fragments would prevent ET umbilical door closure. The vehicle performs a maneuver at separation away from the ET and moves away from possible escaping debris prior to ET umbilical door closure. *This risk factor was acceptable for STS-33.*
12	Right-Hand (RH) stop bolt was bent on centering ring of the forward ET attach/separation assembly. IFA No. STS-34-21 HR No. INTG-051A *Postflight inspection found no bent stop bolts on the ET attach/separation assembly.*	STS-34 postflight inspection at Dryden found the RH stop bolt to be bent, forward and inboard. This bolt, located on the centering ring of the forward ET attach/separation assembly, was found compressed into the centering mechanism. It is used to restrict side motion at the attach/separation assembly between the ET and Orbiter and is not considered to be a structural bolt. Indications were that the assembly sustained a side load. The moment required to bend this bolt is in excess of 10,000 inch-pound (in-lb). The force required to obtain this moment is 900 pound (lb). A side load of this magnitude could lead to early, uncontrolled separation of the Orbiter from the ET. There was no indication that a side load occurred on STS-34 flight. The parts were removed at Dryden and sent to RI/Downey. Research indicated that Ground Support Equipment (GSE) used to mate the forward bipod probably caused the problem. The bolt was analyzed at RI/Downey.

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
12 (Continued)		The most probable cause of this anomaly was improper sequencing of the ET/Orbiter mating procedure resulting in a yaw moment that could bend the bolt. Sequencing employs GSE (H72-0590) that could produce the required loads. Improper sequencing would not lead to early, uncontrolled separation of the ET and Orbiter. However, a bent bolt extended into the airstream could result in excessive localized heating during reentry. There were no anomalies recorded during ET/Orbiter mating. ***This anomaly was resolved for STS-33.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1	Right SRB Holddown Post (HDP) #2 broached and shoe lifted from Mobile Launch Platform (MLP) during liftoff. IFA STS-34-B-01 HR No. INTG-164 B-00-15 B-00-17 *Stud hang-up and broaching did occur on STS-33, HDP #3 (STS-33-B-01 and STS-33-B-02). There was no recurrence of this problem on STS-32.*	The holddown stud at HDP #2 (RH tension post) hung-up at liftoff. This resulted in broaching of the right SRB aft skirt and thread impressions at that HDP bore. Review of the liftoff photographs found that the shoe on the MLP at HDP #2 lifted 2 1/4" at the same time. In addition, thread imprints were noted on 7 of the 8 SRB HDP feet during postflight evaluation. Analysis by United Space Booster, Inc. (USBI) and Marshall Space Flight Center (MSFC) indicated that vehicle launch performance would not be affected if all 8 studs hang up, provided that the frangible nuts are released. Stud hangups were recorded on 5 previous flights (STS-2, STS-4, STS-51I, STS-51J, and STS-61A). Major broaching of aft skirt HDPs was experienced on 4 prior flights. Minor broaching and thread impressions were recorded on 46 HDPs on 10 previous flights. Lifting of MLP holddown shoes was seen on STS-2 and STS-29. MSFC organized a tiger team to investigate stud hangups and the influence of recent design modifications on the holddown Debris Containment Systems (DCSs). Their review of liftoff film found similarities between the STS-34 occurrence and previous launches with stud hangups. Review of build papers relating to HDP installation revealed no anomalies. Frangible attach stud modifications to the debris containment device implemented prior to STS-28 should provide sufficient time for stud ejection. The modification reduced holddown stud ejection velocity from 222 inches per second (in/sec) to 184 in/sec. This increased the time for stud ejection from 52 millisecond (msec) to 63 msec. With a 250-msec SRB liftoff time, 63 msec should be sufficient time for stud ejection.

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1 (Continued)		The tiger team also analyzed the effect of biasing of MLP spherical bearings radially inward on the stud. Biasing of the spherical bearings was performed to increase the aft skirt Factor of Safety (FOS) for STS-34. The total HDP shoe/MLP spherical bearing mismatch for STS-34 was determined to be less than the mismatch on STS-27, STS-28, and STS-30. The compressive load on HDP #2 at frangible nut detonation was sufficient to prevent aft skirt shoe motion. The tiger team investigation determined that there was no evidence that biasing of the spherical bearings contributed to the stud hangup. MSFC analysis indicated that vehicle liftoff would be unaffected even if hangups occurred at all 8 HDPs, provided that all frangible nuts separated properly. An RI analysis conducted in conjunction with MSFC concluded that 1 or 2 stud hangups will not adversely affect vehicle liftoff dynamics or clearances between the vehicle and facility. The RI evaluation also concluded that the spherical bearing/shoe assembly will not break free and become a debris source. Rationale for flight of STS-33 was based on the MSFC analysis and the assertion which indicated that vehicle liftoff performance is unaffected even if hangups occurred at all 8 holddown posts, provided that all frangible nuts released. ***This anomaly was resolved for STS-33.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
2	Right SRB forward segment missing Thermal Protection System (TPS) from forward section of systems tunnel cover. IFA No. STS-34-B-02 HR No. B-60-25 Rev. C-DCN3 ***There were no reports of missing TPS on STS-33.***	A 6" wide by 24" long piece of Marshall Sprayable Ablator No. 1 (MSA-1) was missing from the forward section of a system tunnel cover on the right SRB forward skirt. This tunnel cover was the second cover from the top of the forward skirt. A clean substrate was observed, indicating no evidence of heat effects. Indications were that the piece of MSA-1 was dislodged at water impact. This was based on absence of sooting or heat effects that would result if the missing MSA-1 was lost during ascent. Fuzz was present on fractured edges of the MSA-1 and substrate; this was also an indication that water impact was the likely cause. Also, the resulting debris would not be in the Orbiter debris zone. Rationale for flight of STS-33 was based on the indication that the cause of the missing MSA-1 was water impact and, in any case, the resulting debris would not be in the Orbiter debris zone. ***This anomaly was resolved for STS-33.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Left Solid Rocket Motor (SRM) rock actuator bracket damage. IFA No. STS-34-M-01 HR No. BN-05 Rev. B ***There were no SRM anomalies reported on STS-33.***	During postflight inspection of the LH SRM aft exit cone, the 45° rock actuator bracket was found to be broken, taking part of the aft exit cone shell with it. The part of the bracket remaining on the actuator had a section of the aft exit cone shell (approximately 16" by 6") still attached. The aft exit cone, which contains 2 parts of the bracket, was shipped to Thiokol/Wasatch for further failure analysis. There were no reported functional anomalies during flight associated with the actuator and nozzle vectoring. The conclusion was that the actuator bracket broke on splashdown. Water impact loads on this SRM were higher than the strength of the bracket. The actuator becomes fixed (rigid) after motor separation. Delay in parachute opening, due to the reefer cutter failure, could have caused higher horizontal drift velocities. There was no soot on any surfaces exposed after the bracket failure, which also indicated that failure was caused by water impact. Excised samples from the actuator bracket were tested for mechanical properties; all properties (modulus, strength, and elongation) were at or above specification. Rationale for flight was based on the fact that the actuation system cannot develop loads large enough to fail the actuator bracket. Maximum actuation (stall) load is 103,424 lb; maximum measured actuation load is much lower. Actuator brackets are proof-tested to a 195,132-lb tensile load, with an additional 20,000-lb side load applied. The actuator bracket under flight loads was analyzed to have a structural FOS of 2.1. ***Not a safety concern for STS-33.***

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2	Left SRM factory joint weather seal forward edge unbonds. IFA No. STS-34-M-02 HR No. INTG-037B ***There were no SRM anomalies reported on STS-33.***	Three unbonds were noted on the LH SRM on STS-34. Two of these were on the forward edge. • Center forward factory joint (forward edge), 6.6" circumferentially by 1.75" deep at 0°. • Forward dome-to-cylinder factory joint (forward edge), 225° to 248°, with a maximum axial depth of 2.05". This was associated with paint failure and corrosion immediately adjacent to the debond seal. (The third unbond was at the forward factory joint (aft edge), 20" circumferentially by 3/4".) Adjacent paint was peeled from the case and attached to the edge of the Ethylene Propylene Diene Monomer (EPDM) at the area of the unbond. Corrosion was evident on the case under the EPDM and under the peeled paint. The factory joint unbonds were adhesive failures between the Chemlok 205 primer and the motor case. The weather seal was intact with no missing material. The concern was that unbonds could lead to debris potential during ascent and loss of factory weather seal protection. Structural assessment indicated that flight loads were not sufficient to create debris; the FOS is greater than 6.0. Protuberance loads on localized unbonds are much lower than loads necessary to tear EPDM. A completely unbonded weather seal would remain in-place and intact during flight. Postflight inspection found no sooting or heat affects under the weather seals, indicating that the unbonds occurred at splashdown.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
2 (Continued)		As corrective action, additional Conscan and surface finish requirements were added. All pin retainer band cleaning will be done prior to assembly to eliminate potential contaminants. For STS-33, a visual inspection and 0.005" shim stock edge probing were performed at Thiokol Corporation prior to paint closeout. *Not a safety concern for STS-33.*
3	Putty on right SRM igniter outer gasket and left SRM igniter gasket retainer. IFA No. STS-34-M-03 HR No. BC-03 Rev. B *There were no SRM anomalies reported on STS-33.*	Putty was found up to the aft face of the outer primary gasket and into the seal void/gland area, between 234° and 5° of the right SRM igniter. Putty was also found on the aft face of the gasket retainer (0.011" maximum) and under the retainer from 262° to 297° of the left SRM. The concern was that gasket sealing capability might be impaired by the embedded putty. Although there was no leakage or blowby past the seal (no blowhole in the putty), the seal is not designed to have putty on it. The cause of the problem was attributed to the igniter installation process. Putty located near the gasket and/or excess amounts may squeeze between the sealing surfaces during igniter installation. Corrective action was taken to improve putty layup and igniter installation processes based on results of previous installation tests. These tests demonstrated that putty does not enter the gasket area when laid up to tighter dimensional requirements (layup dimensions closely controlled, key layup dimensions recorded, and putty weight measurements monitored). This corrective action was implemented on the left SRM igniter on STS-33, on both SRM igniters on STS-32, and will be implemented on STS-36 and subsequent SRM igniters at Thiokol Corporation. In addition, after assembly all igniter joints are required to be leak checked for verification. *This risk factor was resolved for STS-33.*

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
4	Left SRM center field joint aft side unbond of K5NA closeout. IFA No. STS-34-M-04 HR No. INTG-037 B-60-24	The K5NA closeout on the trailing edge of the STS-34 LH SRM forward center field joint was debonded from the case wall. The cork leading edge at the 320° radial location also had K5NA debonded. The debond measured 5" circumferential by 1" in axial length. K5NA debonds are aeroheating and debris risks. Indications were that the debonds were caused by the nozzle severance sequence or at water impact. A scrape was found just aft of the unbond area, indicating debris impact from an external source. The unbonded K5NA remained in place. Since the unbond occurred after booster separation, there was no debris hazard to the Orbiter and no impact on flight safety. ***This anomaly was resolved for STS-33.***
5	Left and right SRM aft dome EPDM blisters. IFA No. STS-34-M-5 HR No. BC-10 Rev. B ***There were no SRM anomalies reported on STS-33.***	During disassembly of the booster assemblies, the Carbon Fiber-Filled (CFF)-EPDM in the aft dome of both SRMs was found to have ply blistering that resulted in separations. The separations were within a 32" band from the nozzle boss forward. Separations/blisters occurred intermittently for the full 360° around the SRM: 15 blisters randomly around the RH SRM and 10 randomly around the Left-Hand (LH) SRM. Examination found that the material separated between virgin plies. The largest separation occurred in the RH SRM, measuring 4.5" circumferentially by 5.5" axially. The smallest measured 1 square inch. Separated material appeared to be tacky, indicating that the EPDM may have been improperly cured. Approximately 0.030"-thick ply separated. Normal heat effects were evident in adjacent areas. Overall CFF-EPDM erosion was deemed nominal. The CFF-EPDM is in compression during firing, and the virgin CFF-EPDM is separated from chamber gas flow by a thick char layer.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRM		
5 (Continued)		Blisterings were localized occurrences; the edge of the blisters tore and did not propagate when pulled by hand. The CFF-EPDM lot used on STS-34 SRMs was also used on STS-28. There was no similar blistering occurrence experienced on STS-28. This same lot of CFF-EPDM was also used on STS-33 SRMs. Questions were raised concerning possible cold-soak conditions during transportation. Separations such as those found on STS-34 could have been caused by cold soaking. Thiokol records indicated that there were no cold weather conditions experienced during transportation to KSC. Rationale for flight for future missions was based on the assessment that, assuming total loss/erosion of the CFF-EPDM insulation at ignition, the remaining Nitrite Butadiene Rubber (NBR) provides a minimum FOS of 1.2. ***This anomaly was resolved for STS-33.***

C-2

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
KSC		
1	Connectors on SRBs improperly torqued and lockwired. IFA No. STS-34-K-03 HR No. B-00-15 *There were SRB connector anomalies reported on STS-33 (STS-33-K-01, STS-33-K-02, and STS-33-K-03). Efforts continue to highlight the need for proper connector installation. No adverse conditions resulted from these anomalies.*	During disassembly of STS-34 booster assemblies at KSC, 5 connectors were discovered with either no torque, no lockwire, or improperly installed lockwire. The following is a list of torque/lockwire discrepancies reported: • RH SRB, NASA Standard Initiator (NSI) B strut firing line jam nut was found not to be lockwired. • LH SRB, the jam nut on the 55-pin connector was lockwired in the wrong direction. • Coupling nut of the drogue deploy firing line on the RH forward Integrated Electronics Assembly (IEA) (J26) was found to be lockwired, but not torqued. • Coupling nut of the recovery battery cable on the LH forward IEA (J24) was found not to be lockwired. • LH range safety system coax connector in the rooster tail (at the aft skirt/rooster tail interface) was found to be lockwired not threaded. For the next flight (STS-33), all accessible/viewable connectors on the pad were inspected. Prior to rollout, all connector areas inaccessible at the pad were verified for proper lockwire installation by examination of closeout photographs. These areas included all strut firing line connections in the ET attach ring and all but 3 connectors at the LH forward skirt interface.

STS-34 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>KSC</u>		
1 (Continued)		Review of STS-33 "as-built" paper was performed to determine if assembly teams associated with the STS-34 installation were involved in the STS-33 installation. An assembly team is comprised of a Lockheed technician, a Lockheed quality inspector, and a NASA quality inspector. This review found that, while the exact same technician/inspector/inspector team did not perform any of the assemblies on STS-33, individuals were involved with STS-33 assemblies that did the STS-34 assemblies, but in different team compositions. All technicians and quality inspectors involved with the STS-33 assemblies were individually interviewed to determine if the assembly process, which could not be verified through visual inspection or closeout photographs, was properly performed and inspected. All interviewed indicated that all tasks were accomplished properly. Prior to the STS-33 Flight Readiness Review (FRR), the KSC Safety, Reliability, and Quality Assurance (SR&QA) Director reviewed the findings of the investigation with Space Shuttle Program Management to ensure that all concerns relative to the flight worthiness of STS-33 were met. Space Shuttle Program Management accepted any residual risk associated with the findings of the KSC investigation. For future flights, more stringent test and inspection procedures are being implemented to ensure proper connector installations. ***This anomaly was resolved for STS-33.***

SECTION 6

STS-29 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the STS-29 mission (previous flight of OV-103). Each anomaly is briefly described, and risk acceptance information and rationale are provided.

SECTION 6 INDEX

INTEGRATION

1 Excessive vapor at Liquid Hydrogen External Tank/Orbiter umbilical area during prelaunch and ascent.

ORBITER

1 Reaction Control System jet R1U failed off during mated coast.
2 Power Reactant Supply and Distribution cryogenic Hydrogen tank #3 pressure was high and manifold pressures were erratic.
3 Gaseous Oxygen Flow Control Valves can lock up due to contamination.
4 Payload bay door B "close" indication failure.
5 Water Spray Boiler #3 low relief valve reseat pressure.
6 Water Spray Boiler #1 exceeded specification leak rate.
7 17" Liquid Hydrogen disconnect leak.
8 Liquid Hydrogen 4" disconnect slow to close.
9 Flash Evaporator System primary controller B outlet oscillation.
10 Fuel Cell #1 water relief valve temperature overshoot.
11 Main Propulsion System Liquid Hydrogen feed manifold leak.
12 Hydraulic leak in aft compartment.
13 Thermal blankets in payload bay found to be loose and damaged.
14 Orbital Maneuvering System deck delta pressure anomaly.

SSME

1 Main Combustion Chamber aft bond line leak on engine #2031.
2 Cupwashers found rotated past stake on High Pressure Oxidizer Turbopumps #2222 and #4501.

SRB

1 Extensive damage to Solid Rocket Booster Thrust Vector Control components.
2 Structural crack found in the left aft skirt intermediate ring cap.
3 Debris Containment System plunger did not properly seat at Holddown Post #8.

SRM

1 Left aft center factory joint had several adhesive unbonds of the Ethylene Propylene Diene Monomer vulcanized weather seal.

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Excessive vapor at Liquid Hydrogen (LH_2) External Tank (ET)/Orbiter umbilical area during prelaunch and ascent. IFA No. STS-29-06 STS-29-ET-01 HR No. ET-S.06 INTG-015 *No vapor cloud was seen on STS-33. Both H_2 detectors read zero throughout ET fill operations.*	On STS-29, excessive vapor was observed during prelaunch and ascent. The quantity of vapor was larger than observed on past flights. The vapor cloud formation rate was estimated to be 1-3 cubic feet/minute. During periods of LH_2 fast fill, ET pressurization, and during Space Shuttle Main Engine (SSME) thrust buildup, heavy vapor was seen around the LH_2 ET/Orbiter umbilical. Ice/frost buildup was noted at the disconnect starting with ET fast fill. Numerous drops were also observed falling from the vicinity of the LH_2 umbilical. The drops produced vapor trails that may be indicative of cryogenic fluid vaporization. Ice/frost formed on the LH_2 umbilical, and a portion of the ET 17" feedline was reported to have one of the worst cases of buildup ever observed. During the 85% to 98% reduced fast fill procedures, slight melting of accumulated ice was observed in the vicinity of the 2" pressure disconnect, but it refroze a few minutes later in the same area. During the 98% to 100% fill topping operations, the ET vent valve was opened and the ullage pressure was reduced from approximately 48 pounds per square inch absolute (psia) to 18 psia in a short time period (approximately 10 minutes). At this time, rapid ice melting was observed on both the LH_2 umbilical and the ET 17" feedline, but all ice previously formed did not melt. The Ice Frost Team was instructed to carefully inspect the LH_2 umbilical. Upon inspection, they reported no evidence of an external hydrogen leak. Based on the Ice Frost Team report and other recommendations, the decision was made to fly as is. Review of post-launch film and video showed that the cloud and cryogenic droplets reappeared when the LH_2 system was brought up to flight pressure at about launch minus 2 minutes. The condition persisted and was exacerbated by engine start. The vapor cloud was in evidence as long as the cameras could view the area.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>INTEGRATION</u>		
1 (Continued)		Extensive investigation and analysis were conducted to determine the source and cause of the excessive vapor cloud. LH_2 leakage could not be ruled out, because an LH_2 leak will cause a vapor cloud and will produce liquid air. There was a possibility of a thermal short in the area of insulation removal and rework. A thermal short will contribute to formation of a vapor cloud and production of precipitation. Tests indicated that the cloud formation rate due to a thermal short increases with wind velocity until a velocity sufficient to disperse the cloud is experienced. This phenomenon was verified by analysis. Launch Commit Criteria (LCC) were developed for determining when vapor clouds and possible leaks in the LH_2/Orbiter umbilical area are of sufficient concern to stop the launch flow. Kennedy Space Center (KSC) installed 2 quick release Hydrogen (H_2) detector mechanisms to monitor the 17" disconnect area for LH_2 leaks during ET fill operations. No vapor cloud was observed in the LH_2/Orbiter umbilical area, and both H_2 detectors read zero throughout the ET fill operations for STS-30, STS-28, STS-34, and STS-33. ***This anomaly was resolved for STS-33.***

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
1	Reaction Control System (RCS) jet R1U failed off during mated coast. IFA No. STS-29-01 HR No. ORBI-203 *No RCS jet anomalies were reported on STS-33.*	RCS jet R1U failed off during mated coast due to low pressure. A trickle current test was performed which verified that the electrical path was good. It was suspected that the O_2 propellant valve failed closed. The jet was deselected by software, and redundant jets were available. The jet was removed and replaced prior to STS-33. *Not a safety concern for STS-33.*
2	Power Reactant Supply and Distribution (PRSD) cryogenic H_2 tank #3 pressure was high and manifold pressures were erratic. HR No. ORBI-089 *No PRSD H_2 tank pressure anomalies were reported on STS-33. There was, however, a PRSD O_2 tank check valve which stuck closed, allowing pressure to build in the O_2 manifold (STS-33-08).*	The PRSD cryogenic H_2 tank #3 pressure was erratic. Manifold pressures also indicated several pressure spikes, one of which resulted in exceeding the Maximum Expected Operating Pressure (MEOP) in the PRSD manifold. This condition caused the relief valve to crack until the pressure returned to an acceptable level. The pressure did not exceed the design pressure for the manifold. Further, the relief valve is sized to handle the maximum flow rate from the LH_2 tank with a Factor of Safety (FOS) of 5. The PRSD tank could have been isolated, and the fuel cells could have been operated from other tanks. The impact would have been to shorten the mission; not a safety impact. The physical mechanism that caused this phenomenon is not completely understood. Similar behavior occurred on STS-26 (same tank), but the MEOP redline was not violated. Circumstances peculiar to the STS-29 flight included: • Fuel cells were in the purge mode when the anomaly occurred; therefore, demand was high. • The tank had higher liquid content since it was earlier in the mission than for the STS-26 usage.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
2 (Continued)		• The tank was being used singly instead of in parallel with another tank as was the case on STS-26. Theories on how these factors contributed to the anomaly and the potential for peculiarities with this tank (none known) were postulated. Safety features designed into the system permitted retesting of the tank in orbit. There was no anomaly, and the tank was used additionally without incident. While further investigation is needed to understand the cause of the anomaly, the level of safety risk is small enough to warrant continued flight with these tanks. Rockwell International (RI)/Downey provided analysis and a written report to close this item. ***Not a safety concern for STS-33.***
3	Gaseous Oxygen (GOX) Flow Control Valves (FCVs) can lock up due to contamination. IFA No. STS-29-04 HR No. INTG-150A ***No GOX FCV anomalies were reported on STS-33.***	During checkout of the OV-104 GOX FCVs at KSC, it was found that in the pull-in/drop-out test at low voltage (0-8 volts (V)) the FCV for Main Engine (ME) #2 would not move from the high to low position (i.e., pull in). The valve was then cycled from 0 to 22 V; still no movement was observed. The valve was then energized with 28 V, and the armature moved to the low-flow position. Power was then removed, and the valve should have moved to the high-flow (i.e., drop out) position; no movement was observed. As a result of the testing, it was determined that the armature was jammed. All 3 valve solenoid/poppet assemblies were removed and sent to RI/Downey for failure analysis. The following results were observed: (1) all 3 valves were contaminated, with ME #2 valve the worst of the 3; and (2) the particulate was similar to that previously seen in these valves, with the size distribution conforming to a class 100 distribution with the exception of two 200+ micron-sized particles. The materials are 21-6-9 and 304 stainless, with the larger particles being the stainless.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3 (Continued)		These valves were cleaned on OV-103 after STS-26 and flew previously on OV-104 on STS-27. This contamination was not different from the contamination observed in every valve that was examined and was less severe than the contamination in 1 of the STS-26 valves. Tolerances were opened up; however, it appears that in-specification contamination can still lock up the valves. The rationale for risk acceptance to permit flight on STS-33 was essentially the same as that arrived at for STS-29. Triple redundancy is represented by the 3 GOX valves. A positive safety margin exists in all worst-case failure scenarios (the probability of which is considered very small), except Return to Launch Site (RTLS). Sluggish and delayed opening occurred again on STS-29 during the first cycle (valves #1 and #3). This resulted in a low ET/Liquid Oxygen (LO_2) tank ullage pressure. Additional delay response could result in tank structural limit violation. The cause of this anomaly was a combination of contamination and thermal transient effects. There was no mission degradation, and the valves cycled well on all cycles subsequent to the first (after the thermal transient period when temperature has stabilized). STS-30 was the first flight with no GOX FCV inflight anomalies since the return to flight, and no problems were experienced in subsequent flights. All 3 FCVs on OV-103 were inspected, cleaned, and polished since the last flight. The ET Oxygen (O_2) tank prepressurization level is to be lowered 2 pounds per square inch (psi) to ensure that the valves are open at launch and stay open for approximately 60 seconds (sec) into flight. Temperature stabilization occurs at approximately T+40 sec. Once the temperature was stabilized, the history of these valves indicates normal cycling occurs. ***Based on the steps taken to clean and polish the valves, and the lowering of the prepressurization level, this anomaly was resolved for STS-33.***

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
4	Payload bay door B "close" indication failure. IFA No. STS-29-09 HR No. ORBI-305A *No anomaly was reported on STS-33.*	The payload bay door port aft close limit switch in the ready-to-latch module failed to indicate "closed". The switch module was removed and sent to the vendor who verified a bad switch contact (switch #4). A new switch was installed and successfully tested. *This was an instrumentation problem and not a safety concern for STS-33.*
5	Water Spray Boiler (WSB) #3 low relief valve reseat pressure. IFA No. STS-29-10 HR No. INTG-072 INTG-113 *No similar WSB anomaly was reported on STS-33.*	WSB #3 relief valve appears to have reseated, then developed a slow leak approximately 10 minutes later. It was fully reseated at 26.7 psia. Testing could not repeat this anomaly. This anomaly was closed as unexplained. *Not a safety concern for STS-33.*
6	WSB #1 exceeded specification leak rate. IFA No. STS-29-11 HR No. INTG-072 INTG-113 *WSB #2 GN_2 leak rate exceeded specification by 0.06 psi/hr (STS-33-17). No problems were reported with WSB #1.*	WSB #1 exceeded the specification leak rate of 0.04 psi/hr. Subsequently, the leak stopped. Decay testing at KSC prior to STS-33 was successful. *Not a safety concern for STS-33.*

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
7	17" LH_2 disconnect leak. IFA No. STS-29-12 HR No. INTG-035 ET-P.07 *No similar anomaly was reported on STS-33.*	A blowing leak was discovered on the 17" LH_2 Orbiter disconnect during post-landing inspection. Inspectors found nicks on the flapper valve seats, and a black streak and small nicks on the inner bore. The cause of the nicks was not determined; however, investigation showed that they were present at ET/Orbiter mate. The actual leak rate was measured to be 1080 standard cubic inches per minute (scim) versus the specification allowable of 1000 scim. Subsequent to OV-103 return to KSC, the 17" Quick Disconnect (QD) was cleaned, and a following leak test was within specification (209 scims). The leak was determined to be caused by build-up of dry lubrication on the flapper valve. The flapper was cleaned and relubricated prior to STS-33. *This anomaly was resolved for STS-33.*
8	LH_2 4" disconnect slow to close. IFA No. STS-29-13 HR No. ORBI-035 *No anomaly was reported on STS-33.*	The time from close power applied to close was approximately 5 sec; the specification is 1.2 sec maximum. Ambient tests of OV-103 indicated no problem. Belleville spring inspection on OV-103 also indicated no problem. The valve/actuator was removed and tested to verify the failure mode. It was determined that the actuator was faulty at cryogenic temperatures due to a cracked isolator. A replacement unit was installed. *Not a safety concern for STS-33.*

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
9	Flash Evaporator System (FES) primary controller B outlet oscillation. IFA No. STS-29-14 HR No. ORBI-276B ORBI-300 *This anomaly recurred on STS-33 (STS-33-13). Temperature sensors on FES B will be replaced prior to STS-31.*	During 3 different inflight startups, the FES control temperature oscillated between 38°F and 41°F and damped out in approximately 6 cycles. The probable cause lies in the FES primary B control or midpoint temperature sensors. This phenomenon contributed to momentary FES shutdown during reentry. The sensors were repacked, reinstalled, and successfully retested. *Not a safety concern for STS-33.*
10	Fuel Cell (FC) #1 H_2O relief valve temperature overshoot. IFA No. STS-29-16 HR No. ORBI-284 *No anomaly was reported on STS-33.*	The crew configured FC #1 H_2O relief heaters to the B-Auto position per the heater reconfiguration on the morning of Flight Day 3. The B thermostat immediately turned the heater on, because its temperature was 70°F. The temperature rose to 130°F before there was a normal H_2O line cooldown. STS-26 data on this thermostat showed that the temperature never rose above 105°F during its cycling, when OV-103 was in a cool attitude. The thermostat was removed, replaced, and successfully retested at KSC prior to STS-29 flight. Thermostats #1 and #2 were swapped prior to STS-33, and retest found readings to be satisfactory. *Not a safety concern for STS-33.*

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
11	Main Propulsion System (MPS) LH_2 feed manifold leak. IFA No. STS-29-21 HR No. ME-B1A ME-B1C ME-B1M ME-B1S ***No anomaly was reported on STS-33.***	The relief valve cycled only once after Main Engine Cutoff (MECO), indicating a possible manifold leak. Postflight investigation indicated no evidence of a leak. Valve operation is within previous flight experience, although on the low side. RI/Downey provided rationale for closure of this anomaly. ***This anomaly was resolved for STS-33.***
12	Hydraulic leak in aft compartment. IFA No. STS-29-23 HR No. ORBI-036 INTG-016 ***No anomaly was reported on STS-33.***	Postflight inspection found hydraulic fluid in the aft compartment. Inspection found a loose B-nut in the leakage collection line from the SSME #1 accumulator. One-half to 1 ounce (oz) of hydraulic fluid was found. Hydraulic leaks in the aft compartment are a major concern for loss of hydraulics and the potential for fire and explosion. A fine spraying leak poses the greatest threat for fire and explosion. However, this is a low-pressure system, and a spraying leak is not a concern. The quantity was low because the leak occurred at the accumulator overflow. The B-nuts were not secured because holes were not drilled for the wiring. However, this is not a high-pressure system, and proper torquing should suffice. The B-nuts were inspected and checked again as part of the applicable Operations and Maintenance Instruction (OMI). The system was in the correct configuration for STS-33. ***This anomaly was resolved for STS-33.***

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
13	Thermal blankets in payload bay found to be loose and damaged. IFA No. STS-29-25 HR No. ORBI-249A ***There were no loose thermal blankets found after STS-33.***	Postflight inspection of the STS-29 payload bay found 4 thermal blankets loose and damaged. The damaged thermal blankets were located at the 1307 bulkhead, where the C-stiffeners were modified. Similar looseness and damage were experienced on STS-26 and STS-27; however, only 1 thermal blanket on each flight was affected. Damaged blankets could result in debris contamination in the payload bay and potentially adversely interfere with the safe operation of some payload equipment. The probable cause of the damage was air/venting of the payload bay and the acoustical environment during ascent causing the blankets to vibrate and flap. A new design was approved for the thermal blankets to cover the aft side with Beta cloth to protect the aluminized Mylar. Thermal blankets in the affected area (1307 bulkhead) were replaced with new design blankets on OV-103. ***This anomaly was resolved for STS-33 by installation of the new design blankets.***

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
1	Main Combustion Chamber (MCC) aft bond line leak on engine #2031. IFA No. STS-29-E-02 HR No. ME-B5A ME-B5C ME-B5M ME-B5S *No anomaly was reported on STS-33.*	Postflight leak check and borescope inspection of engine #2031 indicated a small leak in the MCC/nozzle bond line which was internal to the nozzle. Two visible leaks, approximately 0.020" and 0.060", were confirmed by ultrasonic inspection; each was coincident with a nozzle protrusion of 0.028". Engine flight data showed no evidence of a leak. The Failure Modes and Effects Analysis/Critical Item List (FMEA/CIL) lists the aft region internal fuel leak as Criticality 1. If the leak occurred rapidly and was large enough, the High-Pressure Fuel Turbopump (HPFTP) would cavitate, and the pump and engine would operate oxidizer rich. This could result in a catastrophic failure of the engine and result in loss of the Orbiter and crew. There was no fabrication or assembly history indicative of a problem on this unit. It had a history of 10 starts and 2580 sec. The liner aft corner was trimmed for additional nozzle clearance (the bond line was not machined). There has been no liner-to-nozzle tube contact on any prior assembly. OV-103 units passed leak test and had no proof test disbond indicative of marginal bonding. Proof test should detect any gross bond deficiency. Disbond propagation rate is slow based on previous testing. The 3 OV-103 engines and the remaining engines in the field were ultrasonically inspected; no anomalies were found. ***Based on the review of flight data, inspections performed, and test results, this anomaly was resolved for STS-33.***

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SSME</u>		
2	Cupwashers found rotated past stake on High Pressure Oxidizer Turbopumps (HPOTPs) #2222 and #4501. IFA Nos. STS-29-E-04 STS-29-E-05 HR No. ME-C1A Rev. A ME-C1C ME-C1D ME-C1M ME-C1P ME-C1S ***No rotated cupwashers were found on HPOTPs of STS-33 SSMEs.***	During disassembly of HPOTP 2222R-1 from STS-29, 3 of 11 cupwashers were found to have rotated (loose). Disassembly of HPOTP 4501R-1, also from STS-29, found 2 of 11 cupwashers experienced a similar occurrence. Investigation indicated no cup cracking. This was the first time that detente had been overridden without cracking. There have been no problems of this kind since 1986. Cup cracking problems involve material and material hardness deficiencies, but the material used in these cupwashers met requirements. Also, the staking processing was reviewed and was satisfactory. The rotation was never seen on Left-Hand (LH) threaded cup-seal applications, only on the Right-Hand (RH) threaded applications. This indicated the possibility of a force generating enough torque to back out the cupwasher. Rationale for the flight of STS-33 included: • Screw preload is not functionally required to carry loads. The delta pressure load is in the direction of seating the retainer/silver seal against the RH vane (screws do not carry the loads). • Maximum cupwasher rotation experienced to date is 75°, resulting in a margin of 4 for one full rotation. A minimum of two rotations are required to disengage the locking feature. • No evidence of cracking seen on a rotated cupwasher. • No evidence of heating or ignition in the fretted areas. ***This risk factor was acceptable for STS-33.***

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
1	Extensive damage to Solid Rocket Booster (SRB) Thrust Vector Control (TVC) components. IFA No. STS-29-B-01 STS-29-B-03 HR No. A-20-03 Rev. B A-20-18 Rev. C ***No further anomaly was reported on STS-33.***	The TVC component damage on both SRBs was considered to be due to hydrazine reaction/fire. Indications were that a hydrazine fire was caused during SRB descent following separation of the nozzle extension at apogee. Signs of a hydrazine fire were seen previously only on STS-1; this was the only other flight where the nozzle extension was severed at apogee. On all other flights, the nozzle extension was severed 20 sec after low baroswitch operation. The change to separation at apogee for STS-29 was made to alleviate nozzle damage to the main parachutes after separation; this has been seen on many flights. The extensive damage to TVC components caused by the resulting hydrazine fire could limit reuse of the aft segments of the SRB used on STS-29. For STS-30 and subsequent (SUBS) flights, nozzle severance was changed back to 20 sec after low baroswitch operation. ***Not a safety concern for STS-33 because of return to previous mode of nozzle extension severance at low baroswitch.***
2	Structural crack found in the left aft skirt intermediate ring cap. IFA No. STS-29-B-02 HR No. INTG-158B ***No further anomaly was reported on STS-33.***	Inspection revealed several areas of missing foam around the intermediate ring. The ring cap crack completely penetrated at the fillet radius runout. Damage was attributed to the greater than usual water impact loads on the left SRB. Metallurgical examination of the fractured surface also indicated that the fracture occurred due to impact. ***Not a safety concern for STS-33.***

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
3	Debris Containment System (DCS) plunger did not properly seat at Holddown Post (HDP) #8. IFA No. STS-29-B-04 HR No. B-60-12 Rev. B ***While no NASA Standard Initiator (NSI) debris escaped, the epon shim on HDP #3 was lost. This condition was related to the stud hang-up and broaching at HDP #3 (STS-33-B-01 and STS-33-B-02).***	Postflight inspection of HDP #8 identified several debris chunks missing; most of the NSI booster cartridge and 3 large slivers of the frangible nut. Follow-up inspection at the pad found debris in the post #8 sand box matching the aforementioned missing pieces. This evidence supported the position that the plunger failed to completely seat at liftoff. A total of 10 oz of debris was lost out of a potential 135 oz. Modifications were made and installed for STS-28 and subsequent flights to improve the DCS. Changes include addition of a silicone rubber shock isolator between the stud attachment and plug tip, and material/configuration changes on the stud attachment. Debris containment has now improved to the point where essentially all the debris is being contained. ***Not a safety concern for STS-33.***

STS-29 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRM</u>		
1	Left aft center factory joint had several adhesive unbonds of the Ethylene Propylene Diene Monomer (EPDM) vulcanized weather seal. IFA No. STS-29-M-02 HR No. BC-02 Rev. B BC-05 Rev. B *No anomaly was reported on STS-33.*	Postflight inspection of the left aft center factory joint revealed several adhesive unbonds of the EPDM vulcanized weather seal. All unbonds of the weather seal were adhesive failures (not cohesive). It appears that contamination was the likely mechanism which prevented an acceptable bond of the SRM case to the Chemlok (the bonding agent upon which the EPDM weather seal is vulcanized). Inspection of the pin retainer band verified no damage/breakage, but the band was noted as nominally stretched. Radial expansion of the band is expected during pressurization of joints. The damage was attributed to water impact. *Not a safety concern for STS-33.*

SECTION 7

STS-33 INFLIGHT ANOMALIES

This section contains a list of Inflight Anomalies (IFAs) arising from the OV-103/STS-33 mission. Each anomaly is briefly described, and risk acceptance information and rationale are provided.

SECTION 7 INDEX

INTEGRATION

1	Space Shuttle Main Engine #2107 nozzle bluing.

ORBITER

1	Auxiliary Power Unit #1 lube oil output pressure was high.
2	Cabin air leak through the Waste Collection System.
3	Reaction Control System F1U pressure transducer failure.
4	Commanders' airspeed mach indicator out of specification.
5	Hydraulic Systems #1 and #2 accumulator ascent pressure locked-up low.
6	Power Reactant Storage and Distribution Oxygen tank #1 had a sticky Check Valve.
7	Forward attach point system A and system B connectors found damaged.
8	"Y" Star Tracker door thermal blanket detached.
9	Flash Evaporator System B outlet temperature oscillation.
10	Erratic temperature indication from Auxiliary Power Unit #1 and #3 bypass line "A".
11	Hydraulic System #2 Water Spray Boiler Gaseous Nitrogen leakage was out of specification.

SRB

1	Holddown Post anomalies.
2	Left-hand External Tank Attachment Ring Integrated Electronic Assembly end cover and cable sooted.

KSC

1	Improper installation of cable connector assemblies.

STS-33 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
INTEGRATION		
1	Space Shuttle Main Engine (SSME) #2107 nozzle bluing. IFA No. STS-33-I-01	Postflight visual inspection of Main Engine (ME) #2107 nozzle revealed discoloration or "bluing" on the front face of the aft manifold. The discoloration was centered about the lower centerline (±1.5 feet (ft)), low reentry heating region. Nozzle structure is uninsulated in this region (Inconel 718). No discoloration was evident on ME #2031 nozzle. Discoloration in this region was not observed in previous flight experience. The nozzle discoloration cannot be explained by the predicted heating environment. The time/cause of the discoloration is not yet understood. Worst-case recurrence would impact nozzle reuse. This Flight Problem Report was approved at the Level II Noon Program Requirements Control Board (PRCB) on February 8, 1990. Per Dr. Lenoir at the STS-36 Flight Readiness Review (FRR), this Inflight Anomaly (IFA) was reopened. Further data was requested from other flights using new ME nozzles.

STS-33 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
1	Auxiliary Power Unit (APU) #1 lube oil output pressure was high. IFA No. STS-33-01 HR No. ORBI-036	APU #1 experienced higher than normal lube oil output pressure during ascent. Pressure peaked at approximately 85 pounds per square inch (psi), 25 psi higher than normal. The pressure returned to normal just prior to Main Engine Cutoff (MECO). Two waivers, 1 for high APU gearbox delta pressure and the other for high APU gearbox blanket pressure, were approved prior to STS-33 launch. The seal cavity pressure was higher than the gearbox pressure due to a procedural error, allowing hydrazine seepage into the gearbox. A wax substance, pentaerythritoral, is formed when hydrazine is mixed with lube oil. This substance goes back into solution between 175-200°F, the nominal APU operating temperature. Kennedy Space Center (KSC) performed oil flush and drain, as well as lube oil filter changeout per Operational Maintenance Requirements and Specifications Document/Operations and Maintenance Instruction (OMRSD/OMI) V10078, prior to the next OV-103 flight. KSC was directed to double-bag the filter and send it to Rockwell International (RI)/Downey for analysis. Oil samples were taken prior to system flush.
2	Cabin air leak through the Waste Collection System (WCS). IFA No. STS-33-02 HR No. ORBI-077	Cabin pressure decreased to 14.28 pounds per square inch absolute (psia) before the leak was isolated. The crew isolated the leak to coincide with WCS usage. The leak was verified when the commode slide valve was opened and no discernable air flow was noted. Air transportation of fecal matter was also lost. The crew performed inflight maintenance to manually move the vacuum ball valve from vacuum position to FAN SEP position. Cabin pressure was restored as well as full WCS operation. Inspection of the OV-103 WCS at Dryden by Johnson Space Center (JSC)/Hamilton Standard found a broken pin on the linkage between the handle and the relief valve. Further investigation determined that the wrong pin was installed. OV-102 was checked prior to STS-32 and found to be correct. OV-104 will be inspected at KSC prior to STS-36.

STS-33 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
3	Reaction Control System (RCS) F1U pressure transducer failure. IFA No. STS-33-04A HR No. ORBI-203	The RCS F1U chamber pressure transducer failed during Flight Control System (FCS) checkout in preparation for reentry. Indications were that the jet fired properly on ascent. For reentry, F1U was deselected due to the low chamber pressure indication and was not required for the remainder of the mission. Similar instances of low RCS thruster chamber pressure were experienced on 3 previous flights on all Orbiters. A decision has been made not to repair this transducer until after STS-31, since this jet is mainly used for proximity missions only and STS-31 is not a rendezvous mission.
4	Commander's airspeed mach indicator out of specification. IFA No. STS-33-05	During FCS checkout, the Commander's airspeed mach indicator read 20,500 feet per second (fps); specification is 20,000 fps. This problem was also reported on the 2 previous OV-103 missions since reflight. On STS-26, it read 22,250 fps (IFA No. STS-26-20); on STS-29, 22,050 fps. This anomaly is isolated to OV-103.
5	Hydraulic Systems #1 and #2 accumulator ascent pressure locked-up low. IFA No. STS-33-07	During ascent, Hydraulic Systems #1 and #2 accumulator pressure locked-up low. This anomaly is similar to a problem on STS-26 and STS-29 (IFA No. STS-29-26) where priority valves #1 and #2 experienced low reseats at APU shutdown. The valves are required to lock up at 2600 pounds per square inch differential (psid) pressure (referenced to reservoir pressure). After STS-33 ascent, priority valve #1 locked-up at 2420 psid; valve #2 locked-up at 2340 psid. Lockups have been repeatable during the 3 OV-103 flights since reflight and show no sign of further degradation. During special testing at KSC, 2 of the 6 lockups were below specification. There is no immediate system concern; therefore, these valves were allowed to fly as-is for STS-33. However, the valves were known to be out-of-specification. It is believed that the valves were set low during acceptance testing at the vendor or changed with time. These valves had never flown prior to STS-26. There was no evidence of problems with the priority valves on OV-102 and OV-104 missions since reflight. The 2 valves have been removed and replaced.

STS-33 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
6	Power Reactant Storage and Distribution (PRSD) Oxygen (O_2) tank #1 had a sticky Check Valve (CV). IFA No. STS-33-08 HR No. ORBI-094	PRSD O_2 tank #1 CV stuck twice during the mission. This O_2 tank was not in use when the anomalies occurred. When a 20-psid pressure difference built up across the CV, it opened and operated nominally. Nominal cracking pressure is 3-5 psid. This particular CV experienced a large 180-psid closing force after high O_2 flow associated with the pressure leak through the WCS (see Orbiter 2 above). Stopping the high O_2 flow caused Liquid Oxygen (LO_2) to be trapped in the manifold. Environmental heat converted the LO_2 to gas and pressurized the manifold until the relief valve opened. The CV operated nominally for the remainder of the mission. Sticking CVs have been observed on previous flights subsequent to large closing forces. No remedial action is required. It is believed that this anomaly was caused by transient contamination, compounded by the high checking force during the pressure leak through the WCS.
7	Forward attach point system A and system B connectors found damaged. IFA No. STS-33-10 HR No. ORBI-289	During Orbiter inspection at Dryden, it was found that the tangs on both system A and system B pyro connectors were clocked incorrectly. Clocking was at 30° aft instead of straight-up. One connector (20V77W11J13) had a broken strain relief; the other connector (20V77W12J12) had a loose backshell. A known interference problem existed between these connectors and the forward attach pyro bolt; it is very alignment sensitive. All connectors and harnesses in this area were replaced prior to each flight. JSC and RI engineers prepared a design change to replace these connectors with 90° backshell connectors. This change will correct the interference problem.

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
ORBITER		
8	"Y" Star Tracker door thermal blanket detached. IFA No. STS-33-11 HR No. ORBI-011A	The "Y" Star Tracker door thermal blanket was found totally detached from the door and lying loose on the bottom of the Star Tracker cavity. The blanket was not damaged at the attach points. A small tear on the top of the blanket indicated that it was detached when the door closed. No fastener damage was observed. Investigation of problems during Star Tracker door cycling on OV-104, prior to rollout for STS-34, found that the thermal blankets interfered with the bright-object sensor. Redesigned thermal blankets were installed on all Orbiters. There were no reported problems with the modified blankets on STS-34. Worst-case effects of loose thermal blankets are related to jamming the Star Tracker doors open during reentry. This would allow plasma flow through the cavity resulting in damage to the Star Tracker. Recent RI thermal analysis indicated that the thermal blankets in the Star Tracker cavity were not necessary, based on redefined heating environments. A recommendation was made by RI at the STS-32 FRR to remove these blankets prior to launch. The recommendation was subsequently approved by the PRCB, and the blankets were removed.
9	Flash Evaporator System (FES) B outlet temperature oscillation. IFA No. STS-33-13 HR No. ORBI-276B ORBI-300	During FES B deorbit preparation, when FES B was reconfigured from the "PRI B ON" to the "PRI B GPC" position, it shut down because FES B was above the temperature limits. This was due to FES B inability to bring control band temperatures within shutdown logic limitations. A similar occurrence was experienced on STS-29 (IFA No. STS-29-14). Prior to STS-33, the midpoint sensors were repacked due to a lag that existed between the midpoint temperature sensor and actual Freon Coolant Loop (FCL) temperatures. This was caused by a midpoint sensor manifold design change for OV-103 only, which should have rectified this problem.

STS-33 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>ORBITER</u>		
9 (Continued)		After the first occurrence of this anomaly on STS-33, FES B was recycled; this successfully brought the temperature into the control band before the shutdown logic timed out. FES B operated nominally for the remainder of the flight. This anomaly was believed to have been caused by a tolerance build-up in the lead/lag times of Controller "B" and its 3 temperature sensors.
10	Erratic temperature indications from APUs #1 and #3 bypass line "A". IFA No. STS-33-16 HR No. ORBI-250	Bypass line "A" temperature sensors on both APU #1 and APU #3 demonstrated erratic behavior. This was indicated by erratic bypass line heater operation. The temperature sensors, or thermostats, are mounted on the APU bypass lines. It is believed that these lines experienced vibration which led to loosening of the thermostat sensor mounts. A determination was made to replace both the "A" and "B" temperature sensors on both APUs. APU #1 temperature sensors were replaced and tested satisfactorily. A decision was made to delay the replacement of APU #3 temperature sensors until after STS-31, when the entire APU will be replaced.
11	Hydraulic System #2 Water Spray Boiler (WSB) Gaseous Nitrogen (GN_2) leakage was out of specification. IFA No. STS-33-17 HR No. INTG-072 INTG-113	During on-orbit operations, the WSB for hydraulic system #2 demonstrated excessive GN_2 leakage. Some decay in GN_2 tank pressure is expected. Leakage on STS-33 was at a rate of 0.36 pounds per square inch/hour (psi/hr); the specification limit is 0.30 psi/hr. A similar anomaly was experienced during STS-29 on WSB #1.

STS-33 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1	Holddown Post (HDP) anomalies. IFA No. STS-33-B-01 IFA No. STS-33-B-02 HR No. INTG-164 B-00-15 B-00-17	Orbiter accelerometer readings at Solid Rocket Booster (SRB) ignition indicated a holddown bolt anomaly. The launch film showed the stud at HDP #3 hung-up, similar to the occurrence on STS-34. The stud extended approximately 8" and contacted the aft skirt stud hole wall. This may have caused a piece of the epon shim to pull loose and separate from the skirt foot. An area of epon shim material (approximately 34 square inches) from the bottom of the right SRB HDP #3 was observed falling off during the launch. An RI evaluation of this type of anomaly concluded that the probability of shim material ricocheting and impacting the vehicle is extremely remote as the primary forces acting on the shim particles are gravity, plume impingement, and aspiration. Postflight inspection of the Right-Hand (RH) aft skirt found that it had been broached on the aft side of the HDP #3 bolt hole. Thread impressions were also visible on the forward side of the same hole. One of the 2 pyrotechnic charges used on each frangible nut did not appear to explode properly on HDPs #3, #4, and #8. The frangible nut separation area showed a ductile separation. Nominal operation of the pyrotechnics causes splintering of the nut material at the explosion site. The cause of ductile separation as seen on these nuts is as yet inconclusive. It could indicate explosion was either less powerful than desired or late. The anomalous pyro action might have contributed to the stud hang-up at HDP #3. HDP broaching has occurred on several previous flights, most recently on STS-34. Rationale for STS-33 launch, the next flight after STS-34, was that a Marshall Space Flight Center (MSFC) and RI integration analysis indicated that all 8 HDP bolts could hang-up with no deleterious liftoff performance effects, provided that all frangible nuts are released. However, the potential problem experienced with the skewed firing of the frangible nut pyrotechnic charges identified the need for further

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
SRB		
1 (Continued)		analysis relative to the influence and contribution of the bolt hang-up at HDP #3 and liftoff performance degradation. Further analysis of stud hangup yielded the following information: • Worst-case hangup was defined as the hangup of all 4 studs on 1 SRB. • Worst-case hangup has a minimal effect on post and tower clearance. • Worst-case hangup has a negligible effect on flight controllability. • Worst-case hangups could cause the limit load to be exceeded on some External Tank (ET) and/or SRB hardware based on a conservative quick look analysis (4 stud hangups could possibly reach 1.2 to 1.4 times the limit load). One, 2, or 3 stud hangups yields loads within limit loads. • The probability of a worst-case 4-stud hangup is less than 2 times 10^{-5} with removal of the plunger-to-stud frangible bolt. • RI load analysis concluded that the structure can withstand a 4-post worst-case load plus 3σ dispersed loads. Some SRB personnel believe that stud hangup can be minimized by incorporating a 0.030" bias in the alignment of the skirt to the Mobile Launch Platform (MLP) support post. The incorporation of this bias before assembly is expected to compensate for flexure of the structure due to the loading of the aft skirt during assembly. The MLP spherical bearings would then be properly aligned and allow maximum clearance between the holddown bolt and the bolt hole, thereby significantly reducing the likelihood of holddown bolt hangup.

STS-33 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
<u>SRB</u>		
2	Left-Hand (LH) ET Attachment (ETA) Ring Integrated Electronic Assembly (IEA) end cover and cable sooted. IFA No. STS-33-B-03	Upon removal of the LH IEA covers, sooting was noted on 16 cables and interior painted surfaces of the end cover. Examination of the cable jacket indicated no heating effects (no erosion, clouding of material, or degradation). It was determined that the gap in the RTV-133 sealant allowed hot gases to enter the ETA ring and the IEA cable areas through the aft side of the IEA end cover. The gases entered at the aft side of the end cover, traveled across the wire bundles, and exited through the opposite (forward) side of the end cover. This was determined by the heaviest sooting deposits on the aft side of the IEA end cover and the flow pattern. The direction of hot gas flow entering the end cover indicated that this condition occurred during reentry or descent. The RTV-133 material was missing at the area of soot entry and exit. All cables functioned properly during the mission. There was not adequate heat present to damage the cables or impair the cable function. Corrective action consists of an engineering change (Field Engineering Change (FEC)-10266) that will be effective for STS-32, STS-36, STS-31, and STS-35; Engineering Change Proposal (ECP)-2670 will make this revision to the closeout procedures permanent. This change clarifies the Thermal Protection System (TPS) closeout, thereby assuring proper closeout and preventing recurrence of this anomaly.

STS-33 INFLIGHT ANOMALIES

ELEMENT/ SEQ. NO.	ANOMALY	COMMENTS/RISK ACCEPTANCE RATIONALE
KSC		
1	Improper installation of cable connector assemblies. IFA No. STS-33-K-01 STS-33-K-02 STS-33-K-03	During STS-33 postflight assessment, 2 cable connectors were found incorrectly installed and 2 ground straps were loose due to omitted washers. • The RH forward skirt Range Safety System (RSS) Ground Support Equipment (GSE) cable (Radio Frequency (RF) signal to the Integrated Receiver/Decoder (IRD)) was not fully seated on its mating connector at the forward feed-through. The connector was engaged only 3/4 of a turn; 3 1/2 turns are required for full engagement. The connector was lockwired correctly. The connector insert showed signs of moisture and contained K5NA debris. This cable is not used in flight, but is used during range safety ground checkout. • The LH upper strut separation ordnance connector was finger-loose. The connector was lockwired correctly. The jam nut was retorqued to determine the relationship of the lockwire to the properly torqued connector. Slack in the lockwire indicated that the connector had not been properly torqued prior to lockwire installation. • Two ground straps located between the RH SRB aft IEA bracket and the SRM were loose. The ground strap fasteners bottomed out due to omitted washers. Some washers had not been installed on the fasteners on the forward end of the IEA, but those fasteners had not bottomed out and the ground straps were not loose. All 4 bolts were torqued properly (125-150 inch-pound (in-lb)). The LH brackets had washers installed.

SECTION 8

BACKGROUND INFORMATION

This section contains pertinent background information on the safety risk factors and anomalies addressed in Sections 3 through 7. Information from this section is available upon request.

APPENDIX A

LIST OF ACRONYMS

AFB	Air Force Base
APS	Aft Propulsion System
APU	Auxiliary Power Unit
ARF	Acceptance Refurbishment Facility
ASA	Aerosurface Amplifier
ATP	Acceptance Test Procedure
ATVC	Ascent Thrust Vector Control
BFS	Backup Flight System
BITE	Built-In Test Equipment
CA	California
CAT	Contractor Acceptance Test
cc	Cubic Centimeter
CFF	Carbon Fiber-Filled
CH	Channel
CIL	Critical Item List
CRT	Cathode Ray Tube
CV	Check Valve
CVAS	Configuration Verification Accounting System
DCS	Debris Containment System
DCU	Digital Computer Unit
DFI	Development Flight Instrumentation
DMA	Direct Memory Access
DoD	Department of Defense
DOL	Day of Launch
DR	Discrepancy Report
EAFB	Edwards Air Force Base
ECP	Engineering Change Proposal
EIU	Engine Interface Unit
EPDM	Ethylene Propylene Diene Monomer
EST	Eastern Standard Time
ET	External Tank
ETA	External Tank Attachment

APPENDIX A

LIST OF ACRONYMS (Continued)

F	Fahrenheit
FA	Flight-Critical Aft
FASCO	Flight Acceleration Safety Cutoff System
FCL	Freon Coolant Loop
FCS	Flight Control System
FCV	Flow Control Valve
FEC	Field Engineering Change
FES	Flash Evaporator System
FF1	Flight Forward #1
FIAR	Failure Investigation Action Report
FID	Failure Identification
FIV	Fuel Isolation Valve
FMEA/CIL	Failure Modes and Effects Analysis/Critical Items List
FOS	Factor of Safety
FP	Fuel Pump
fps	Feet Per Second
FRI	Flow Recirculation Inhibitor
FRR	Flight Readiness Review
ft	Feet
GG	Gas Generator
GGVM	Gas Generator Valve Module
GN&C	Guidance, Navigation, and Control
GN_2	Gaseous Nitrogen
GO_2	Gaseous Oxygen
GOX	Gaseous Oxygen
GPC	General Purpose Computer
GSE	Ground Support Equipment
H_2	Hydrogen
H_2O	Water
HDP	Holddown Post
He	Helium
HPFTP	High-Pressure Fuel Turbopump
HPOTP	High-Pressure Oxidizer Turbopump
HPU	Hydraulic Power Unit
HR	Hazard Report
hr	Hour
Hz	Hertz

APPENDIX A

LIST OF ACRONYMS (Continued)

I/O	Input/Output
ICHR	Integrated Cargo Hazard Report
IEA	Integrated Electronic Assembly
IFA	Inflight Anomaly
IMU	Inertial Measurement Unit
in-lb	Inch-Pound
in/sec	Inch Per Second
INTG	Integration
IOM	Input/Output Module
IOP	Input/Output Processor
IRD	Integrated Receiver/Decoder
JSC	Johnson Space Center
Kbit	Kilobit
KSC	Kennedy Space Center
L-2	Launch Minus 2 Day Review
lb	Pound
LCC	Launch Commit Criteria
LDC	Lot Date Code
LH	Left Hand
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LOX	Liquid Oxygen
LPFP	Low-Pressure Fuel Pump
LPFTP	Low-Pressure Fuel Turbopump
LPOTP	Low-Pressure Oxidizer Turbopump
LRU	Line Replaceable Unit
LSFR	Launch Site Flow Review
MCC	Main Combustion Chamber
MDM	Multiplexer-Demultiplexer
ME	Main Engine
MECO	Main Engine Cutoff
MEOP	Maximum Expected Operating Pressure
MET	Mission Elapsed Time
MIA	Multiplexer Interface Adapter
MLP	Mobile Launch Platform
MOD	Mission Operations Director
MPS	Main Propulsion System

APPENDIX A

LIST OF ACRONYMS (Continued)

MRB	Material Review Board
MSA-1	Marshall Sprayable Ablator No. 1
MSE	Mission Safety Evaluation
msec	Millisecond
μsec	Microsecond
MSFC	Marshall Space Flight Center
MTBF	Mean-Time-Between Failure
N_2	Nitrogen
NASA	National Aeronautics and Space Administration
NBR	Nitrite Butadiene Rubber
NDI	Nondestructive Inspection
NPSP	Net Positive Static Pressure
NSI	NASA Standard Initiator
NSRS	NASA Safety Reporting System
O_2	Oxygen
OI	Operational Instrumentation
OMI	Operations and Maintenance Instruction
OMRSD	Operational Maintenance Requirements and Specifications Document
OMS	Orbital Maneuvering System
OPF	Orbiter Processing Facility
OPO	Orbiter Program Office
OPOV	Oxidizer Preburner Oxidizer Valve
ORBI	Orbiter
OV	Orbiter Vehicle
oz	Ounce
P/N	Part Number
PAR	Prelaunch Assessment Review
PASS	Primary Avionics Software System
PDU	Power Drive Unit
PIC	Pyro Initiator Controller
PLS	Primary Landing Site
p.m.	Post Meridiem (Afternoon)
ppm	Parts Per Million
PR	Problem Report
PRACA	Problem Reporting and Corrective Action
PRCB	Program Requirements Control Board
PRSD	Power Reactant Supply and Distribution

APPENDIX A

LIST OF ACRONYMS (Continued)

psi	Pounds Per Square Inch
psi/hr	Pounds Per Square Inch/Hour
psia	Pounds Per Square Inch Absolute
psid	Pounds Per Square Inch Differential
psig	Pounds Per Square Inch Gage
Q	Dynamic Pressure
QA	Quality Assurance
QARSO	Quality Assurance, Reliability, and Safety Office
QD	Quick Disconnect
R	Rankine
RCN	Requirement Change Notice
RCS	Reaction Control System
RF	Radio Frequency
RH	Right-Hand
RI	Rockwell International
RPSF	Rotation, Processing and Surge Facility
RSRM	Redesigned Solid Rocket Motor
RSS	Range Safety System
RTG	Radioisotope Thermoelectric Generator
RTLS	Return to Launch Site
RTV	Room Temperature Vulcanizing
S&A	Safe & Arm
S/N	Serial Number
scim	Standard Cubic Inches Per Minute
sec	Second
SOS	Space Ordnance System
SR&QA	Safety, Reliability and Quality Assurance
SRB	Solid Rocket Booster
SRBTS	Solid Rocket Beacon Tracking System
SRM	Solid Rocket Motor
SRM&QA	Safety, Reliability, Maintainability, and Quality Assurance
SSC	Stennis Space Center
SSME	Space Shuttle Main Engine
SSRP	System Safety Review Panel
SSV	Space Shuttle Vehicle
STS	Space Transportation System
SUBS	Subsequent

APPENDIX A

LIST OF ACRONYMS (Continued)

TAL	Transatlantic Abort Landing
TPS	Thermal Protection System
TVC	Thrust Vector Control
U/N	Unit Number
UCR	Unsatisfactory Condition Report
USBI	United Space Booster, Inc.
V	Volt
WCS	Waste Collection System
WSB	Water Spray Boiler
WSTF	White Sands Test Facility