

STS-3 Orbiter Mission Report

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National Aeronautics and
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STS-3 ORBITER
MISSION REPORT

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TABLE OF CONTENTS

1.0	<u>SUMMARY</u>	1
2.0	<u>ORBITER SUBSYSTEMS</u>	4
2.1	<u>PROPULSION SUBSYSTEMS</u>	4
2.1.1	<u>Main Propulsion Subsystem</u>	4
2.1.2	<u>Orbital Maneuvering Subsystem</u>	8
2.1.3	<u>Reaction Control System</u>	12
2.2	<u>POWER SYSTEMS</u>	12
2.2.1	<u>Auxiliary Power Unit Performance</u>	12
2.2.2	<u>Hydraulic Subsystem</u>	14
2.2.3	<u>Power Reactant Storage and Distribution</u>	15
2.2.4	<u>Power Generation Subsystem</u>	17
2.2.5	<u>Electrical Power Distribution and Control</u>	17
2.3	<u>AVIONICS SUBSYSTEMS</u>	18
2.3.1	<u>Orbital Insertion Operations</u>	18
2.3.2	<u>On-Orbit Flight Control Operations</u>	18
2.3.3	<u>Entry/Landing</u>	19
2.3.4	<u>Communications and Tracking</u>	22
2.3.5	<u>Hardware Performance</u>	26
2.3.6	<u>Data Processing Subsystem</u>	27
2.3.7	<u>Backup Flight System</u>	27
2.3.8	<u>Displays and Controls</u>	27
2.3.9	<u>Instrumentation</u>	28
2.3.10	<u>Systems Management</u>	29
2.3.11	<u>Redundancy Management</u>	29
2.4	<u>ENVIRONMENT CONTROL AND LIFE SUPPORT SYSTEM</u>	29
2.4.1	<u>Active Thermal Control Subsystem</u>	29
2.4.2	<u>Air Revitalization Subsystem</u>	30
2.4.3	<u>Air Revitalization Pressure Control Subsystem</u>	33
2.4.4	<u>Airlock Support Subsystem</u>	34
2.4.5	<u>Water and Waste Management Subsystem</u>	34
2.5	<u>CREW STATION AND EQUIPMENT</u>	34
2.5.1	<u>Cabin Temperature Survey</u>	34
2.5.2	<u>Flight Crew Equipment Cameras</u>	36
2.5.3	<u>Ancillary GFE and Flight Crew Compartment</u>	37
2.6	<u>STRUCTURES</u>	37
2.6.1	<u>Flutter/Buffer</u>	37
2.6.2	<u>Stress Evaluation</u>	37
2.6.3	<u>Entry Flight Loads</u>	37
2.6.4	<u>Window Cavity Conditioning Systems</u>	39
2.7	<u>MECHANICAL SYSTEMS</u>	39
2.7.1	<u>Purge and Vent and Drain Subsystem</u>	39
2.7.2	<u>Orbiter/External Tank Separation and Umbilical Devices</u>	41
2.7.3	<u>Payload Bay Doors and Radiators</u>	42
2.7.4	<u>Manipulator Positioning Mechanism</u>	46
2.7.5	<u>Star Tracker Doors</u>	46
2.7.6	<u>Air Data Probe</u>	46
2.7.7	<u>Landing/Deceleration Subsystem</u>	46
2.7.8	<u>Aerothermal Seals and Thermal Barriers</u>	47
2.7.9	<u>Crew Escape System</u>	47
2.8	<u>THERMAL</u>	50
2.8.1	<u>Thermal Control System</u>	50
2.8.2	<u>Thermal Protection System and Leading Edge Structural Subsystem</u>	57
2.8.3	<u>Aerothermodynamics</u>	74

TABLE OF CONTENTS - Concluded

2.9	<u>ENTRY AERODYNAMICS</u>	74
2.9.1	<u>Performance and Trim</u>	74
2.9.2	<u>Flight Test Requirements Accomplished</u>	80
2.9.3	<u>Stability and Control</u>	80
2.9.4	<u>Mass Properties Comparison Based on Deorbit Maneuver Data</u>	80
2.9.5	<u>Quarter Hertz Oscillation</u>	82
2.9.6	<u>Aerodynamic Coefficient Instrumentation Package</u>	82
2.10	<u>REMOTE MANIPULATOR SYSTEM</u>	82
2.10.1	<u>FTO Accomplishment</u>	82
2.10.2	<u>RMS Problems</u>	84
3.0	<u>ENVIRONMENTS ANALYSIS</u>	84
3.1	<u>ACOUSTIC ENVIRONMENT</u>	84
3.2	<u>VIBRATION ENVIRONMENT</u>	89
4.0	<u>BIOMEDICAL EVALUATION</u>	89
4.1	<u>EVALUATION OF CREW HEALTH</u>	89
4.2	<u>HEALTH STABILIZATION</u>	89
4.3	<u>RADIATION</u>	89
4.4	<u>MOTION SICKNESS PROPHYLAXIS (DSO S343)</u>	91
4.5	<u>TOXICOLOGY</u>	91
4.6	<u>MICROBIOLOGY</u>	91
4.7	<u>BIOINSTRUMENTATION</u>	92
5.0	<u>DESCENT TRAJECTORY</u>	92
6.0	<u>ORBITER ANOMALY SUMMARY</u>	92

1.0 SUMMARY

This STS-3 Orbiter Mission Report presents a summary of the evaluation and analysis of the Orbiter subsystems performance, a discussion of all Orbiter and GFE (government furnished equipment) anomalies, and a discussion of the flight test objectives accomplished. Also included are the problem closeout reports and the corrective action required for STS-4.

The STS-3 mission was originally planned for a landing at Edward AFB, CA., but due to adverse (wet) lakebed runway conditions (fig. 1-1), the primary landing site was moved to White Sands, N.M.

The successful STS-3 mission was launched on March 22, 1982, at 15:59:59.875 G.m.t. from Kennedy Space Center, FL., for a planned duration of 7 days. However, because of adverse weather conditions at the primary landing site, the mission was extended 1 day. The Orbiter was landed satisfactorily on the eighth day at 16:04:46 G.m.t., March 30, 1982. The crew for this third orbital flight test was Col. Jack R. Lousma, Commander, and Col. C. Gordon Fullerton, Pilot.

The major activities of this flight were the thermal testing and (RMS) remote manipulator system testing which also had thermal aspects to it. The major thermal testing consisted of placing the Orbiter in four inertial attitudes for extended periods of time to determine the thermal responses of specific areas. These attitudes were tail-to-Sun in orbital rate, nose-to-Sun twice orbital rate, top-to-Sun solar inertial, and passive thermal control. Temperatures in the tail and nose-to-Sun attitudes were maintained within required ranges with heater duty cycles less than predicted because of slower thermal responses as demonstrated on STS-1 and STS-2. All payload bay door closures during the various attitudes were successful except during the tail-to-Sun attitude. This situation was cleared after reorienting the Orbiter to the top-to-Sun attitude for approximately 15 minutes followed by a short period of passive thermal control.

Approximately 48 hours of RMS testing were completed during STS-3. The major compromise to the RMS tests were caused by the loss of the wrist TV camera. This loss prevented the deployment of the IECM (induced environment contamination monitor) and the PDP (plasma dynamics package) was used in the IECM's stead. The PDP weighs about 500 lb less than the IECM, thus reducing the effectiveness of the dynamic data.

All spacecraft systems operated satisfactorily throughout the STS-3 mission with only minor problems that did not impact the conduct of the mission. The sequence of events for the flight is presented in table 1-I.

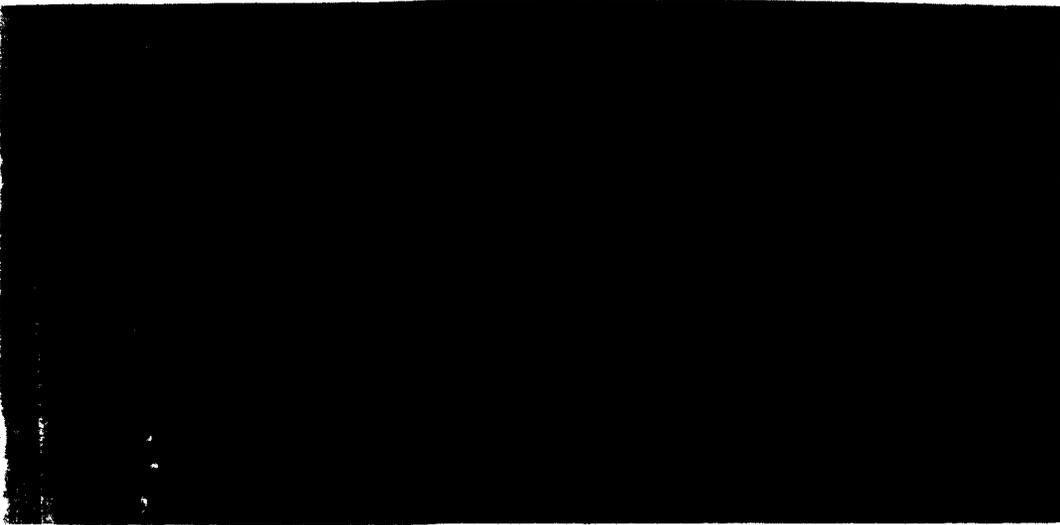
All of the Orbiter services required by the Payload Integration Plan, annexes, etc., were supplied satisfactorily.

Standard units of measurement are used throughout the report. Unless otherwise specified, all given times are referenced to Greenwich mean time (G.m.t.) with lift-off specified as 81:15:59.875 G.m.t. (day:hour:minute:second). All weights are referenced to earth gravity.

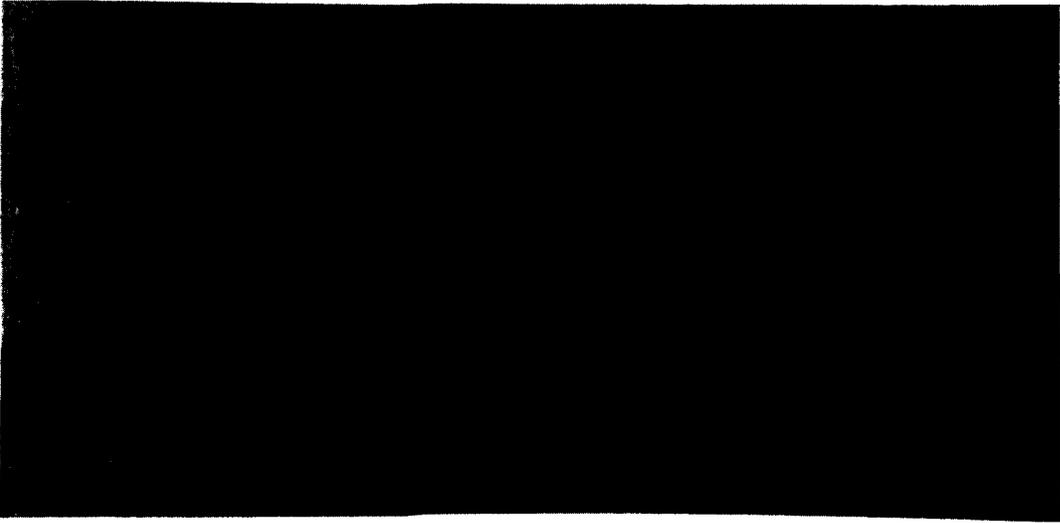
TABLE 1-1.- STS-3 SEQUENCE OF EVENTS

Event	Planned* G.m.t.	Actual G.m.t.
APU activation (1)	81:15:54:57	81:15:55:10
(2)	-----	81:15:55:11
(3)	-----	81:15:55:12
MPS start command (Engine 3)	81:15:59:54	81:15:59:53.2
MPS 90 percent thrust (Engine 1)	81:15:59:57.4	81:15:59:57.2
SRB ignition command from GPC (lift-off)	81:16:00:00	81:15:59:59.9
Main engine throttledown to 68 percent thrust	81:16:00:30.8	81:16:00:31.9
MPS throttleup to 100 percent thrust	81:16:00:52.8	81:16:00:53.5
Maximum dynamic pressure	81:16:01:05	81:16:01:03
SRB separation command	81:16:02:05	81:16:02:07.8
MPS throttledown for 3g acceleration	81:16:04:56	81:16:04:57.8
3g acceleration	81:16:07:36.6	81:16:07:38.1
Main engine cutoff (MECO) command	81:16:08:31.2	81:16:08:33.3
External tank separation command	81:16:08:56	81:16:08:51.3
OMS-1 ignition	81:16:10:34	81:16:10:33.4
OMS-1 cutoff	81:16:12:07	81:16:11:58.6
APU deactivation (1)	81:16:15:00	81:16:15:11
(2)	-----	81:16:15:11
(3)	-----	81:16:08:11
OMS-2 ignition	81:16:40:51	81:16:40:50.4
OMS-2 cutoff	81:16:42:22	81:16:42:18.4
Payload bay doors start opening command	-----	81:17:39:15
Payload bay doors open	-----	81:18:38:32
Payload (OSS-1) activation	-----	81:18:55:59.8
Maneuver to passive thermal control attitude	81:17:03:18	81:19:08:00
Start passive thermal control roll	81:17:03:18	81:19:18:00
Pallet experiment activation	-----	81:20:25:59.8
Maneuver to tail-to-sun attitude	82:02:50:00	82:02:50:00
Start orbital roll rate	82:03:00:00	82:03:00:00
Begin payload bay door exercise	-----	82:23:02:59
Port aft bulkhead latch problem	-----	82:23:14:04
Maneuver to top-to-sun to close door	-----	83:02:01:52
Payload bay doors closed	-----	83:02:48:30
Door exercise completed	-----	83:02:54:15
Maneuver to nose-to-sun attitude	-----	83:13:33
Begin nose-to-sun solar inertial attitude	83:13:50:00	83:13:50
End nose-to-sun attitude	86:01:19	86:01:19
Begin payload bay door exercise	-----	86:20:20
End payload bay door exercise	-----	86:21:42
Maneuver to top-to-sun attitude	-----	86:22:
OMS-3A ignition	-----	86:22:00:00
OMS-3A cutoff	-----	86:22:00:02.2
OMS-3B ignition	-----	86:22:04:04
OMS-3B cutoff	-----	86:22:04:18
APU 3 start for checkout	-----	87:14:42:30
APU 3 shutdown	-----	87:14:53:10
Payload bay doors closed*	-----	89:12:15:15.4
APU 1 activation	88:18:26:00	89:15:08:30
Deorbit burn ignition	88:18:31:47	89:15:13:30
Deorbit burn cutoff	88:18:34:17	89:15:15:57.2
APU 2 and 3 activation	88:18:44:00	89:15:21:44
Entry interface (400,000 ft)	88:18:58:01	89:15:34:44
End blackout	88:19:16:54	89:15:38:09
Terminal area energy management	88:19:25:03	89:15:58:56
Main landing gear contact	88:19:26:00	89:16:04:44.8
Nose landing gear contact	88:19:26:10	89:16:04:59.7
Wheels stop	88:19:26:24	89:16:06:09
APU deactivation completion	88:19:40:24	89:16:24:39

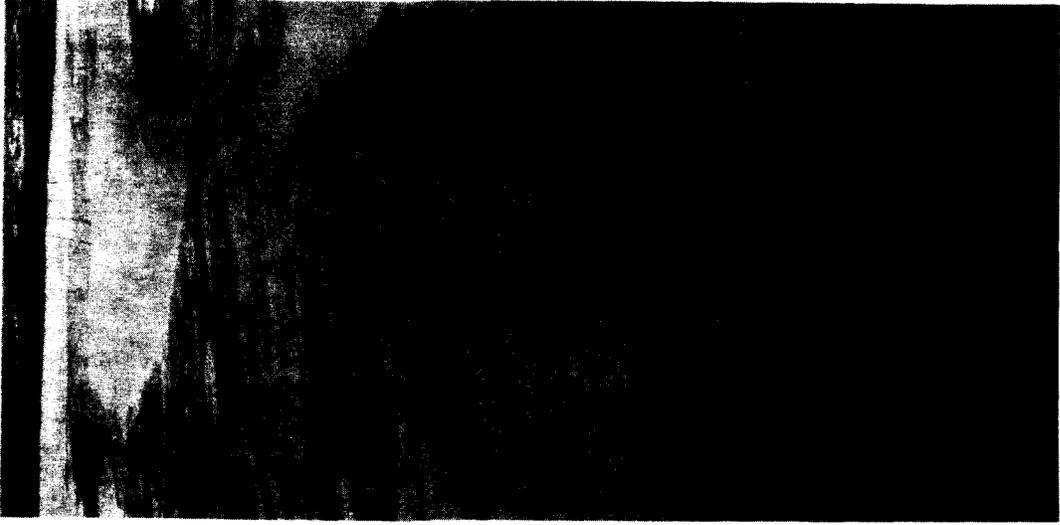
*As a result of the poor weather conditions existing at the White Sands Northrup strip, the decision was made to lengthen the mission approximately one day.



(a) Runway 23



(b) Runway 15



(c) Runway 17

Figure 1-1. - Unacceptable runway conditions (sand and water) at Edwards AFB.

2.0 ORBITER SUBSYSTEMS

All Orbiter systems performed satisfactorily and within the specification and Shuttle Operational Data Book limits during the flight. Several system problems developed during the flight, but acceptable alternate operations permitted the flight to proceed as scheduled. The anomalies conditions that occurred are mentioned under their respective subsystems and a more detailed discussion of each anomaly is contained in section 6.0 of this report.

2.1 PROPULSION SUBSYSTEMS

2.1.1 Main Propulsion Subsystem

The MPS (main propulsion subsystem) performance during ascent was satisfactory. The engine start and cutoff commands occurred as planned, and the subsystem responded to all throttling and gimbaling commands until the early shutdown of an APU (auxiliary power unit) caused main engine 3 to be hydraulically locked up thus resulting in a satisfactory pneumatic shutdown of the engine. The detailed assessment of the overall MPS operation and flight test requirements can be seen in the Marshall Space Flight Center and Integrated flight reports. This section covers overlapping areas that require Orbiter evaluation.

The helium system performed satisfactorily. Approximately 2 1/2 hours before lift-off, the MPS (main propulsion system) helium bottles were loaded from 2000 psia to 4300 psia. The mid-body helium bottles reached a peak temperature of 180° F, due to compressive heating, compared to the design limit of 200° F. The aft-body bottles had a lower peak temperature, due to exposure to the cryogenic propellant lines. The loaded helium mass was 211.2 lbm compared with 212.6 for STS-2, 208.9 for STS-1, and 214.9 predicted.

The oxygen concentration before launch was essentially 0 ppm except for a few short transient spikes that were below 100 ppm. A large oxygen concentration transient occurred about 45 minutes prior to launch when the cabin was vented. The actual gaseous hydrogen concentration in the aft fuselage was below 400 ppm and well within the 500 ppm redline. The concentrations during ascent in the five good sample bottles were similar to the STS-1 concentrations and well below the flammability curve, indicating that a hazardous concentration did not exist.

For the STS-3 launch, all MPS launch commit criteria were within limits.

Unlike STS-1 and STS-2, the liquid oxygen 100-percent 1 liquid-level sensor operated properly during the loading and replenish periods and was used as the prime replenish control sensor. The sensor indicated approximately 60-percent wet when it was monitored for compliance at loading termination - well within the 15 to 75 percent limits. The engine inlet pre-start propellant conditions were within the SODB limits as shown in figure 2-1.

The main propulsion system performed normally until a mission elapsed time of 489 seconds, at which time main engine 3 went into a hydraulic-lockup condition because APU (auxiliary power unit) 3 was shutdown. This engine remained at an 82-percent power level until guidance cutoff at which time the engine responded as expected. No performance drift or abnormal operating conditions were noted during the hydraulic lockup and pneumatic shutdown of main engine 3. The engine inlet conditions during main stage operations were within the requirements established in program documentation as shown in figure 2-2. Guidance-commanded MECO occurred at 081:16:08:33.3 G.m.t. The total firing time on each of the main engines was approximately 8 minutes, 39.8 seconds, which includes approximately 7 seconds for the start transient and burn time prior to the SRB (solid rocket booster) ignition command. Ignition of the main engines commenced at 081:15:59:53.2 G.m.t. with main engine 3 starting first as planned.

STS-3 data: Engine 1 ○ Engine 2 △ Engine 3 □

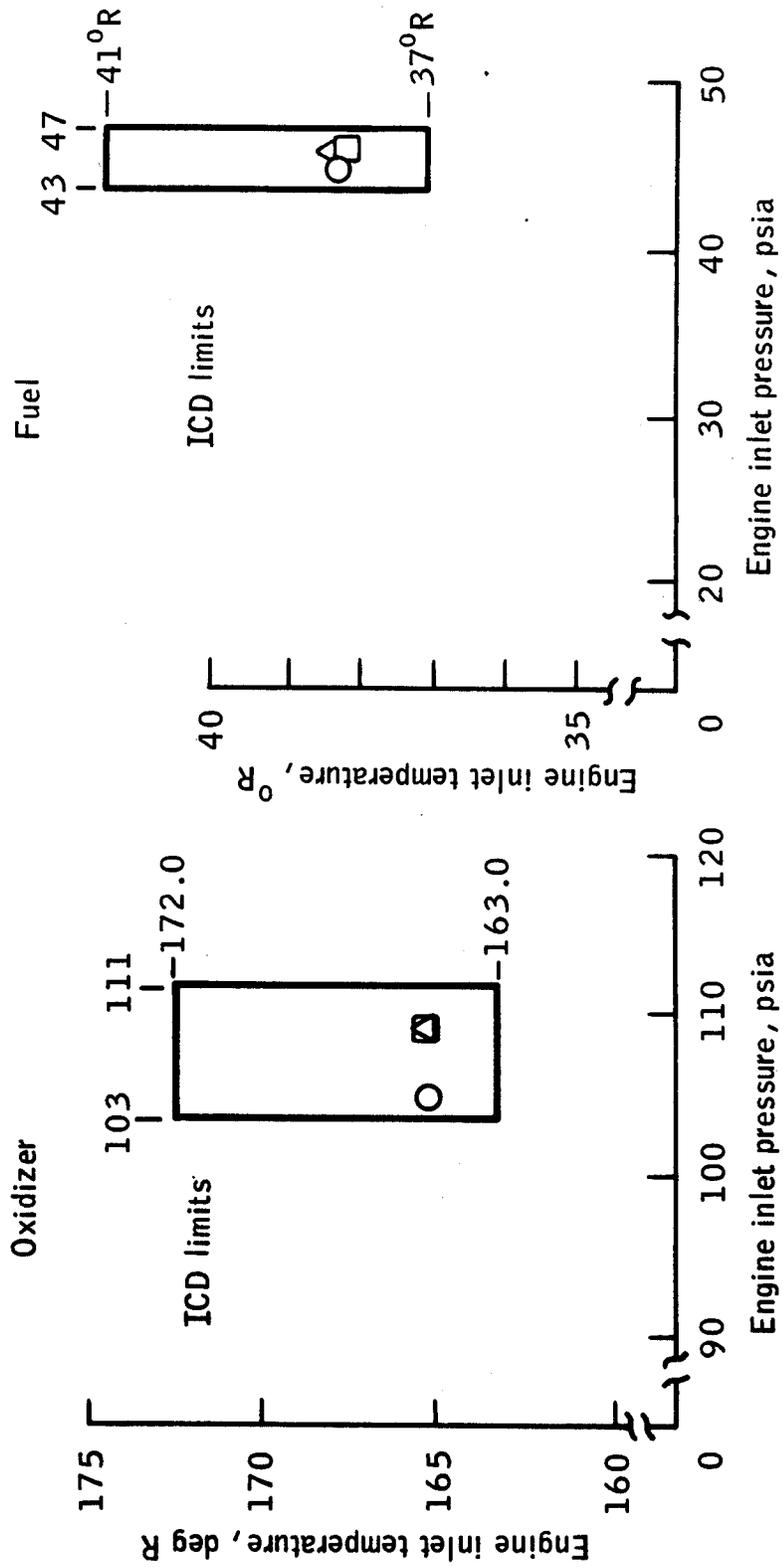


Figure 2-1.- Prestart propellant conditions.

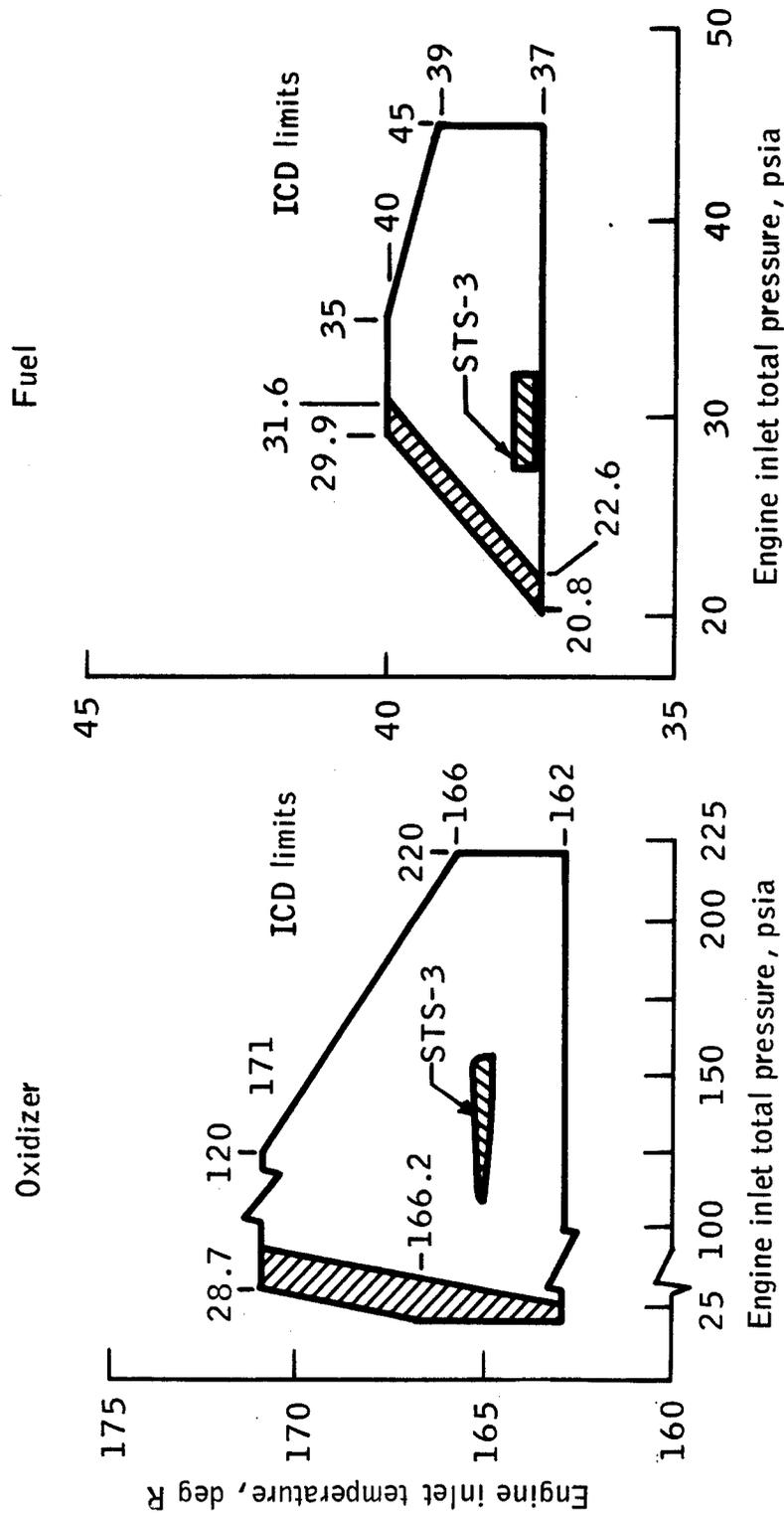


Figure 2-2.- Engine propellant inlet conditions (mainstage operation).

None of the Orbiter MPS valves were slammed during the flight. However, some peculiarities were noted. At the end of the liquid hydrogen 30-second manifold relief, which is performed using the RTLS ball valves, the inboard ball valve was slow closing. The valve required 2.13 ± 0.1 , -1.0 seconds to close, but the requirement is 1.0-second maximum. At the beginning of the first vacuum inerting procedure, the outboard liquid hydrogen fill and drain valve was fast in opening in 4.0 seconds versus the requirement of 5.0 seconds minimum.

The liquid oxygen overboard bleed valve has two switches (A and B) that indicated closed. This valve was closed 8 seconds before T-0. If the valve does not close, it would be a launch delay. Switch B indicated the valve closed in 0.5 ± 0.1 , -1.0 second, which was within the specification tolerance of 0.5-second maximum, and the launch went on without any delay. Switch B took 4.6 seconds to indicate closed, which is well above the requirement. Further investigation into switch B is required to determine the cause of the delayed switching.

Propellant dump from the Orbiter feedlines was initiated 2 minutes after MECO and continued for 3 minutes and 3 seconds. Within a few minutes after the dump was completed, the Orbiter feed system was vacuum inerted by opening the fill and drain valves for approximately 16 minutes and 50 seconds. Dumping of the liquid oxygen feed system was conducted through main engines 1 and 2 only since the shutdown of APU 3 resulted in the loss of the capability to open the main oxidizer valve on main engine 3 to dump through that engine. The dump appeared normal, but pressures after the dump were higher than normal until vacuum inerting. After the first vacuum inerting, particles of "snow" were observed by the crew coming from the aft end of the Orbiter. These particles may be oxygen coming from the high-pressure oxygen turbopump primary-seal drain. Pressures in the feed system remained essentially "zero" following the vacuum inerting, thus verifying that the "snow" did not cause a problem to the MPS. In addition, during entry the feedline repressurization inerts the system with helium.

A second vacuum inerting was performed at which time the hydrogen feed system pressures decreased indicating that some quantity of hydrogen had remained in the hydrogen feed system after dumping and the first vacuum inerting.

The reconfiguration of the helium system prior to entry was performed satisfactorily for this flight, with the change in prelaunch procedures to delete the MDM (multiplexer/demultiplexer) open commands that produced a problem on STS-2. Following rollout, however, the helium usage was higher than predicted. It has been determined that this excessive usage was due to a failed-open POGO (longitudinal oscillations) return line check valve at the main engine 3 interface. (Section 7.0, flight test problem report 39).

The leakage problem with the 8-inch liquid hydrogen disconnect that occurred on the STS-1 and STS-2 propellant loadings has been resolved. After STS-1, the preload on the interface seal was increased. This fix was accomplished prior to the first STS-2 tanking. However, the disconnect still leaked. Before the next STS-2 tanking the hazardous gas sensor was replaced with two recalibrated sensors. Also, the gaseous helium purge was increased from 22.3 scfm to 73.5 scfm in the 8-inch disconnect umbilical purge box. This brought the gaseous hydrogen concentration down to less than 1 percent for the next STS-2 tanking. Upon landing, the T-0 8-inch disconnect inserts were checked and were found to be loose. A new insert installation and torquing procedure that was used prior to STS-3. During both the tanking test and propellant loading for STS-3, the hazardous gas concentration was essentially zero.

2.1.2 Orbital Maneuvering Subsystem

The OMS (orbital maneuvering subsystem) was used for 5 firings during the STS-3 mission, and OMS propellants were supplied to the RCS (reaction control subsystem) from both the left and right pods. Data from all firings indicate normal system performance. The only problems encountered during the mission were known preflight (either carried over from STS-2 or from pre-mission ground operations). These problems were a recurrence of the erroneous crossfeed valve position indications and of erroneous readings from the gaging system.

The OMS cold-engine restart DTO (detailed test objective) was performed without incident, however a recorder failure resulted in the loss of the data necessary for evaluation of the DTO.

A summary of the STS-3 mission events and propellant quantities is shown in table 2-I. A comparison between planned and actual ignition times, firing times, and differential velocity is shown for each OMS maneuver in table 2-II. The OMS-1, OMS-2, and deorbit firings were in the normal feed mode using 2 engines. All data reviewed from these firings indicate normal system performance. The OMS 3A and 3B firings were performed with the left pod to satisfy the OMS cold restart FTO (flight test objective) (344-01). These firings were inadvertently performed in an ullage blowdown mode; this caused no system problems, but did lengthen the 3B firing sufficiently for the gaging system to update (> 14 seconds).

The OMS-to-RCS interconnect was used a total of 9 times during the mission; the propellant supplied to the RCS was calculated at 1188 lbm from the left pod and 1117 lbm from the right pod.

The left pod crossfeed valve position indicator problem seen in STS-2 recurred at 88:15:33 G.m.t. and again the crew placed the switch in GPC to remove power to the valve. The OMS propellant quantity gauging system experienced a number of anomalies problems, and all of the significant inflight anomalies are associated with previously defined problems that have subsequently been resolved with design changes or proposed design changes that have not been incorporated into the Orbiter 102 hardware.

Table 2-III shows the reconstructed engine performance data based on the observed propellant tank and engine inlet pressures. Engine performance was as expected, and engine chamber pressure and regenerative cooling jacket outlet temperature compare well with observed values, within the accuracy of the instrumentation. Engine valve timing, and start and shut-down transients were normal. The engine pneumatic systems functioned normally. The reduced chamber pressure of the left engine during the OMS-3A and OMS-3B firings reflects the fact that these were performed in the blowdown mode. This mode of operation resulted from a procedural error which prevented the helium isolation valves from opening at ignition for both firings.

The OMS-3A and OMS-3B firings were performed to fulfill the requirements of FTO 344-1 (OMS Engine Cold Restart). The purpose of this FTO is to demonstrate the capability of the engine to perform a restart under worst case conditions (minimum off-time between firings and cold hardware temperatures). The real time pre- and post-firing data were normal, and the data reviewed later looked completely normal. It was learned postflight, however, that the FM wideband recorder failed several hours before the firings were performed. Without the engine injector accelerometer data (which is on the wideband recorder), no accurate assessment of the engine transient response to the restart test can be made. An effort will be made to reschedule this FTO on one of the remaining flights with DFI (development flight instrumentation).

TABLE 2-I.- STS-3 OMS EVENTS

Event	Approximate G.m.t.	Actual G.m.t.	Propellant quantity, percent			
			Left hand oxidizer	Left hand fuel	Right hand oxidizer	Right hand fuel
OMS-1 ignition		81:16:10:33.35				
OMS-1 shutdown		81:16:11:58.55	56.0	55.4	55.4	55.4
OMS-2 ignition		81:16:40:50.35	42.9	42.4	42.2	42.4
OMS-2 shutdown		81:16:42:18.35	-3.3	-3.3		
Interconnect to LH OMS	81:14:27		39.6	39.1		
Interconnect return	83:02:58					
LH repress (A Leg)	83:03:03					
Interconnect LH OMS	83:15:45		-0.5	-0.5		
Interconnect return	84:03:18		39.1	38.6		
Interconnect to LH OMS	84:15:36		-2.0	-2.0		
Interconnect return	85:03:11		37.1	36.6		
LH repress (A Leg)	85:03:45					
Interconnect RH OMS	85:14:12				-5.1	-5.1
RH repress (A Leg)	85:14:40					
Interconnect return	85:22:58				37.1	37.3
Interconnect LH OMS	85:22:58		-0.8	-0.8		
Interconnect return	86:02:55		36.3	35.8		
Interconnect RH OMS	86:12:30				-3.5	-3.5
OMS-3A ignition		86:21:59:59.99				
OMS-3A shutdown		86:22:00:02.19				
OMS-3B ignition		86:22:04:03.99				
OMS-3B shutdown		86:22:04:17.99	34.1	33.6	33.6	33.8
Interconnect return	87:01:08					
LH repress (B Leg)	87:01:09					
RH repress (B Leg)	87:01:10					
Interconnect LH OMS	88:18:09		-0.5	-0.5		
Interconnect return	88:23:00		33.6	33.1		
Interconnect LH OMS	89:09:03		-2.1	-2.1		
Interconnect return	89:14:27		31.5	31.0		
Deorbit firing ignition	89:15:13:30	89:15:13:29.99				
Deorbit firing shutdown	89:15:15:57.5	89:15:15:57.19	9.6	9.4	11.5	12.2
Total interconnect propellant, lbs			1188			1117

TABLE 2-II.- STS-3 OMS FIRING DATA

Event	Planned			Actual		
	Ignition time, mission elapsed time	Burn time, seconds	ΔV , ft/sec	Ignition time, mission elapsed time	Burn time, seconds	ΔV , ft/sec
OMS-1	00:00:10:34	89.3	147.9	00:00:10:33.4	85.2	152.2
OMS-2	00:00:40:51	88.2	151.7	00:00:40:50.4	88.0	151.9
OMS-3A	04:07:35:00	2.0	1.8	05:06:00:00	2.2	1.8
OMS-3B	04:07:51:30		11.2	05:06:04:04	14.0	12.0
Deorbit	07:02:15:00	158.5	287.0	07:23:13:30	147.2	270.3

TABLE 2-III.- ENGINE PERFORMANCE SUMMARY

Left OMS engine							
	Reconstructed					Observed	
	Isp, seconds	Mixture ratio	W _{TOT} , lbm/sec	Chamber pressure, percent	Regenerative cooler temperature, °F	Chamber pressure, percent	Regenerative cooler temperature, °F
OMS-1	314.7	1.663	19.13	104.5	224	103.0	232
OMS-2	314.7	1.664	19.13	104.5	224	103.0	232
OMS-3A*	314.5	1.654	18.98	103.7	224	102.7	**
OMS-3B*	314.5	1.673	18.42	100.6	227	99.5	228
Deorbit	314.7	1.664	19.13	104.6	224	104.3	224

Right OMS engine							
	Reconstructed					Observed	
	Isp, seconds	Mixture ratio	W _{TOT} , lbm/sec	Chamber pressure, percent	Regenerative cooler temperature, °F	Chamber pressure, percent	Regenerative cooler temperature, °F
OMS-1	314.5	1.680	19.24	104.9	225	103.7	233
OMS-2	314.5	1.680	19.25	104.9	225	102.1	234
Deorbit	314.5	1.681	19.24	104.9	225	104.3	224

*OMS-3A and OMS-3B firings were performed in blowdown mode. Data taken just prior to engine shutdown.

**Did not reach steady-state.

2.1.3 Reaction Control System

The RCS (reaction control system) performance was normal. Early in the flight, a 60° F difference between the fuel and the oxidizer leak detector on F5L was noted (Section 7.0, flight test problem report 6).

Propellant consumption from the RCS differed from the predicted usage due to changes in the timeline, and the additional day in-orbit. A tabulation of propellant used from the RCS as a function of mission phase is as follows:

QUANTITY OF RCS PROPELLANT USED

Mission phase	Left RCS, lb	Right RCS, lb	Forward RCS, lb
	Actual	Actual	Actual
Ascent	219	230	134
On-orbit	691	724	1653
De-orbit to entry interface	25	38	175
Entry interface to landing	633	640	-

In addition to the propellant used from the RCS tanks, the aft systems also consumed 1188 lbs of propellant from the left OMS tank, and 1117 lbs from the right OMS tank.

The thermal environment for STS-3 was more severe than the previous flights due to the thermal attitudes and the thermal FTO's performed on the RCS. Two thermal FTO's, a 100-second firing on primary thruster L2U, and a vernier thruster pulse test on L5D, were successfully performed.

During entry, at about 5 minutes after entry interface, thrusters L2U and R4U were deselected because their fuel leak detectors had fallen below the 30° F leak detector limits. At this same period of time on the two previous flights, significant cooling had been noted on the fuel leak detectors (down to 37° F on STS-1, and 33° F on STS-2). This was believed to be caused by the fuel dribble volume pooling in the injector because of the effects of gravity, which would cause more cooling of the leak detector. Because the propellant was colder than on the previous flights for entry, there was a greater chance of a false leak indication. For this reason, the crew was instructed to reselect any upfiring engines which were declared leaking by redundancy management. The leak detection limit on fuel is being changed from 30° F to 20° F for STS-5, and subsequent flights.

2.2 POWER SUBSYSTEMS

2.2.1 Auxiliary Power Unit Performance

Overall APU (auxiliary power unit) performance met all hydraulic system demands throughout STS-3. System performance as well as the problems encountered during the mission are discussed in the following paragraphs.

The APU's were started 5 minutes before launch and 2 of the 3 APU's were shut down after MPS dump. APU 3 was shut down earlier after the single engine press-to-MECO trajectory position was passed (approximately 8 minutes before planned shutdown time). The on-orbit flight control checks were performed with APU 3. For entry, APU 1 was started at 5 minutes

before ignition for the deorbit maneuver and APU's 2 and 3 were started at entry interface minus 13 minutes. Total run times are shown in the following table.

STS-3 TOTAL RUN TIME

Phase	APU run time, hr:min:sec		
	1	2	3
Ascent	00:19:50	00:19:51	00:12:51
On-orbit checkout			00:10:42
Entry	1:15:59	1:01:41	1:01:42
Total run time	<u>1:35:49</u>	<u>1:21:32</u>	<u>1:25:15</u>

Fuel consumption for the mission was 796 lb, divided as follows - APU no. 1 - 270 lb; APU no. 2 - 260 lb; and APU no. 3 - 226 lb. The fuel consumption was determined from the pressure-volume gaging which is inaccurate. Actual usage will not be known until servicing for STS-4 when the residuals are offloaded.

The performance of all 3 APU's during the various mission phases was normal with the exception of the APU 3 lubrication system temperatures, a leak in the APU 2 seal cavity drain line, and a slight leak in the APU 3 fuel system. In addition, a bubble was evident in the APU 2 chamber pressure trace for the first time. APU 1 and 3 have experienced bubbles on previous flights, however in all instances, the bubbles have in no way impacted the operation of the system.

During ascent, the APU 3 lubrication system temperatures increased to 345° F as a result of a malfunctioning water spray boiler. As a result of the overtemperature, the flight rules were modified to allow a transient oil temperature up to 400° F under certain conditions. (Section 7.0, flight test problem report 4).

The fuel system pressure decay was caused by a leaky gaseous nitrogen quick disconnect on system 3. The leak was discovered after servicing for STS-3 when the cap had been loose. After the cap was tightened, no further decay occurred before launch. During the on-orbit period, the pressure decayed 12 psi. (Section 7.0, flight test problem report 24).

APU 2 seal cavity drain slowly leaked down to 0 psia over a 24-hour period. During entry, the seal cavity pressure closely approximated the ambient pressure. Postflight checkout found the drain relief valve to be leaking grossly. (Section 7.0, flight test problem report 5).

The measured seal cavity leakage quantities drained from the catch bottles during postflight operations were APU 1 - 50 cc, and APU 3 - 28 cc. There was no liquid in the APU 2 catch bottle because of the leak. The analysis of the effluent indicated that over half of the liquid was alcohol left over from the seal cavity flush prior to STS-3.

The primary water system isolation valve was on for 6 hours after APU shutdown for the on-orbit check. The temperature of the valve increased from 60° F to 123° F before the valve was closed. The temperature was not detrimental to the valve, but in the future, care should be taken to close the valve after 2 1/2 hours.

The thermal control system heaters for the APU fuel, lubrication oil, and water systems maintained temperatures within critical limits throughout the on-orbit APU non-operational periods. Both systems A and B were exercised.

2.2.2 Hydraulic Subsystem

The performance of the hydraulic system was satisfactory except for the malfunction of the system 3 WSB (water spray boiler) during ascent. The WSB failed to cool the APU lubrication oil and, therefore, the APU had to be shut down prior to main engine cutoff.

On-orbit thermal conditioning was conducted satisfactorily maintaining the hydraulic system temperatures greater than -10° F. All test objectives were achieved.

During servicing, the circulation pumps were operated to fill voids in the landing gear systems. No voids were indicated during the operation. Also, WSB 3 was initially serviced to 2 lb preload, but the preload was reduced to approximately 0.151 lb prior to launch.

During the ascent-to-orbit operation, the crew reported that APU 3 lubrication oil temperature was rising and reported a caution and warning alarm at 4 minutes 30 seconds into the flight. WSB 3 was switched to controller B 15 seconds later, and the lubrication oil temperature continued to rise. At 8 minutes 2 seconds, APU 3 was shut down and APU's 1 and 2 were switched to high speed. At that time, the APU 3 lubrication oil temperature was approximately 345° F and the bearing temperature was slightly higher. (The system 3 WSB also froze during ascent on STS-1 and STS-2.) (Section 7.0, flight test problem report 4). Operation during on-orbit checkout, entry and landing was normal.

At MECO main engines 1 and 2 were shut down normally with the hydraulic systems and main engine 3 was shut down with the backup pneumatic system. Post-insertion, the rudder speed brake return line temperature measurement failed off-scale high at $+305^{\circ}$ F. (Section 7.0, flight test problem report 10).

During the on-orbit phase of the mission, the circulation pumps were used to thermally condition the systems. There were fewer warm-up cycles than anticipated with 34, 24 and 30 cycles for systems 1, 2 and 3, respectively, probably because of temperature stratification of the hydraulic oil. This stratification is attributed to the slow cooldown rates. The predicted and actual cooldown rates are summarized in the following table

SYSTEM COOLDOWN RATES

	<u>Passive thermal control and tail-to-Sun attitudes</u>		<u>Nose-to-Sun attitude</u>	
	<u>Predicted, deg/hr</u>	<u>Actual, deg/hr</u>	<u>Predicted, deg/hr</u>	<u>Actual, deg/hr</u>
Left outboard elevon	15	4.5	22	8
Left inboard elevon	13	4.4	22	12
Right inboard elevon	15	3	22	6.7
Right outboard elevon	15	4.4	22	7
Rudder speed brake	18	5.5	23	8.25
Body flap	5.8	1.8	45	22.3
Brakes	8.5	2	-	-
Mid body	3.6	2	-	-

The predicted "on" time for the circulation pumps was 3 to 8 minutes, and the actual operating time was close to this prediction; however the predicted "off" time was 1 hour and, in actuality, the "off" time was about 2.5 hours.

The lower temperature limit established for the mid-body landing gear lines was -50° F. This limit was approached during the flight with readings of -48° F. During the mission, this limit was lowered to -65° F for the nose-to-Sun attitude since there would be no flow with the landing gear isolation valve closed. Additional data will be obtained to determine if this measurement is in agreement with its structural driver.

During on-orbit circulation pump operation, circulation pump 2 exhibited a low pressure (198 psia) when the pump was turned on and a higher pressure (225 psia) after the landing gear isolation valve was opened. It is suspected that the lower than expected pressure was caused by the elevon actuators moving, but additional data will be required for a complete evaluation. Circulation pump 2 also exhibited a high pressure (431 psia) for the first duty cycle in the nose-to-Sun attitude and 330-to-360 psia for all other cycles. All thermal bypass valves were closed for the high pressure cycle and open for all other cycles.

De-orbit preparations were initiated 22 minutes before the planned deorbit maneuver on orbit 115. At that time, circulation pumps 2 and 3 were activated to close the landing gear isolation valves. After the isolation valves were closed, all three circulation pumps were turned off, thus terminating the on-orbit automatic thermal control. On orbit 116, the decision was made to scrub the landing due to weather conditions at the Northrup strip.

De-orbit preparations were again initiated at deorbit maneuver ignition minus 22 minutes on orbit 128 and the APU prestart procedures repeated. APU 1 was started at deorbit maneuver ignition minus 5 minutes and the system remained in the depressurized mode. At entry interface minus 13 minutes, APU's 2 and 3 were started and all three hydraulic pumps were placed in the normal or high pressure mode.

The landing gear isolation valves were opened in a system 3, 2 and 1 sequence via software. System 3 valve was opened when the Orbiter was at a velocity of 19,000 ft/sec followed by the system 2 valve opening 5 minutes later. The system 1 valve was opened approximately 13 minutes after system 3 valve was opened. When the system 1 isolation valve was opened, the system 1 reservoir showed a fluid volume decrease of 2.4 percent.

The landing gear was deployed at a velocity of 270 knots and down and locked approximately 2 seconds before touchdown. Gear deployment resulted in a decrease of 13.6 percent in the system 1 reservoir.

The hydraulic load test was performed after rollout with systems 2 and 3 at normal pressure while system 1 was depressurized and then repeated with system 3 depressurized and systems 1 and 2 at normal pressure. System 3 was repressurized and the SSME's were moved to the "rain drain" position. The reservoir volume changes were -2.4 percent for systems 1 and 3 and -3.2 percent for system 2.

2.2.3 Power Reactant Storage and Distribution

During the time between loading and launch, remote power controller 2 in the hydrogen/oxygen control box 1 failed to the "on" position. This caused the hydrogen tank 1 heater B to remain on continuously while the control switch was in the "auto" mode. After an analysis was made and a test run to verify the analysis was completed, the decision was made to fly with this failure. (Section 7.0, flight test problem report 1).

At lift-off plus 35 seconds, heater A for oxygen tank 3 tripped. The CLD (current level detector) safes the heater (i.e., inhibit) when it trips. The CLD was reset and functioned properly for the remainder of the flight. (Section 7.0, flight test problem report 3).

The heater management plan that was used for the flight is outlined in table 2-IV. In general, the plan was not modified except during the mission extension phases of the flight. These modifications were made primarily to enhance the system capability for further mission extensions.

The quantities of hydrogen and oxygen used during the mission were 2001.7 lb of oxygen and 249.41 lb of hydrogen. This equates to an energy production of 94,010.5 Ahr with 21.83 lb of oxygen being supplied to the ECLSS.

TABLE 2-IV.- HEATER MANAGEMENT PLAN

Mission elapsed time, hr:min:sec	Oxygen tanks				Hydrogen tanks			
	1	2	3	4	1	2	3	4
Launch								
0:02:30	Auto	Auto			Auto	Auto	.	
4:00:00	Off	Off	Auto	Auto	Off	Off	Auto	Auto
72:00:00	Auto	Auto	Off	Off	Auto	Auto	Off	Off
128:40:00	Off	Off		Auto	Off	Off		Auto
151:00:00			1/2Auto	Off			Auto	Off
177:00:00	Auto	Auto	Off		Auto	Auto	Off	
185:00:00	Off	Off	1/2Auto	1/2Auto	Off	Off	Auto	Auto
190:00:00			Off	Off			Off	Off

At 39 hours into the mission, the quantity gage in hydrogen tank 2 went to an "off-scale-high" reading. (Section 7.0, flight test problem report 10). The measurement returned to a normal value just before landing.

During the mission, DTO (detailed test objective) 343 part 01 and 02 were completed. This DTO, cryogenic tank heat leak demonstration was conducted in two parts, i.e., one for the cold case and one for the hot case. During the test periods, the heaters in the tanks of interest were inhibited (heaters placed in the "off" position) and the quantity decrease was noted. Since the only energy entering the tank for these periods was heat leak, the quantity decrease could be related to a flow rate which could be related to a tank heat leak. Also, since there were extended periods of heater-off activity for all the tanks, a great deal of supporting data were also obtained.

2.2.4 Power Generation Subsystem

The fuel cell powerplants operated normally throughout the entire mission. The actual Orbiter power levels during ascent, orbit, and entry were lower than predicted.

Fuel cells 1, 2, and 3 were activated at 81:08:11, 81:08:33, and 81:08:53 G.m.t., respectively. The fuel cells were operated for approximately 224 hours. The mission average power level was 14.8 kW, giving a total energy output of 2851 kWh. All three fuel cells operated above their predicted performance curve during the entire mission. The fuel cells were shutdown at 90:17:10, 90:16:23, and 90:17:05 G.m.t. The total number of operating hours on fuel cell 1 is 301 hours with fuel cell 2 and 3 having 417 and 355 hours, respectively.

The automatic-purge portion of FTR V45VV011 was successfully completed during STS-3. The manual portion was completed during STS-1, but because of the malfunctioning flow meters, the automatic portion could not be successfully demonstrated at that time. For STS-3, the flow rate limits were set off-scale high in the GPC (general purpose computer), and this changing of the flow rate limits allowed completion of the automatic purge portion of this FTR.

2.2.5 Electrical Power Distribution and Control

The EPDC (electrical power distribution and control) performance during STS-3 was normal. All of the Orbiter bus voltages were well within design limits throughout all mission phases. During prelaunch operations, there were adequate voltage margins for all of the launch commit criteria redlines.

The ac power system supported all of its power requirements adequately throughout the mission.

Orbiter power services to the OSS-1 payload were as planned and met all payload requirements.

The EVCON (events control) subsystem performance was exceptional. Lift-off and ascent functions were completed in a comparable timeframe with STS-1 and STS-2. There were no inadvertent pyrotechnic firings or failure of redundant circuits to fire. During this flight, a deviation from normal procedures required the MEC's (master events controllers) to be powered "on" and "off" in flight day 8, then "on" again on the final day. An anticipated phenomenon known as "MIA burping" occurred, causing an I/O (input/output) error associated with MDM FF4 (Section 7.0, flight test problem report 37).

Prior to launch, the hydrogen tank 1 heater "B" was failed "on" in the automatic mode (Section 7.0, flight test problem report 1). The second problem was the mid-body motor control assembly 3 operational status number 4 was at "0" and should have been "1" following the pre-deorbit vent door closure sequence (Section 7.0, flight test problem report 34). Further, the measurement entitled (PCA - MPS LOX pre-valve 3 close RPC C on) came on, but the valve command measurement stayed off (Section 7.0, flight test problem report 38).

2.3 AVIONICS SUBSYSTEMS

2.3.1 Orbital Insertion Operations

The transition DAP (digital autopilot) performance was normal. The RCS/structural interaction that occurred during STS-2 during mated-coast SSME-slewing did not recur. Recurrence was averted because the Orbiter pitch rate was near zero at the beginning of SSME slewing. That is, the Orbiter "flexural" rate due to slewing (approximately 0.25 deg/sec amplitude) plus the rigid body rate (approximately 0) did not exceed the rate limit (0.5 deg/sec) and no firings occurred. The rigid-body rate was near zero because the early APU shutdown caused high MECO tail-off rates (greater than the 0.5 deg/sec rate limit) which initiated rate damping prior to slewing. For STS-2, the magnitude of the tail-off rate was about 0.35 deg/sec. For STS-4, there is still the potential for interaction, but a software change to the slewing sequence on STS-5 and subsequent should preclude the interaction after STS-4.

2.3.2 On-Orbit Flight Control Operations

Overall performance of the primary RCS (reaction control system) was normal. The only unexpected events were three yaw pulses inadvertently commanded via the aft RHC (rotational hand controller) during the RMS/PRCS interaction tests (FTO 352-05). Two of these occurred during part I of the unloaded arm test and the third one occurred during the loaded arm test. In each case, the yaw pulse was inadvertently commanded during the RHC + roll deflection (vehicle - roll). Each of these firings imparted a sizeable vehicle roll rate ($\approx + 0.4$ deg/sec) and yaw rate (≈ -1 deg/sec) to be subsequently damped by the vernier RCS. This contributed to the excessive vernier RCS activity discussed in a following paragraph.

Performance was as expected, with the exception of the minimum deadband test, FTO 372-01, discussed in a following paragraph. Of particular significance from the vernier RCS engine lifetime viewpoint, excessive pulsing and excessive on-times occurred during rotational maneuvers at higher rates (0.5 deg/sec) than should be performed with the vernier system. These maneuvers took place during the first part of the RMS/primary RCS unloaded arm test on flight day 2. The vernier RCS DAP performs rotational maneuvers at rates up to 0.2 deg/sec efficiently, however performance degrades rapidly as the maneuver rate increases because of overshoots at the end of each maneuver. Once an overshoot has occurred, the DAP must perform another maneuver back to the desired attitude. At a maneuver rate of 0.5 deg/sec, several overshoots may occur before the system converges on the desired attitude. Each overshoot results in wasted propellant and additional engine pulsing. This, combined with the inadvertent primary RCS yaw firings during the unloaded arm test resulted in an excessive usage of 200 pounds of propellant. Further, continuous engine firings greater than 60 seconds and upwards of 400 pulses per engine were observed from the flight data.

Flight test objective 372-01 (Vernier Minimum Deadband Test) was performed as planned. Performance during the 0.033-degree deadband portion of the test was normal. During the 0.01-degree deadband part of the test, however, the attitude errors in each axis reached

values almost twice the deadband value (i.e., ± 0.02 degree) on several occasions. Analysis to determine the source of the problem is continuing.

Flight test objective 372-02 (Passive Gravity Gradient Test) was completed as planned, but the results were inconclusive because of the limited test time interval. Analysis of the data indicates that the initial roll angle (determined from preflight analysis) was in error by about 4 degrees. The test is scheduled to be performed again on STS-4.

Flight test objective 372-03 (Vernier RCS Plume Impingement) was performed as planned and ACIP (aerodynamic coefficient identification package) data were obtained. Preliminary analysis of these data generally confirms preflight predictions of significantly reduced thrust of the vernier engines L5D and R5D caused by plume impingement on Orbiter structure beneath the engines.

2.3.3 Entry/Landing

Entry and TAEM (terminal area energy management) performance was generally as predicted. However, both surface and high altitude winds on the planned landing day exceeded verification testing levels (Section 7.0, flight test problem report 41). The actual "wave off" was made based only on surface winds. However, between 30K and 25K feet altitude, winds exceeded those to which the system had been verified. Simulation testing on "wave off" day developed confidence that the right-turn option of TAEM was much more benign in the presence of high winds aloft. A criteria was quickly developed on which to base a "wave off" due to high altitude winds. These data are being refined for STS-4 and a placard is to be baselined for STS-4.

During nose gear slapdown, the vehicle pitched up to around 6 degrees after having been down to -3 degrees pitch. (Section 7.0, flight test problem report 41). This occurred because the pilot was preventing premature nosewheel contact. Subsequent analysis has shown the vehicle responded essentially as analysis predicted and as seen on the Ames simulator, both prior to and after the mission, in which nearly identical responses occurred.

The autoland system was flight tested starting at the normal engage altitude of 10,000 ft to approximately 140 ft when downmoding occurred and touchdown was performed manually. The switchover conditions between TAEM and autoland were very normal and smooth. A small pitch rate "bump" of about 1 deg/sec was observed in the data while the roll rate transient was not measurable. The speed loop was stabilized at 285 knots at the TAEM/autoland interface.

Figure 2-3 shows the EAS (estimated air speed) and descent trajectory during the final 10,000 ft of descent. Figure 2-4 shows wind speed with respect to runway position, speed brake position, and flightpath angle during the final 10,000 ft of descent. At 8000 ft altitude, the wind began shearing from 2 knots tailwind to a 3 knots headwind at 7000 ft. Then there was drastic shearing of the wind up to 22 knots at 6000 ft. This wind change caused the EAS to increase which in turn caused the speed brake to be driven out to a maximum value of 98.6 degrees at 5000 ft. The wind level started decreasing at 6000 ft, and the EAS reversed its trend at 5500 ft. This resulted in an overshoot of the 285 knot target-value down to 273 knots at 3500 ft and because the speed brake setting at 4000 ft (38 degrees) was less than the software criteria of 55 degrees, the system retracted the speed brake to the closed position.

Postflight simulations have confirmed this speed control action for the wind profile flown. The same type of airspeed build up at 6000 ft as in the flight data was reproduced with slight differences in the magnitude (296 knots flight vs. 300 knots simulation). The

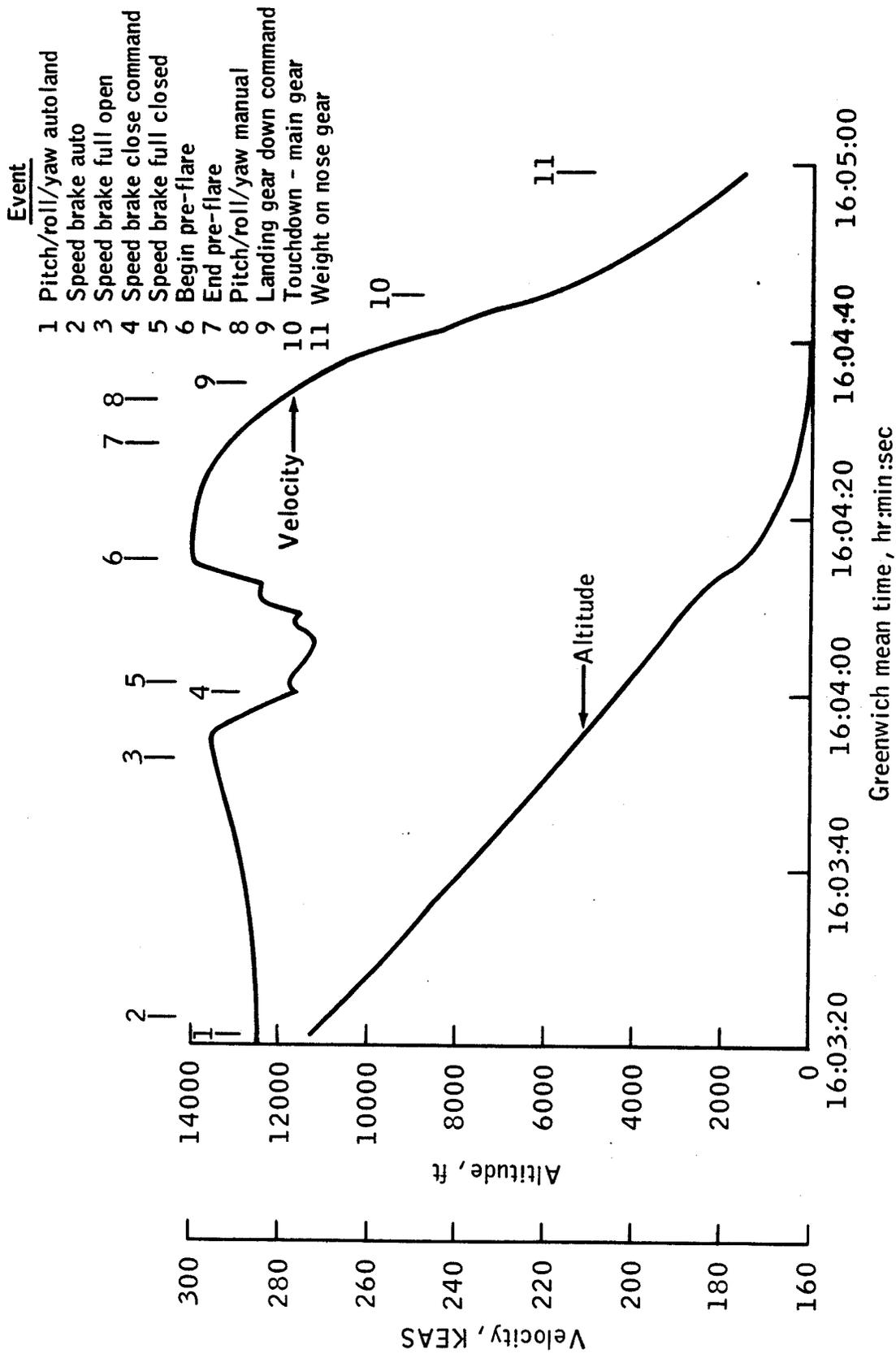


Figure 2-3.- Trajectory and velocity data during final portions of entry.

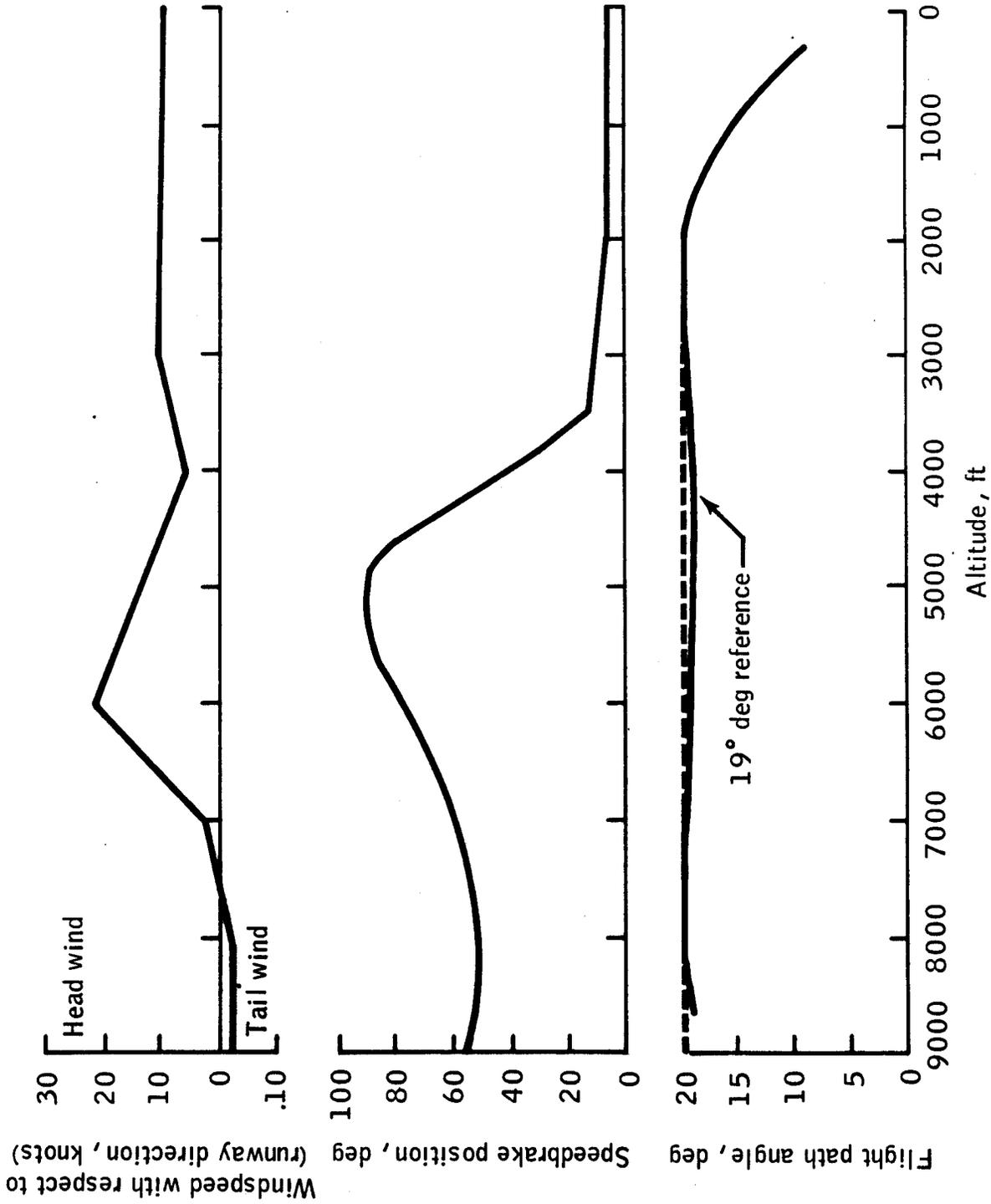


Figure 2-4.- Windspeed, speedbrake position, and flight path angle during final descent.

simulated speed brake data show it was driven to full open as in the flight data and retracted to the closed position at 4000 ft. If the autoland had flown the vehicle to touchdown, the following conditions would have resulted:

Feet from threshold (X_{TD}) = 4233 ft.

Estimated airspeed (V_{EAS}) = 201 knots

Rate of descent (\dot{h}) = -1.3 ft/sec

Landing gear deployment was initiated using airspeed as a cue with the result that the gear were down and locked 2 seconds before touchdown. Additional simulation runs were made using 200-ft altitude as a cue for landing gear deployment rather than 270 knots airspeed. If the gear had been deployed at 200-ft altitude, the touchdown conditions would have been the following:

X_{TD} = 3994 ft

V_{EAS} = 196 knots

\dot{h} = -1.5 ft/sec

STS-4 will use 200-ft altitude as a landing gear deployment cue.

The crew "perceived" that they were low and fast at the takeover from the autoland. Based on flight data analysis and comparisons with simulation trajectory data, the vehicle was very close to the reference trajectory at the crew takeover point. The guidance altitude error was essentially zero. Because of the wind profile, the speed at takeover from autoland was approximately 6 knots faster than predicted at crew takeover which is a 6-percent error. This type of speed dispersion had been seen in the verification data and is normal.

In summary, the autoland did its job within predictable and acceptable conditions. The system exhibited good, stable tracking of the reference trajectory. The speed control was acceptable. The autoland DTO is satisfied and the system is acceptable for testing to touchdown.

2.3.4 Communications and Tracking

The communications and tracking system provided good command, data and voice transmission throughout the mission except for several S-band equipment failures. The teleprinter operation was flawless and the RF navigation aids provided good data.

2.3.4.1 S-Band Network Equipment: The S-Band network equipment performance was normal for the first half of the mission. However, at 84:21:46:20 G.m.t., the PM (phase modulation) transponder 2, power amplifier (PA) 1 and preamplifier 1 were configured for Spaceflight Tracking Data Network (STDN) high power, high frequency (HF) operations to conduct the PDP (plasma dynamics package) EMI (electromagnetic interference) test. A ground command was received from Hawaii at 85:00:37:30 G.m.t. to return to the STDN low-power mode, bypassing power amplifier 1. The PM downlink was immediately lost and transponder 2 receiver AGC (automatic gain control) indicated approximately a 28 dB loss in received signal strength (Section 7.0, flight test problem report 21). Subsequent inflight troubleshooting did not restore the downlink in the transponder 2 low-power mode.

A troubleshooting procedure was developed to recover the transponder 1 downlink by a series of ground commands. At 85:18:13:25 G.m.t., transponder 1 was selected in the STDN high power, HF mode and the downlink was lost. The transponder was cycled through the transmit/receive and receive-only modes via the GCIL, but the downlink was not recovered. At 85:18:18:28 G.m.t., transponder 2 was reselected in the STDN high power, HF mode along with PA 1 and preamplifier 1. The downlink was recovered and the PM system was left in this configuration for the rest of the mission.

An S-band post-landing test procedure was developed and conducted at White Sands. The procedure exercised transponder 1 and then transponder 2 in the failed modes using panel and keyboard originated commands. Midway through the test, the failed functions were recovered. The test resulted in the recovery of all functions for both transponders.

Transponder 1 (serial number 301), was selected for failure analysis. Section 7.0, flight test problem reports 21 and 23 discuss the failure analysis activities.

As stated earlier, transponder 2, PA 1 and preamplifier 1 were selected in the STDN high power mode that they stayed in for the rest of the mission at 85:04:10:49 G.m.t. After 6 to 8 hours of operation, a 10-W drop in the PA 1 RF output was observed (Section 7.0, flight test problem report 28). The rate of drop slowed as the mission continued and since the power output was more than adequate, no change was made in the PA selection for the remainder of the mission. By the end of the mission, the power output had degraded from 130 W to 103 W.

2.3.4.2 UHF Transceivers: The Orbiter EVA/ATC (extravehicular activity/air traffic controller) UHF equipment performance was good during all mission phases. The ground station squelch was optimized for maximum noise coverage during weak signal conditions.

2.3.4.3 Audio Distribution: During the mission, the crew primarily used the WCCU (wireless crew communication unit) and the MHS (mini headset). The Commander used his molded earpiece and the Pilot used a universal earpiece. Neither reported trouble with the earpiece. The Commander had some trouble keeping the MHS on, due to cable stiffness.

During the first sleep period, the crew had trouble configuring the SMU's (speaker microphone unit) for sleep. The SMU's squeal when they were both on or when an active headset was too close. Later in the mission, the SMU's were adjusted to work properly with adequate air-to-ground communication. The final sleep configuration had the SMU's in the listen-only mode and a headset nearby. Using the SMU at night was considered essential since it provided a welcome relief to wearing an earplug.

On the first two nights, the Commander slept using an HIU (headset interface unit). At the northern latitudes of the orbit, interference was heard. Testing on the ground showed the HIU/MHS combination is susceptible to EMI (electromagnetic interference) for signals greater than 2 V/m between 430 MHz and 1200 MHz and susceptible between 600 MHz and 900 MHz at 1 V/m. EMI may have entered the audio system through the high gain amplifier in the HIU, but it is considered unlikely due to the high signal strengths required. The more likely source of the noise was through the WCCU. The WCCU wall units stayed on throughout the mission. Radiation at ultra high frequencies could have been detected by the WCCU receiver and the resulting noise could have been injected into the audio system.

Both crewmen reported excellent audio quality. During launch, audio was completely recognizable and at an adequate level. On-orbit audio was clear and there was adequate range on the volume control. Both reported the VOX (voice operated transmission) was excellent.

2.3.4.4 Wireless Crew Communications: The wireless crew communications system performed satisfactorily throughout the mission.

One malfunction was detected on day 5 when the crew reported that the Commander was unable to transmit, but could still receive (Section 7.0, flight test problem report 25). After replacing the battery did not clear the problem, the spare wireless set was deployed and normal communications were restored.

Of the 28 batteries stowed with the wireless hardware, 8 were returned with the protective tape still in place over the connection and apparently unused. This is well below the anticipated battery usage. A partial explanation is the fact that the crew had difficulty in detecting when batteries became depleted and on several occasions one of them lost communications because of a dead battery. A pre-set note in the crew activities plan is being considered to avoid this on future missions.

During the crew debriefing, it was learned that the wireless wall units were left on throughout the mission. This could have contributed to several noise problems since, in this configuration, two active receivers were connected to the audio system and when the leg units were turned off these receivers could easily pick-up RF emissions in their frequency range. Future crews will be advised to turn off both leg units and wall units when they wish to power down the wireless system.

2.3.4.5 Teleprinter: The teleprinter provided clear message reproduction throughout the mission with no anomalies or failures. A review of the messages transmitted disclosed no character errors other than those which occurred due to loss of signal (LOS) during certain message transmissions.

2.3.4.6 Television: The CCTV (closed-circuit television) provided good quality scenes to the ground in real time and by playback from the onboard video tape recorder (VTR). The CCTV system was controlled by ground- and crew-originated commands. Five problems occurred with the system and these are discussed in the following paragraphs.

The RMS (remote manipulator system) wrist camera and the payload bay camera "C" experienced component failures resulting in electrical shorts (Section 7.0, flight test problem report 8 and 15, respectively). The RMS elbow camera on STS-2 experienced the same failure mode. In each case, the camera temperatures were approximately 0° C.

A VTR cassette jammed in the recorder while the crew was removing the cassette (Section 7.0, flight test problem report 13). The VTR requires approximately 10 seconds to release the tape from the recorder drum mechanism when the recorder is stopped. If either the eject or power off button are pushed before the tape is released, the tape will be stripped from the cassette and jam the VTR.

The VTR run light operated improperly. The crew observed the "NO VIDEO" light on the VTR at various times during the mission (Section 7.0, flight test problem report 40). The illumination of the "NO VIDEO" light is intended to indicate the absence of video at the VTR. However, the crew played back the tapes and verified good recordings.

The aft bulkhead camera B lens assembly zoom function jammed during a real-time TV pass (Section 7.0, flight test problem report 20). The lens assembly was removed at KSC and returned to the vendor. A replacement lens assembly was installed on Camera B.

2.3.4.7 RF Navigation Aids.-

2.3.4.7.1 TACAN: - The TACAN's were turned on prior to de-orbit and began to receive intermittent data at an altitude of approximately 150K ft and a range of 300 nmi. Solid lock-on of both the range and bearing for all three units occurred at an altitude of 135K ft and a range of 218 nmi. Thirty-eight seconds later, the data were allowed to be incorporated into the navigation computer, successfully guiding the Orbiter to the landing area.

As on previous flights during roll maneuvers, 40° bearing jumps were experienced in the data and these jumps resulted in the deselection of unit 2 bearing by RM (redundancy management) at approximately 102K ft altitude and a range of 94 nmi. As before, the data returned to normal but was not reselected by the crew. At this same time, the unit 2 range lock was lost for 12 seconds and this resulted in a switch from the lower to the upper antenna for that unit. Several intermittent 40° jumps occurred later at various times on all three units, but were not of significant duration to cause any additional RM problems.

On-orbit TACAN data were obtained in support of DTO 379, to determine the feasibility of utilizing TACAN range data for orbital navigation. Range data from 17 different ground stations around the world were recorded with consecutive lock times varying from 4 seconds at Brisbane, Australia, to 95 seconds at Laverton, Australia. Although considerable bearing data were obtained, there is no plan to attempt to use it for on-orbit navigation due to the unreliability at high elevation angles and the possibility of co-channel interference.

2.3.4.7.2 MSBLS: The performance of the MSBLS (microwave scanning beam landing system) navigation equipment was as expected and their data output supported the autoland portion of the landing. The three functions azimuth, elevation and distance, for all three units were properly locked as follows:

	Greenwich mean time, day:hr:min:sec	Average data
Azimuth	089:16:02:34	14.3°
Elevation	089:16:01:53	29.6°
Distance	089:16:02:26	14.06 nmi.

The three functions met the selected criteria as follows:

	Greenwich mean time, day:hr:min:sec	Average data
Azimuth	089:16:03:03	3.2°
Elevation	089:16:03:03	15.47°
Distance	089:16:02:21	14.14 nmi.

As expected, azimuth angle differences between the three units were noted. These differences became as large as 0.3 degree with the most probable cause being the errors introduced by the MSBLS ground station azimuth antenna cross-polarization radiation (a change to the MSBLS ground station azimuth antenna to reduce this cross-polarized radiation has been approved for STS-5 and subsequent). Azimuth data were provided during the rollout until the Orbiter went outside the azimuth coverage of 15°. The units provided excellent elevation data until the Orbiter passed the elevation ground station, when the data were properly flagged invalid. Also, three units provided excellent distance data, with very small differences between the three. Distance data were provided through rollout and continued after the Orbiter had stopped.

2.3.4.7.3 Radar altimeter: Radar altimeter no. 1 locked on at an altitude of 3921 ft and no. 2 locked on at 5240 feet. Both tracked very well through touchdown. A modification had been made to both units for STS-3 to correct the problem of nose landing gear interference that was experienced on STS-1 and STS-2. There were no indications in the data of any influence of the landing gear on the altitude readings; however, since the gear deployment was closer to the ground than normal, a further evaluation of the modifications will be done on STS-4.

2.3.5 Hardware Performance

2.3.5.1 Star Trackers.- All scheduled IMU/ST (inertial measurement unit/star tracker) alignments were successfully accomplished. The crew monitored the status of target suppress shutter closure, opening the shutter via GPC (general purpose computer) commands as required. This resulted in obtaining 86 star-of-opportunity alignments, 8 of which were used to align the IMU's instead of scheduled alignments. This demonstrated the star-of-opportunity alignment mode and saved some RCS fuel and crew time.

The star tracker acquired and intermittently tracked the faintest star in the onboard catalog (no. 60, ENIF, 3.0 magnitude) when it was approximately 30 degrees from the sun, thus demonstrating the light shade performed as required for sun protection in daylight.

Also acquired (and rejected) were several cases of sunlit particles and debris in the vicinity of the spacecraft. None of these were misidentified as a star, indicating that the software logic to reject such acquisitions is working properly.

A thermal DTO (312-04) to monitor ST and navigation base temperatures during prolonged thermal soaking (hot and cold) was performed. Temperatures stayed in a 60 to 90° F range, with less than 2° F differentials across the navigation base indicating a very low thermal stress condition.

Anomalies were the same as on STS-1 and STS-2; the target suppress function operated as expected with shutter closures occurring at various times. The ST bite function detected several more Manchester coding errors in command words received from the ST MDM's (multiplexer/demultiplexers).

Postflight visual inspections at both White Sands and the KSC-Orbiter Processing Facility showed a minimal amount of particulate contamination in the ST light shades; this was vacuumed away and will not impact STS-4.

The -Z star tracker detected many single occurrences of improper word codings in the command word coming from the MDM. Approximately 120 instances were detected during the 172 hours of orbital star tracker operation for an error rate of 1.94 per million. The star tracker operation and performance were not affected. This problem was also seen on STS-2.

2.3.5.2 IMU Performance.- IMU performance was normal. All three IMU's were selected alternately for navigation by the onboard RM. The IMU relative performance was the same as STS-2 which was slightly larger than one sigma.

Approximately 12 hours into the mission, the IMU 3 X-Y accelerometer bias shifted -321 μ g and +383 μ g, respectively. These biases were updated and remained stable throughout the mission. The Z accelerometer of IMU 3 had a one sigma shift throughout the mission. (Section 7.0, flight test problem report 11). The other IMU accelerometer biases were within 40 μ g throughout the mission ($1\sigma = 50\mu$ g).

There were nine gyro bias compensation updates made. The largest gyro bias update was 0.047 deg/hr for the IMU-3 Z gyro. The IMU 2 X gyro bias exhibited some bias trending at the beginning of the mission which was commensurate with the warm-up characteristics

previously determined for this IMU. Gyro bias changes from prelaunch to pre-entry were less than 1 sigma (0.035 deg/hr) for all axes except the IMU 2 axis which was 2 sigma. The mission could have been safely completed without any updates.

2.3.5.3 Actuator Subsystems.-

2.3.5.3.1 OMS actuation subsystem: The OMS actuation subsystem performed normally throughout the mission. The rates observed at the pre-deorbit profile point were in agreement with the rates noted at prelaunch.

2.3.5.3.2 Aerosurface actuation subsystem: The aerosurface actuation subsystem performed normally throughout the STS-3 mission. Secondary pressure oscillations were noted on the left-outboard elevon actuator channels 1 and 3 and on the right-outboard elevon actuator channels 3 and 4 during the third through sixth elevon ramp command cycles of the entry hydraulic thermal conditioning procedure. The worst-case secondary pressure oscillation occurred on the left-outboard elevon actuator channel 3 and was 200 psid peak-to-peak at an frequency of 6 to 7 Hz. The frequency of oscillation is most probably a higher frequency which has been reduced to 6 to 7 Hz by the aliasing effect of the 25 Hz data sampling rate. Elevon actuator pressure oscillations similar to the STS-3 oscillations have been observed in FCHL (Flight Control Hydraulics Laboratory) actuator testing and have been determined to be tolerable.

Secondary pressure oscillations that could cause audible chatter were not evident in the data around Mach 3, the time at which the crew reported hearing a 10 to 12 Hz buzz or chatter. However, since the actuator return line pressure is not instrumented, oscillations could have existed in the return line pressure that could have caused audible chatter. In a few of the FCHL test cases, a low-level return line pressure oscillation was noted where no appreciable secondary pressure oscillations existed.

2.3.6 Data Processing Subsystem

Data processing subsystem hardware performance was essentially normal with the exception of a failed keyboard switch on the CDR's keyboard (Section 7.0, flight test problem report 29). The problem was rectified by inflight replacement using a keyboard switch from the aft station. The specific switch failure mechanism is not totally unexpected in view of the age of the keyboard and known characteristics of the particular switch. A superior switch design is being phased in on an attrition basis.

2.3.7 Backup Flight System

The BFS (backup flight system) performed its function as expected throughout all flight phases.

2.3.8 Displays and Controls

The performance of the D and C (displays and controls) system was excellent. The D and C system provided the crew with the capability to control and monitor the vehicle rotation, translation and flight path; control and monitor the status of onboard systems; and to detect and make safe hazardous conditions. The flight displays all worked properly and the crew reported that they were readable during all mission phases. The requirements of FTR 71VV02 were successfully completed for STS-3.

The C and W (caution and warning) subsystem operated very well with no system failures.

The cabin interior lighting and payload bay lighting were adequate for all mission phases. The remote manipulator arm light was also adequate, especially with its "instant-on" feature.

Two D and C system anomalies occurred during the STS-3 flight and are discussed in the following subparagraphs.

Prior to launch, the three-phase circuit breaker (MCA Power AC 3, CB11) on panel MA73C would not stay latched when depressed. An external mechanical latch was placed on the circuit breaker to support the flight (Section 7.0, flight test problem report 2).

During preparations for deorbit, the flight crew noted that the talkback (Payload Main Bus C Talkback, DS17) on panel R1A1, indicated power-off when the primary main C power was turned on; however the downlink measurement indicated power on. (Section 7.0, flight test problem report 32).

2.3.9 Instrumentation

2.3.9.1 Operational Instrumentation.- The data from seven OI (operational instrumentation) measurements were of questionable validity during STS-3. (Section 7.0, flight test problem report 10). Two of the measurements were in the MPS (main propulsion system) and three were in the hydraulics subsystem. One of the MPS measurements (V41T1261A, main engine 2 gaseous hydrogen pressurant temperature) had also failed on both STS-1 and STS-2. The measurement went off-scale high for 8 minutes starting at 81:16:00:33 G.m.t. and then returned to normal. This same failure mode was experienced on STS-2, but troubleshooting failed to isolate the cause of the malfunction. During the turnaround activities in preparation for STS-4, the sensor was replaced.

The other MPS measurement (V41P1300C, main engine 3, liquid hydrogen inlet pressure) had not failed before, nor had any engine 3 MPS measurement. It failed off-scale high at 81:16:07:40 G.m.t. and indicated that condition for the remainder of the flight.

The three hydraulic subsystem measurements that were questionable were V58T1143A, V58T0398A, and V58T0198A. The first, hydraulic system 1 mid-fuselage return-line temperature A, did not agree with other nearby temperatures as expected. The second, hydraulic system 3 rudder speed brake return-line temperature, indicated off-scale high by the telemetry system. This measurement had a history of wiring problems that are scheduled to be cleared up during turnaround activities. The third measurement, rudder speed brake return line, responses lagged another measurement on the same line.

A sixth measurement was located in the PRSD (power reactant supply and distribution system), and is provided with the PRSD tankage, but conditioned by the OI. This measurement, PRSD hydrogen tank 2 quantity (V45Q2205A) failed high at 83:06:45:22 G.m.t. on both the telemetry and the cabin meter.

The last measurement (V62Q9150, water tank E quantity) shifted from 85 percent full to zero and then returned to the normal reading. The problem was known to be caused by contamination of the sensor wiper, but this tank was removed for STS-4 and no action was taken.

There were no other failures in the more than 3000 OI components.

2.3.9.2 Development Flight Instrumentation: The DFI (development flight instrumentation) system performed satisfactorily except for loss of the wideband mission recorder during sequence 2 of the RCS engine LD5 firing. The wideband ascent recorder operated satisfactorily during ascent and OMS-1 and -2 firings. The PCM (pulse code modulation) recorder operated during ascent, OMS-1 and -2 firings, and during the deorbit firing, entry and landing. On-orbit, the PCM recorder generally remained in the low-sample-rate mode except for selected DTO's. The wideband mission recorder operated satisfactorily during ascent, OMS-1 and -2 firings and during the RCS engine L2U firing and sequence 1 of the RCS engine LD5 firing. However, during sequence 2 of the RCS engine LD5 firing at 86:17:08:15 G.m.t., the recorder failed due to loss of tape tension servo control (Section 7.0, flight test problem report 27). However, eight wideband flutter measurements for entry were satisfactorily transmitted on the DFI RF downlink. As no GSE recorder-playback facilities were available at the Northrup Strip landing site, the three DFI recorders were removed from the Orbiter and satisfactorily dumped at JSC.

Approximately 1 percent of the 3500 DFI measurements had newly discrepant conditions during this flight. These will be repaired where accessible prior to STS-4. The DFI RF downlink transmission operated satisfactorily, and PCM data were recorded at the S-band ground stations during vehicle signal acquisition periods.

The extended DFI/DATE system (27 measurements on OSS-1 pallet and 11 on Orbiter side of pallet interface) operated satisfactorily through the OEX recorder during ascent and entry.

2.3.10 Systems Management

The SM (systems management) for STS-3 operated normally.

2.3.11 Redundancy Management

The RM performance was excellent, with significant events occurring within the RCS and TACAN areas. In all areas, RM successfully provided the best source data to all users (selection filtering) while maintaining a comfortable margin of component performance evaluation when compared to the RM fault detection thresholds.

RCS leak alerts and engine deselections occurred on vernier engines F5L and F5R during on-orbit operation. Although it is known that neither of these engines were actually leaking, the RM alert is not considered to be a false alarm deficiency, but rather a known potential resulting from procedures requiring vernier engine heaters to be powered off prior to entry and the resulting cool down of those engines below the RM leak thresholds used in the on-orbit software.

RCS leak alerts and engine deselections also occurred on two primary engines (L4U and R2U) during the early portion of entry. Again, these alarms did not result from actual leaking conditions on either engine, but were the results of a known phenomena in the hardware operation under conditions as seen in STS-3. The alarms had been predicted and the crew was advised of their probability prior to their occurrence. Software modifications will alleviate this condition starting with STS-5.

TACAN 2 failed and was deselected at 89:15:57:37 G.m.t. due to errors in bearing readings. This alarm resulted from a known hardware performance sensitivity (also occurred on STS-1 and STS-2) and the unit could have been manually reselected had the need arisen later in the flight. Software modifications to eliminate this sensitivity are under consideration.

2.4 ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM

2.4.1 Active Thermal Control Subsystem

The active thermal control subsystem performed satisfactorily during the mission. Several unexpected conditions are discussed in the following paragraphs.

The STS-2 active thermal control subsystem configuration and mode of operation were identical to STS-2 with the exception of the entry flow through the radiators occurring at an altitude of about 150,000 ft as compared to the STS-2 entry flow at about 100,000 ft. The developmental flight instrumentation coolant loop, package 2, Freon outlet temperature was operational during STS-1 and determined to be inoperative prior to STS-2 was intentionally left inoperative for STS-3 since it was considered noncritical for future flights. No launch commit criteria were violated during prelaunch operations.

The STS-3 data portions of FTR 63VV001 (Active Thermal Control System Performance), 63VV002 (Orbiter Inherent Thermal Capacity), and 63VV003 (ATCS Flash Evaporator Subsystem) are expected to be satisfied upon acquisition and review of flight data. Data covering FTO 312-05 (Topping Duct and Feedwater Line Thermal Response), FTO 312-06 (Flash Evaporator High Load Duct Thermal Response), and FTO 363-01 (Radiator Thermal Capacity Test - Cold Case) are being reviewed and the objectives are expected to be satisfied.

During ascent, about 1.5 minutes after lift-off, the unexplained cooling of the Freon coolant loops between the ammonia boiler inlet and the flash evaporator outlet occurred. This cooling also occurred during the STS-1 and STS-2 missions and is possibly due to the evaporation of condensation in the flash evaporator cores. Flight data indicates that the high load flash evaporator was inhibited too early during initial radiator activation. At the time it was inhibited, the evaporator inlet temperature was about 74° F, higher than the 60° F maximum inlet temperature for topping evaporator operation with the high load evaporator inhibited. No over-temperature shutdown of the evaporator occurred as a result of this error. However, the evaporator did lose control for about 100 seconds, during which time the evaporator outlet temperature peaked at 48° F. About 2.8 hours after launch, upon positioning the flow proportioning modules to the payload position, fault messages were triggered because the interchanger flowrates dropped below the 1950 lb/hr lower limit that is applicable while in the interchanger position and during the transition from the interchanger to the payload position.

The minimum radiator panel outlet temperatures and Freon coolant loop accumulator quantities occurred at the end of the first sleep period during the tail-to-Sun Orbiter attitude. Based on a total active thermal control subsystem heat load of 54,000 Btu/hr, the radiator outlet temperatures were expected to be -96° and -92° F for loops 1 and 2, respectively, and the accumulator quantities were expected to be 17 percent and 16 percent for loops 1 and 2, respectively. The actual flight readings with a heat load of 54,000 Btu/hr were -75° and -82° F in loops 1 and 2, respectively, and 17 percent and 15 percent in loops 1 and 2, respectively. For comparison, the accumulator quantities dropped to 20 percent and 18 percent during the two pre-entry radiator coldsoaks. To conserve consumables, the topping duct and nozzle heaters were turned off from 84:17:19 to 86:18:50 G.m.t., with the exception of the period during the conduct of the topping duct and feedwater line thermal response FTO (functional test objective) 312-05, from 85:01:36 to 85:17:07 G.m.t. During the payload bay door thermal gradient performance FTO (356-03), the flash evaporator primary B controller shut down. Figure 2-5 shows this shutdown and the subsequent successful restart with the primary A controller. A description of this shutdown and subsequent flash evaporator anomalies is given in section 7.0, flight test problem report 31.

From about 88:21:30 to about 89:13:00 G.m.t., the development flight instrumentation multiplexer/demultiplexer DC02 was deactivated. This in itself caused no ATCS-related problems. Due to the loss of some ATCS instrumentation, however, visibility of a possible Freon leak in either Freon coolant loop, and visibility of a possible failure of any of several high load or topping duct heaters was lost.

The ammonia boiler subsystem failed to provide proper temperature control at $34 \pm 3^\circ$ F with either the system A or B secondary controllers when activated after touchdown. Figure 2-6 shows this anomalous behavior and a description of it is given in section 7.0, flight test problem report 35.

Ground support equipment cooling was established at touchdown plus 30 minutes. A leaky quick disconnect, however, forced termination of this cooling. Later, at touchdown plus 35 minutes, proper ground support equipment cooling was established.

2.4.2 Air Revitalization Subsystem

The performance of the ARS (Air Revitalization Subsystem) was normal and within predicted values and no launch commit criteria were violated. Manual cabin air temperature control procedures identified for the thermally severe attitudes flown during STS-3 provided a comfortable crew environment. However, the crew prepared for the first two sleep periods (during, and just after the tail-to-Sun attitude) by dressing in warm clothes. Adequate avionics cooling was also provided.

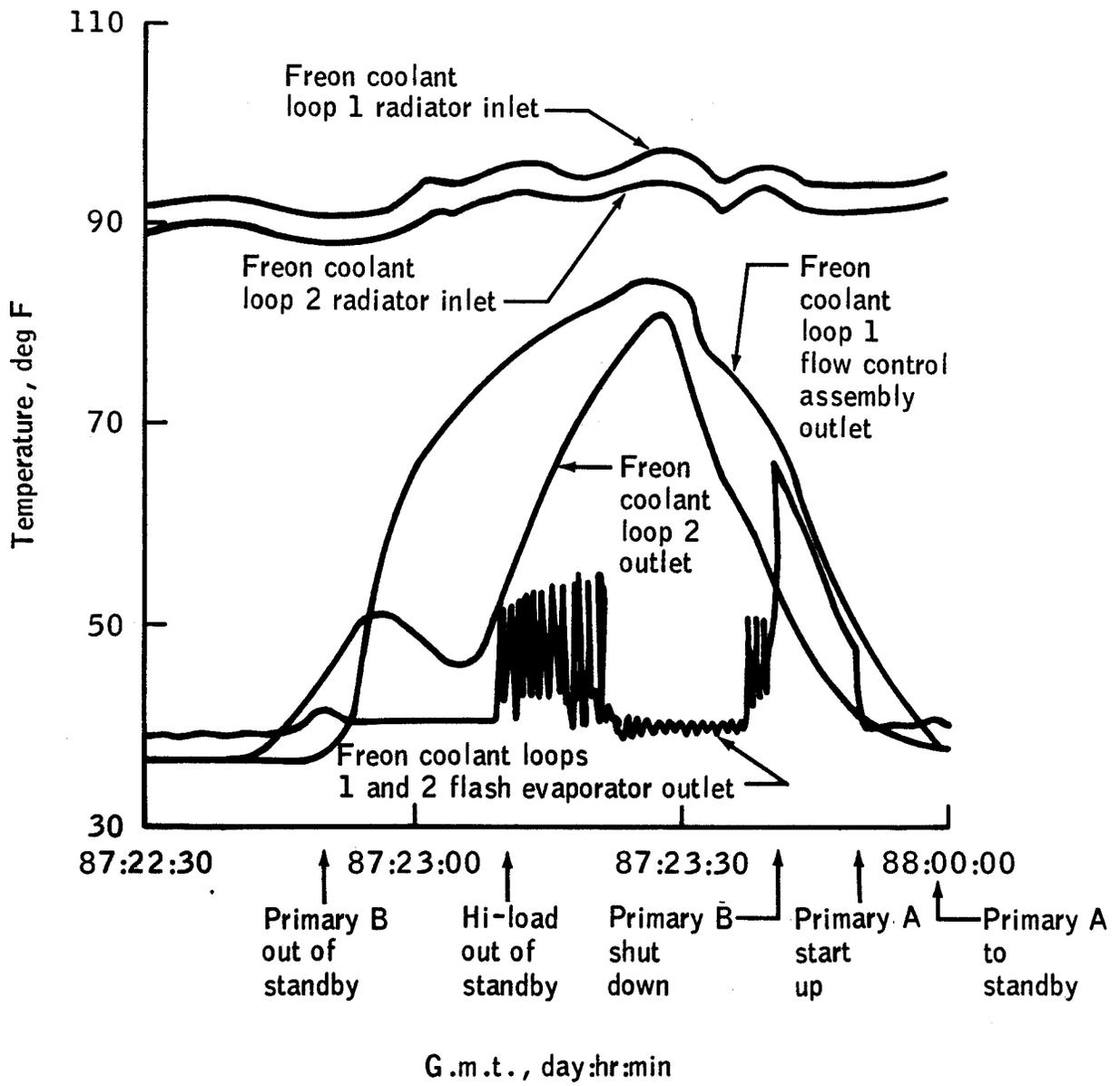
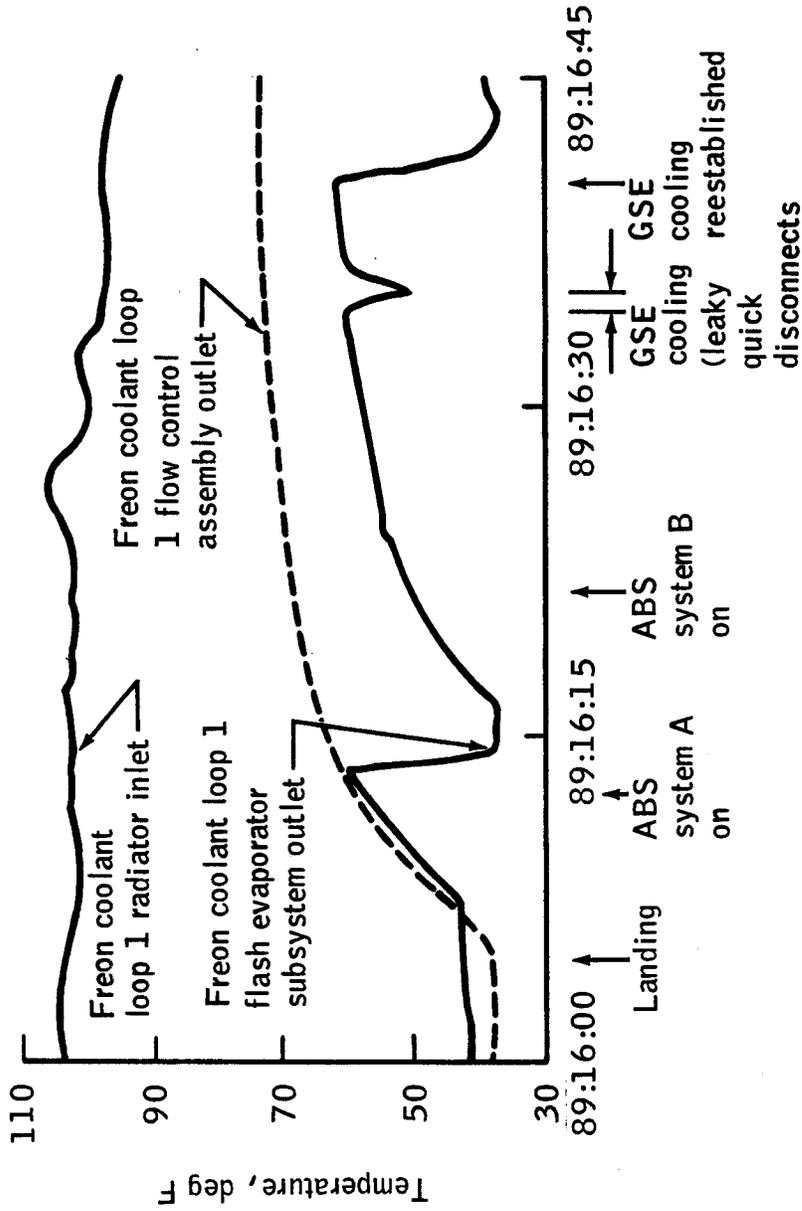


Figure 2-5.- Flash evaporator system primary B shutdown.



G.m.t., day:hr:min

Figure 2-6.- Ammonia boiler subsystem (ABS) postlanding anomaly.

The STS-3 air revitalization subsystem configuration differed from STS-2 in one respect. Full interchanger water flow was used for the first time as a normal planned procedure for cabin temperature control during cold attitudes. All other cabin temperature control was maintained through the normal use of the cabin heat-exchanger air-bypass valve.

Condensation formed on the aft flight windows in the tail-to-Sun and nose-to-Sun attitudes as had been predicted. This condensation was caused by low window pane temperatures because of exposure to extremely low payload bay temperatures. At the time the condensation was first mentioned by the crew, the cabin dew point temperature was approximately 48° F. This dew point is well within the specified 39° F to 61° F range. A collection of water was also noted and photographed by the crew, at the base of the flexible duct located on the airlock (Section 7.0, flight test problem report 44).

Specified test conditions for the stabilized on-orbit high and low cabin heat load requirements were completed, therefore, the STS-3 portion of FTR 61VV001 (ARS Performance) has been satisfied. The requirements and the test conditions of DTO 362 (Cabin Temperature Measurements) were accomplished. A total of nine cabin temperature surveys were obtained at selected times in various attitudes. The data recorded by the crew will satisfy the objective of FTR 61VV001a (Cabin Temperature Survey).

Cabin PPCO₂ remained at or below 0.6 mm Hg for most of the mission. Abnormal changeout of the LiOH cartridges resulted from confusion associated with real-time, in-flight changes to the detailed crew activity plan. This resulted in some cartridge usage exceeding the 48-hour life limit. After the entry day wave off, one partially expended cartridge and one fresh cartridge were reinstalled. The PPCO₂ was then reduced from about 6.4 mm Hg to 1.9 mm Hg. The PPCO₂ level was maintained at 1.9 mm Hg for approximately 2.5 hours and then slowly began to increase. As a result of the high PPCO₂ concentration (approximately 3.0 mm Hg compared to a nominal 0.6 mm Hg) before the cartridges were removed for entry, the cabin PPCO₂ reached a maximum of 8.4 mm Hg before the side hatch was opened after touchdown.

During postflight inspection of the LiOH cartridges and stowage area, it was noted that one of the used LiOH cartridges had a split lengthwise along the wire material next to the seam. In addition, it was noted that some LiOH cartridges had not been properly secured for entry and that wet trash had been stowed in the LiOH storage bin. Both the expended and unused LiOH cartridges will be investigated for any contamination due to exposure to the wet stowage. Investigation will also be required to determine the reason for the split in the one LiOH cartridge.

2.4.3 Air Revitalization Pressure Control Subsystem

The ARPCS (Air Revitalization Pressure Control Subsystem) performance was normal throughout the mission with the exception of leakage from the GN₂ system in the tail-to-Sun and nose-to-Sun attitudes (Section 7.0, flight test problem report 17). The cabin pressure shell leak rate was well below the specification leak rate. During ascent, the cabin dP/dt again exceeded the -0.05 psi/min alarm point and activated the Klaxon alarm. This limit has been exceeded in each of the first three test flights and is caused by the normal cabin expansion as the differential between cabin and ambient pressure increases. A change is being processed to establish a higher Klaxon alarm actuation rate of -0.08 psi/min.

There was no major configuration change to the ARPCS since STS-2 and no launch commit criteria were violated. The STS-3 portion of FTR 61VV001 was satisfied as the leak in the GN₂ system did not affect the ability of the ARPCS to provide cabin pressure control and oxygen for crew consumption.

During the cold tail-to-Sun and nose-to-Sun attitudes, a GN₂ leak of approximately 1.2 lb/hr was noted. This GN₂ leak disappeared during vehicle orientation to warm attitudes. Figure 2-7 reflects leak start and stop times, valve configuration and N₂ tank temperatures for the entire mission. A detailed description of this GN₂ leak is given in section 7.0, flight test problem report 17).

Because of the reduced atmospheric pressure at the higher elevation of White Sands, N.M., it was intended that the cabin be vented to ambient pressure through the cabin vent and cabin vent isolation valves before the side hatch was opened. This procedure was recommended because the side hatch was not certified for opening with a pressure differential of greater than 1.6 psid. However, the cabin was not vented and the side hatch was opened at a pressure differential of 2.38 psid. No apparent damage was noted.

2.4.4 Airlock Support Subsystem

There were no airlock or extravehicular activities scheduled for STS-3.

2.4.5 Water and Waste Management Subsystem

During STS-3, the water and waste management subsystem performance was satisfactory and no launch commit criteria were violated. Several anomalies occurred during the flight and are discussed in the following paragraphs.

On the first flight day, the crew reported excessive gas bubbles in the drinking water, but the bubbles decreased to a normal level during the rest of the flight. The supply water tank E quantity measurement reading was erratic because of instrument contamination; however, the instrument is self-cleaning and performed properly during the last two flight days.

Abnormal waste collection subsystem slinger operation occurred during STS-3 beginning the second flight day. The slinger circuit breaker subsequently opened while the slinger was in the high speed mode. In the low speed mode the slinger speed was about 60 rpm instead of the normal 200 to 300 rpm. Crew use of the waste collection subsystem was continued in the low-speed mode. A description of this anomaly is given in section 7.0, flight test problem report 14.

The waste collection subsystem vacuum vent knob came off its shaft. This problem is described in section 7.0, flight test problem report 19.

Smoke detector A in avionics bay 3A and the left flight deck smoke detector failed their self-tests during the flight. These detectors had already exceeded their life limit and were waived for flight until they failed. The self-test failures are caused by excessive current, as described in section 7.0, flight test problem report 7. The avionics bay 2 smoke detector A, gave erroneous alarms during the flight. The crew reset the alarm twice and received an erroneous alarm each time. The alarm was deactivated for the remainder of the flight by opening the circuit breaker. The circuit breaker also deactivates smoke detector B in avionics bay 3A. Since this left avionics bay 3A with no operating smoke detectors, a portable fire bottle was discharged into that bay in accordance with mission rules. A discussion is given in section 7.0, flight test problem report 33.

2.5 CREW STATION AND EQUIPMENT

2.5.1 Cabin Temperature Survey

The crew successfully completed the cabin temperature survey on STS-3. The crew obtained the eight temperature survey readings as required in the crew activity plan, and one additional survey was obtained due to the mission extension. The data obtained during STS-3 is

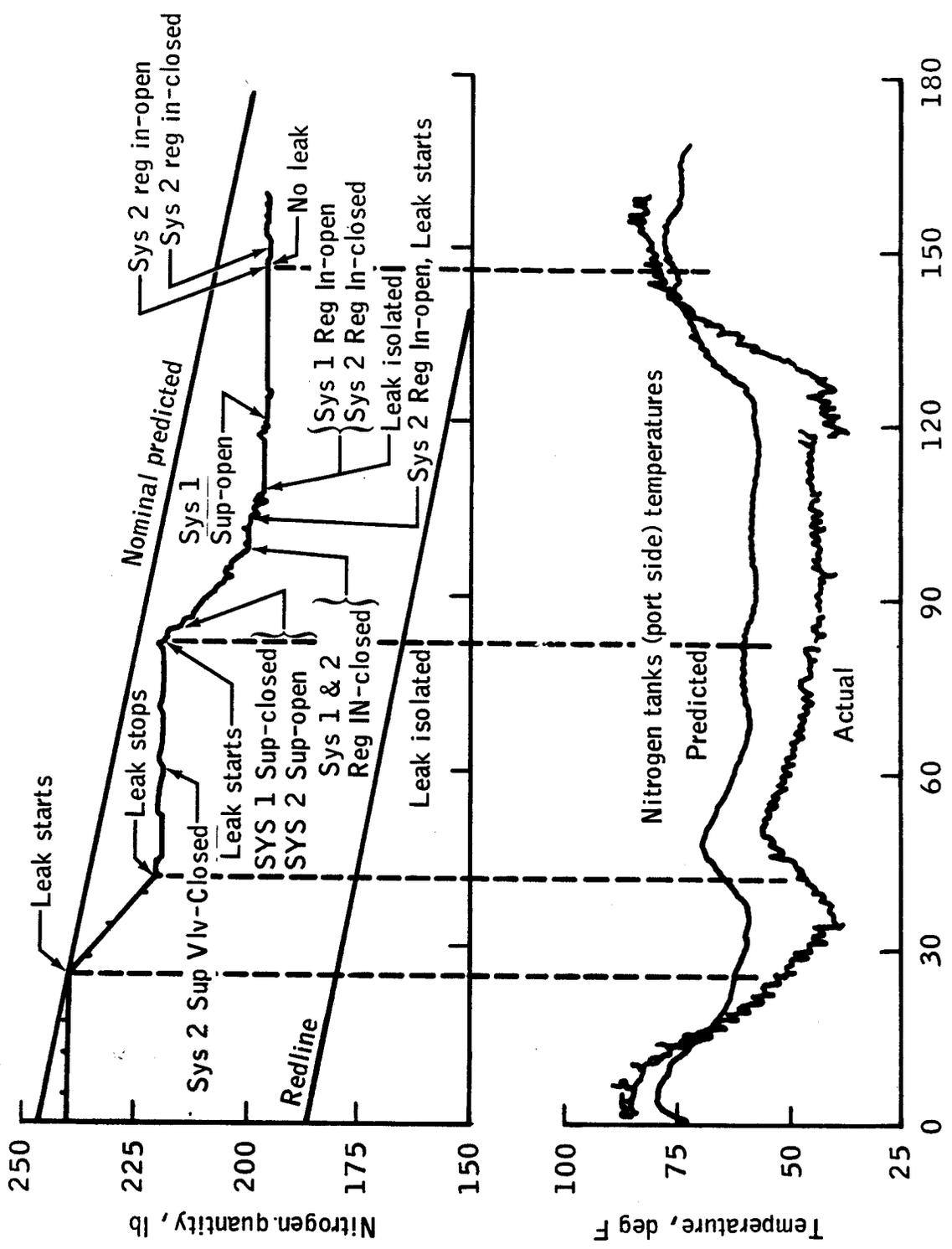


Figure 2-7.- Nitrogen leak data.

considered sufficient to select a new location for the two cabin temperature sensors used for cabin temperature measurement (downlink) and automatic cabin temperature control.

2.5.2 Flight Crew Equipment Cameras

2.5.2.1 General Operations.- The camera systems provided for STS-3 were used on previous programs except for a new high-speed 16mm camera and replacement of the Apollo 35mm camera with the NASA standard 35mm camera developed for Shuttle operations. Several newly developed films, some of which are commercially available, were evaluated on the flight. Most of the new films exhibited improved capabilities and will be used on future flights. The high-speed interior color film provided much improved photographs of the in-cockpit operations.

2.5.2.2 16mm Systems.- Eleven 16mm cameras were used on STS-3. One of the cameras was a high-speed unit operating at 100 frames per second to record debris through window W-4 in the cabin. This camera failed 1 1/2 seconds after SRB (solid rocket booster) ignition. (Section 7.0, flight test problem report 42). Other cameras inside the cabin were two data acquisition cameras (DAC's) from the Apollo Program. Twenty-one of the 22 available film magazines were used. Two magazines suffered loop loss and contained degraded imagery. The new ASA-400 film provided excellent exposures and crew usage of the system was excellent. The crew reported that the end-of-film light was inoperative on one of the DAC's in the crew compartment. (Section 7.0, flight test problem report 45).

Six modified DAC's, installed in environmental enclosures, were used as PDRS (payload deployment and retrieval system) viewing cameras during the payload bay operations. The heaters were not activated on orbit in sufficient time to allow camera temperature stabilization prior to operation, and five of the six cameras fuses opened from induced loads. (Section 7.0, flight test problem report 18). The sixth camera operated; however, preflight off-nominal handling of the magazine caused loop loss and image jitter.

The remaining 16mm cameras were two high-speed, 240-frame per second units located in the left-hand umbilical well to record SRB and ET (external tank) separation. These cameras performed as required; however, the cameras were inadvertently cycled during preflight operations at KSC, resulting in a reduced film quantity for flight use. Consequently, the latter part of the ET separation sequence was not recorded.

2.5.2.3 35mm Camera System.- Development of the new NASA-standard 35mm camera for the Shuttle Program was accelerated so that the camera could be used on the STS-3 mission to replace the manual 35mm camera used on previous flights. The new unit is a modified Nikon F3 camera and excellent photography was obtained using the camera. The unit uses a through-the-lens flash control system and a motor wind control. The camera was used to document experiments, crew activity, and some payload operations. The automatic exposure control system was used satisfactorily for out-the-window operations. The 35mm lens provided satisfactory wide-angle photographs. A second 35mm camera, one of the Apollo units, was enclosed in a thermal bag within one of the umbilical wells of the Orbiter and used to photograph the external tank and record data relative to the insulation. This camera functioned satisfactorily and provided data; however, the debris from separation impaired the through-the-window visibility for the camera.

2.5.2.4 70mm Camera System.- Three 70mm Hasselblad cameras were used to take 491 photographs during the STS-3 mission. Two cameras were used for Earth science and payload operations. Photography was excellent and no camera anomalies occurred. Exposure settings were taken from the standard tables and from the hand-held spotmeter for both the 70mm system and 16mm system. Photographs of the payload bay provided detailed documentation of the configuration. These units were also used to support the VCAP (Vehicle Charging and Potential) experiment. The third Hasselblad was of the Reseau type used on previous programs and supported the EEVT (Electropheris Equipment Verification Test) experiment.

2.5.3 Ancillary GFE and Flight Crew Equipment

Two hand-held calculators were flown on STS-3, with one worn by each crewman. The configuration changes from STS-2 were the addition of extended memory and function modules and a time module to each calculator. A flight calculator kit was configured with these same modules along with the two programs. Both the units and the programs functioned satisfactorily and no anomalies were indicated.

All other miscellaneous flight crew equipment such as pens, pencils, watches, timers, and binoculars performed satisfactorily.

2.6 STRUCTURES

2.6.1 Flutter/Buffer

Examination of response data from lifting and control surface instrumentation yielded no indication of flutter during entry (FTR 08VV010) (Lifting Surface - Control Flutter - Descent). Low-level control surface buffet (FTR 08VV012) was detected in the transonic region; however, this was anticipated. Response to structural PTI (programmed test inputs) was detected on all surfaces during both ascent and entry. The maximum accelerations noted on the lifting and control surfaces during STS-3 entry were essentially the same as noted on STS-1 and 2 and all were within design limits. Due to a failure of the wideband mission recorder, the only entry flutter data obtained were eight real-time accelerometers.

2.6.2 Stress Evaluation

Review of airframe and control surface strain data indicate that stress levels are similar to those measured on previous flights and that no areas of the structure experienced strains in excess of design values.

Strain data were obtained for the first time on this flight on the bottom skin of the mid-fuselage at station X=880 in., the region of highest predicted stress due to combined thermal and mechanical loading during entry. Strain data recorded at this location during STS-3 indicated stress levels higher than anticipated from pre-flight predictions. This was due primarily to higher than expected thermal stresses. Although the stress levels were higher than expected, positive margins of safety existed during entry maneuver conditions ($N_{zmax} = 1.6$ g's). In addition, a positive margin of safety can be shown for the mission limit load factor of $N_z = 2.0$ g's combined with STS-3 thermal stresses. Thermal stresses were obtained from the flight-measured stresses during maneuvers by subtracting induced mechanical-load stresses from the measured stress levels. The mechanical portion of the total stress was obtained using the ASKA finite element structural math model for a 2.5 g maneuver design case and ratioing to the STS-3 measured normal acceleration of 1.38 g's which occurred at the time of highest measured stress. These thermal stresses were then compared to thermal stress values calculated from a sectional 2-dimensional math model using flight-measured temperatures.

Measured and allowable stress levels are presented in Table 2-V along with thermal stresses derived from measured flight data and 2-dimensional analytical thermal stresses.

2.6.3 Entry Flight Loads

Entry loading conditions were within design limits for the Orbiter and within flight restrictions established for the STS-3 mission. The maximum vertical load factor experienced during the STS-3 entry was 1.6 g compared to a flight restriction of 2.0 g. The maximum load factors measured in the Orbiter payload bay are presented in the following table along with comparative values for STS-1 and STS-2.

TABLE 2-V.- MANEUVER STRESS VALUES

($N_z = 1.38 \text{ g's}$)

Gage location (X = 880)	Measured stress, psi	Allowable stress, psi	Measured thermal stress	2-D Calculated thermal stress
Y=0	-6895	-14,300	-2990	-3218
Y=30"	-5801	-14,455	-1880	-2296
Y=90"	-10,067	-16,330	-6845	-7149

ENTRY LOAD FACTORS

	STS-1	STS-2	STS-3
N_x	0.4/~0	0.4/~0	0.3/~0
N_y	0.1/~0.15	±0.2	<±0.1
N_z	1.65/~0	1.9/~0	1.6/~0

Maximum dynamic pressure during STS-3 entry was 315 psf at Mach 0.5 (the maximum allowable was 375 psf at a Mach number ≤ 5.0).

Landing gear vertical velocities at touchdown were within flight restrictions, but were the highest experienced so far in the test program. Main gear vertical impact velocity was approximately 5.7 ft/sec versus a 6 ft/sec restriction, and nose gear impact velocity was 8.8 ft/sec versus an 11.0 ft/sec design limit. A summary comparison of the responses at the time of nose gear accelerometers located in the crew cabin and mid-fuselage are presented in Table 2-VI.

2.6.4 Window Cavity Conditioning System

The window dessicant system functioned normally. No frost or fog was observed by the crew on the inner cavity window surfaces during the flight. Postflight inspection revealed a film deposit on the aft edge of the inner thermal pane surface of window number 5. It is believed that the film was deposited when the silicone thermal barrier contacted the hot window surface.

2.7 MECHANICAL SYSTEMS

The following mechanical systems performed the required functions during STS-3: ingress/egress hatch, purge and vent door drives, Orbiter/ET separation, payload bay door drives and latches, radiator deploy/stow and latch, manipulator positioning mechanism and retention latches, star tracker door drives, air data probes deploy/retract, and landing and deceleration. The aerothermal seals subsystem provided thermal protection for structural elements during ascent and entry. The ejection seats were required to perform the following primary functions: crew support and constraint; vertical positioning; back-angle positioning for ascent; suit oxygen and ventilation connections; and communication and biomedical connections. The airlock hatches A and B and seat ejection access door operational subsystems were not operated.

2.7.1 Purge, Vent, and Drain Subsystem

All purge and vent requirements were satisfactorily accomplished with no anomalies during the STS-3 flight. The prelaunch purge changeover from air to gaseous nitrogen occurred at 81:05:56 G.m.t. The prelaunch gaseous nitrogen purge flowrates and temperatures were:

	Flowrate, lbs/min	Temperature, °F
Forward fuselage purge circuit	98	90
Mid-fuselage purge circuit	179	70
Aft-fuselage purge circuit	109	105

The flowrates and temperatures were maintained through lift-off.

TABLE 2-VI.- ORBITER RESPONSE TO NOSE GEAR IMPACT AT LANDING

Axis	Measurement location			Airframe response, g			
	X ₀	Y ₀	Z ₀	STS-1 h=5.7 ft/sec	STS-2 h=5.1 ft/sec	STS-3 h=8.8 ft/sec	Preflight verification limit
Normal Y	511	3	424	0.1	0.14	0.2	0.21
Normal Z	511	3	424	1.85	1.48	3.7	4.08
Normal Z	825	-102	407	1.62	1.28	2.8	2.79
Normal Z	974	102	407	1.41	1.28	2.3	2.07
Normal Z	973	-102	407	1.40	1.22	2.3	2.23
Normal Y	979	-11	302	0.1	0.17	0.2	0.42
Normal X	1294	-2	297	0.58	0.52	0.6	0.84
Normal Y	1294	-2	300	0.1	0.14	0.2	0.23
Normal Z	1294	-2	289	1.38	1.21	2.2	1.92

The vent-door commands to open the vent doors commenced at T-38 seconds, and all vent doors were fully open by T-10.4 seconds, thus satisfying the launch commit criteria of all vents being open by T-4 seconds. All limit switches and actuator motors operated normally.

The vent doors remained open until the planned entry day when all the vent doors were closed except the right-hand forward and left-hand aft-fuselage/OMS pod vent doors, which remained open as planned. When the de-orbit maneuver was cancelled, the vent doors were reconfigured to the open position. This closure operation was repeated about 1 hour prior to the final-day de-orbit maneuver. The forward fuselage and aft-fuselage vent doors which remained open during the de-orbit maneuver were closed just before entry interface. The vent doors are closed prior to the high heating phase of entry to protect the structure from overheating. The vent doors were commanded open during descent when the ground relative velocity reached 2400 ft/sec and the vent doors were fully open 7 seconds later.

After landing, the crew commanded the vent doors to the post-landing purge configuration and the post-landing purge flowrates and temperatures were:

	Flowrate, lbs/min	Temperature, °F
Forward fuselage purge circuit	89	60
Mid-fuselage purge circuit	170	60
Aft-fuselage purge circuit	70	60

After landing, the forward fuselage purge flowrate was increased to 100 lb/min and the left-hand forward and aft vent doors were closed to preclude sand ingestion due to high winds.

A post-flight inspection of the wing vent relief doors showed that both the ascent and descent relief doors remained closed. Both the left and right descent relief doors opened on STS-2.

Measured internal compartment pressures have been reviewed for both ascent and entry. Predicted and measured pressure differentials for all compartments were within structural design limits. The data review also showed the motor 1 of vent door 3 did not receive feedback on its door closed status. Motor 1 remained on until the vent door system was commanded off. (Section 7.0, flight test problem report 34). This anomaly did not affect door operation in any way.

2.7.2 Orbiter/External Tank Separation and Umbilical Devices

The Orbiter/ET (external tank) separation subsystem (i.e., umbilical separation and retract, orbiter/ET structural separation and umbilical door closure) performed normally during STS-3.

Postflight inspection of the liquid oxygen and the liquid hydrogen umbilicals indicated that the aft retractor on the liquid hydrogen had not retracted. The cause was the premature shutdown of APU 3. The three liquid hydrogen umbilical retractors were at or above the mechanical locks. Loss of the retract function on the liquid hydrogen umbilical caused no problems.

Damage was noted in all of the liquid oxygen umbilical separation system blast containers. Two of the blast containers had external wall stress marks and bulges. The remaining blast container had a 3/4-in. long by 3/16-in. wide fracture in the sidewall area. The Kevlar wrapper around the blast containers performed satisfactorily.

The Orbiter/ET structural separation system performed normally on STS-3. Postflight inspection of the separation hardware showed that the forward structural attachment functioned as

required. The separation bolt/monoball assembly was rotated to the flush position by the centering mechanism, and the shear-bolt piston was recessed within the outer moldline 0.008 in., which is well within the aerothermal smoothness requirement of ± 0.030 in.

Separation of the aft structural attachments was also normal. All parts were intact and in good condition based on a postflight inspection at the landing site. The problem with loss of the ET inserts during STS-2 did not recur during this flight. Following the STS-2 incident, corrective action consisted of installing the inserts, close-out photographs of the insert installation just prior to ET mate, and removal of the alignment tips from the hole pluggers to minimize any interaction of the hole plugger and ET attach bolt during release and retraction of the bolt.

Review of the ET separation films and photographs confirmed that the attach bolts were retracted inside the ET fittings. Postflight inspection of STS-3, also confirmed that the aft attach hole pluggers, which minimize the escape of debris after separation, had closed off the Orbiter bolt holes. No debris was found inside the cavity between the ET umbilical and umbilical doors when these doors were opened after landing.

The ET umbilical doors closure was initiated (i.e. command to stow centerline latches) 20 minutes 28 seconds after lift-off, 32 seconds before the planned time. Door closure was completed 49 seconds later. The accumulative time for the operation of each ET door mechanical function to achieve door closure was 35 seconds. The specification time for two motor operation is 36 seconds. Therefore, the ET doors functioned normally during the STS-3 mission.

2.7.3 Payload Bay Doors and Radiators

The payload bay door operations conducted during STS-3 explored door performance under various thermal conditions thereby determining the operational temperature envelope. Table 2-VII summarizes temperatures measured during the start of payload bay door operation for the first three flights. Initial door operation followed 23 hours of tail-to-Sun (coldest environment) exposure. Following this exposure, the radiators were stowed and latched and the port door was closed. During latching of the bulkhead latches, a latch indication was not received on the aft bulkhead latch (Section 7.0, flight test problem report 9-2). Upon reopening the port door, the three forward ready-to-latch switches failed to transfer to the off position. (Section, 7.0, flight test problem report 9-1). The Orbiter was then maneuvered to the top-to-Sun attitude for thermal conditioning of the doors for 15 minutes followed by approximately 42 minutes of PTC (passive thermal control) after which the port door was once again closed and latched. This operation was normal and the ready-for-latch indicators yielded the proper indication following door opening. The doors functioned normally throughout the remainder of the flight. Operations following nose-to-Sun and tail-to-Sun attitudes were normal in all respects. Theodolite measurements were successfully taken for initial and final door alignments.

Post-mission analysis indicated that the cold environment experienced should not have effected latch actuator performance. Inspection at KSC revealed that the port door aft bulkhead latch actuator housing sustained damage to two of four mounting holes (figure 2-8). One crack at each hole was found (Section 7.0, flight test problem report 9-2). The ready-for-latch switch module was inspected in-place and no visual damage or debris were found. Three qualification units have been tested with the following results: Unit 1 switch mechanism experienced binding of the ready-for-latch switches at less than -80° F, the door closed switch at -170° F. A dimensional check of the switch mechanism indicated that close tolerances existed between the moving parts. Units 2 and 3 experienced no binding or actuation load changes in the -40 to -140° F temperature range and dimensional checks are in progress.

TABLE 2-VII.- TEMPERATURES FOR PAYLOAD BAY DOOR OPERATIONS

MEASUREMENT, °F	STS-1 ^a				STS-2 ^b				STS-3 ^c				
	1	2	3	4	1	2	3	3	1 ^d	2	3	4	5
Forward bulkhead latch motor (V37I9171)	76	49	19	-18	62	122	120			-2	60	95	21
Aft bulkhead latch motor V37I9172)	70	88	44	82	44	0	28			25	-2	128	45
Port drive actuator (V37I9181)	73	70	29	3	65	49	55			-75	-33	79	-60
Starboard drive actuator (V37I9182)	73	65	17	14	67	17	17			-90	-37	77	-75
Door structure (V34I9158)	55	7	0	-40	54	87	22			-70	10	203	37
Door structure (V34I9159)	52	20	0	-23	52	-15	-32			-78	-	190	30
582 Frame (V34I9200)	73	63	27	14	65	49	57			-53	-6	68	-33
582 Frame (V34I9201)	70	50	19	13	67	43	67			-62	0	58	-34
582 Frame (V34I9458)	-	-	-	24	65	40	57			-65	-10	68	-33
582 Frame (V34I9202)	73	62	29	13	60	14	18			-58	-16	70	-43
582 Frame (V34I9203)	70	54	23	21	60	14	25			-75	-10	60	-47
582 Frame (V34I9204)	70	55	22	29	65	29	45			-69	0	81	-33

TABLE 2-VII.- TEMPERATURES FOR PAYLOAD BAY DOOR OPERATIONS (Concluded)

MEASUREMENT, °F	STS-1 ^a				STS-2 ^b			STS-3 ^c				
	1	2	3	4	1	2	3	1 ^d	2	3	4	5
Aft bulkhead seal support (V35T9109)	80	64	37	18	81	50	40		-25	53	64	0
Sill longeron (V34T9127)	73	49	18	12	65	45	43		-75	-28	90	-35
Sill longeron (V34T9129)	67	47	18	13	65	57	57		-42	-20	90	-33
Sill longeron (V34T9130)	68	42	16	17	65	52	52		-50	-17	112	-38
Sill longeron (V34T9131)	73	52	18	27	60	13	18		-80	-28	100	-48
Sill longeron (V34T9133)	67	49	18	19	60	18	18		-50	-23	107	-41
Sill longeron (V34T9134)	65	52	13	18	60	25	22		-56	-20	106	-44
Seal (sill) (V37T9145)	75	24	21	24	58	32	15		-118	-50	150	-11
Payload baydoor pressure seal (V09T9545)	72	50	17	0	68	101	110		-62	-4	57	-28

NOTES:

^a 1. 1st Operation
2. 2nd Operation
3. 3rd Operation
4. 4th Operation

^b 1. 1st Operation
2. 2nd Operation
3. 3rd Operation

^c 1. 1st Operation
2. Tail-Sun Operation
3. Nose-Sun Operation
4. Top-Sun Operation
5. Tail-Sun + PTC

^d Temperatures not calculated.

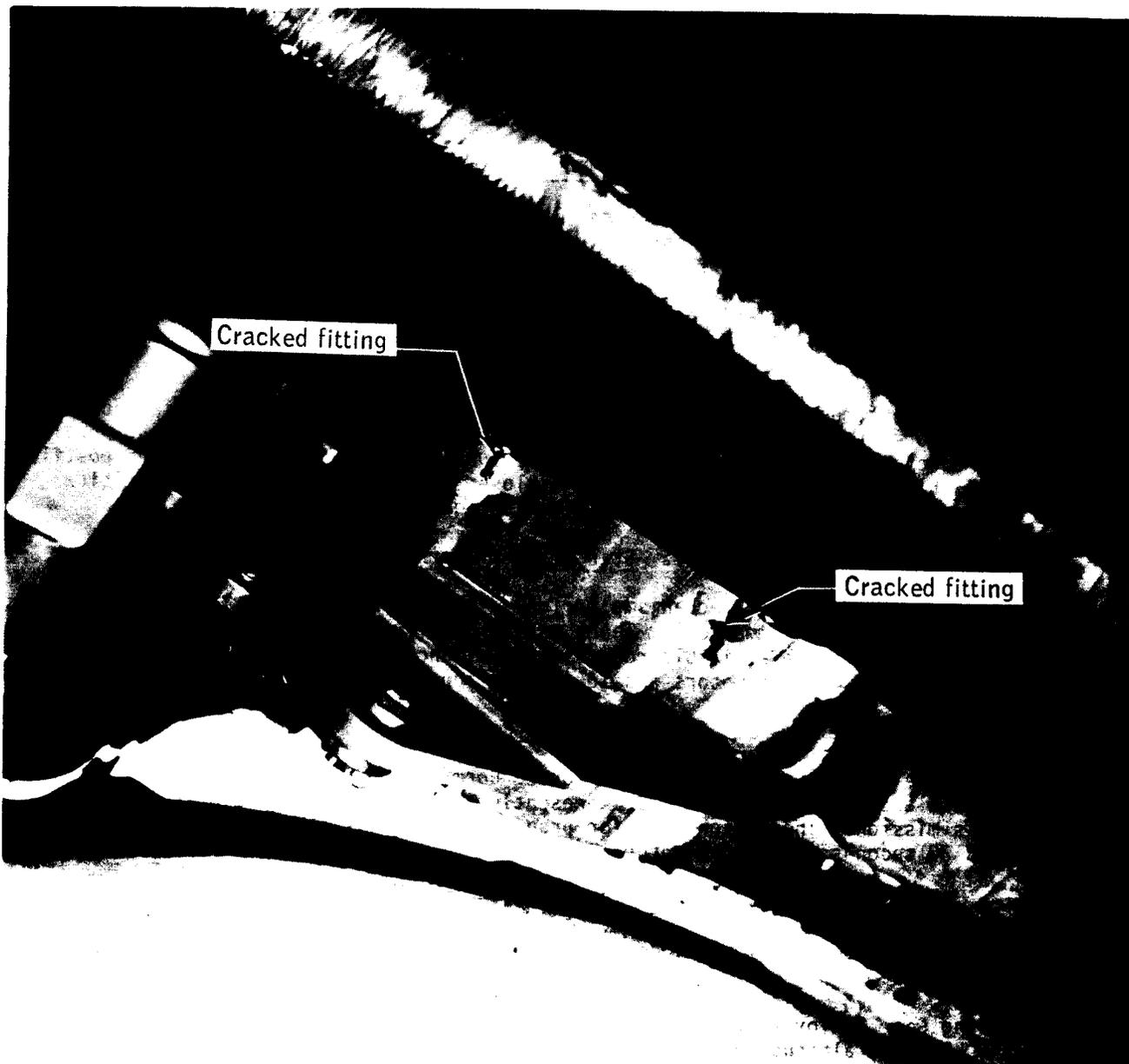


Figure 2-8.- Payload bay door hardware failure.

Radiator deployment and latching operations were normal for all functions except for several instances of the latch-release command switch being taken past the center off position to the release position. This was corrected by proper positioning of the switches to the latched position. However, only some latch motors were driven which would have resulted in less travel of the latching linkage. About 4 minutes prior to touchdown both port (1 through 6) latched indications and one starboard (7 through 12) latched indication transferred to indicate unlatched (Section 7.0, flight test problem report 36). Postlanding, all latch switches were placed in a latched position and within 1 second the latched indication returned, thus confirming that the latches were in a safe condition during landing.

2.7.4 Manipulator Positioning Mechanism

The RMS positioning mechanism performed normally during the STS-3 mission. The crew reported that the MPM (manipulator positioning mechanism) functioned satisfactorily during deployment and stowage operations. The MPM actuation times were within two-motor predicted time requirements. Postflight inspection revealed the actuation mechanism pedestal and linkage were properly aligned and locked. Also, the thermal blankets were secured and in place.

The MRL (manipulator retention latches) actuation times were normal, i.e. within predicted two-motor operation times. All switch indications were normal except for the failure to indicate ready-for-latch on one switch on one occasion. (Section 7.0, flight test problem report 12). However, this corrected itself on the subsequent operation. Detailed postflight inspection of this switch assembly failed to disclose the cause. This is not a critical switch function and no further corrective action is required.

2.7.5 Star tracker Doors

The star tracker door actuation mechanisms performed normally during the STS-3 mission (i.e. doors open and closed when commanded). There was no crew report of prolonged operation times, and no other data are available regarding this function. The design time for door actuation is 8 seconds for two motors and 16 seconds for single motor.

2.7.6 Air Data Probes

The air data probe deployment mechanisms performed normally during the STS-3 mission (i.e. deployed at Mach 3.5). No flight data are presently available to ascertain the deployment times for this mission. The design time for probe deployment is 15 seconds for two-motor operation and 30 seconds for single-motor operation.

2.7.7 Landing/Deceleration Subsystem

The landing/deceleration system provided normal performance during the STS-3 deployment, landing, and rollout.

2.7.7.1 Landing Gear Deployment.- Deployment of the landing gear was initiated at an air speed of 270 KEAS, an altitude of 99 feet, and 9 seconds before touchdown. The initiation time specified in the flight rules is 15 seconds before main gear contact, and at a velocity of 270 KEAS. Gear deployment will be initiated at 200-ft altitude on future flights. All landing gear were down and locked 2 seconds before touchdown. The deployment time of 7 seconds was well within the design requirement of 10 seconds. All deployment mechanisms, hydraulics, and pyrotechnic devices performed normally with no backup pyrotechnic systems being used.

2.7.7.2 Landing and Rollout.- Touchdown occurred at 89:16:04:44.845 G.m.t., at a sink rate of 5.7 ft/sec. Table 2-VIII provides detailed performance values regarding speed during landing. Figure 2-9 shows the time history of the ground speed during landing. Differential braking was used to maintain a course within 79 feet to the right of the runway centerline. Nose wheel steering was engaged at low speed near the end of the rollout. During the rollout, only light braking was applied resulting in a total roll distance of 13,732 ft.

Postflight inspection of the tires revealed that all tire inflation pressures were satisfactory and that one of the nose tires had an 18 in. long scrape on the sidewall. In addition, a small crack was found in the outboard beryllium rotor of the right-hand-outboard brake (Section 7.0, flight test problem report 43). As a result, all brakes were removed for inspection and failure analysis, and all tires have been changed.

2.7.8 Aerothermal Seals and Thermal Barriers

Postflight inspection revealed only slight-to-moderate damage to the aerothermal seals and thermal barriers as follows:

The nose landing gear door thermal barrier fabric was frayed in localized areas and additional debonding since STS-2 occurred along the mold line. Approximately 50 percent of this seal will be replaced prior to STS-4.

Several frayed and torn areas were found on the main landing gear door thermal barriers. These were less than 2.5 in. in length, and appeared to be caused by excessive RTV applied to the flow barriers which adhered to the thermal barriers and damaged them when the doors were opened. The ET umbilical door thermal barriers had two tears up to 3/4 in. in the outer cover, loose stitching on repair patches, and frayed areas, but not to the extent of prior missions. The 1 and 2 vent door thermal barriers, both left and right sides, had only minor frayed areas.

The payload bay door environmental seal expansion joints received minor damage typical of past missions.

The rudder speed brake perimeter Inconel spring seal was flattened and the end broken at the forward split line. Similar damage has been found at other locations on prior flights. The edge of the Macor support for the rudder speed brake split line thermal barrier was broken away for about 1 and 1/2 inches on the right-hand lower aft section near the trailing edge. The extent of the above damage was typical of that experienced during previous flights.

An inspection of the vertical tail thermal conical seal has revealed damage to 10 graphite seal blocks on the rudder longitudinal seal, which runs along the forward edge of the rudder panels. Damage consists of broken tabs of approximately 1/4 in. cube. These tabs are not required to retain the seal blocks in their channels and are acceptable for STS-4 (Section 7.0, flight test problem report 46).

2.7.9 Crew Escape System

The crew system ejection seats provided the crew with support and constraint, vertical positioning, back angle positioning for ascent, oxygen and ventilation connections for suited operations, and communications and biomedical connections. No difficulties were reported during the mission.

TABLE 2-VIII.- LANDING/DECELERATION SUBSYSTEMS PERFORMANCE STS-3

Parameter	Velocity, knots	
	Equivalent air speed	Ground relative velocity
Main gear touchdown ^a	220	230
Nose gear touchdown	173	178
Braking initiated	142	149 on 36 off 24 on 0 off
Nose Wheel Steering		36 on 16 off
Touchdown points from threshold		
Left main, ft.		1163
Right main, ft.		1092
Distance from main to nose wheel contact, ft.		5169
Distance from nose contact to brake initiation, ft.		2072
Braked roll, ft.		5554
Braked duration, seconds		52
Pitch rate at nose wheel contact, deg/sec		7.2
Sink rate at main gear touchdown, ft/sec		5.7
Total rollout, ft ^b		13,732
Rollout duration, seconds		84
NOTES:		
^a Touchdown 29 ft. to right of runway centerline.		
^b Drifted 78.74 ft. to the right of the centerline during rollout, cross 8 ft to left of centerline during nose wheel steering and stopped at 9 ft to the right of the centerline.		

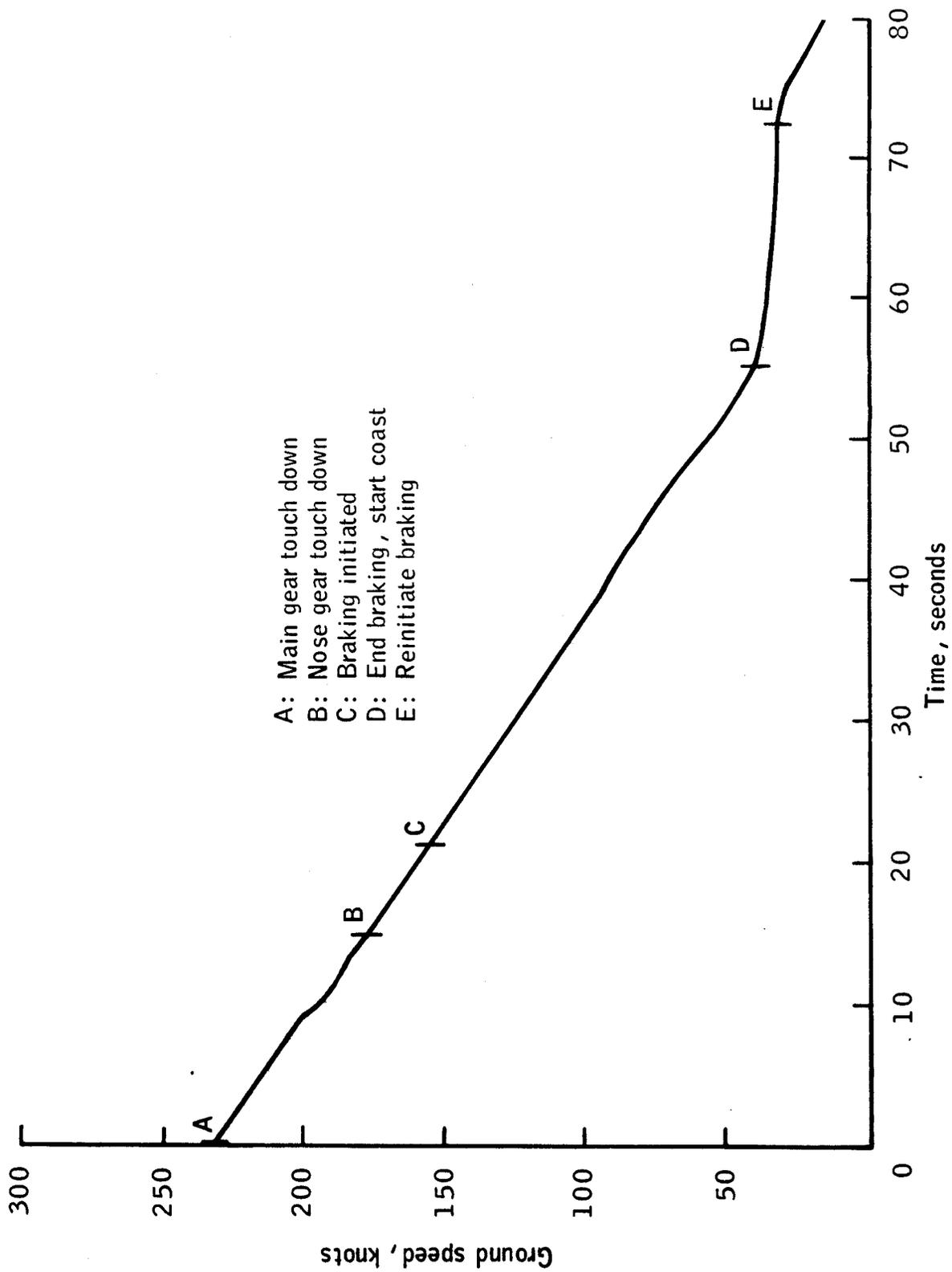


Figure 2-9.- Speed during rollout and landing.

2.8 THERMAL

2.8.1 Thermal Control System

Three major attitudes, tail-to-Sun orbital rate, nose-to-Sun with a one revolution per orbit roll, and top-to-Sun solar inertial followed by a passive thermal control mode (PTC) (four revolutions per hour roll about the X-axis with the X-axis perpendicular to the solar vector) were accomplished during STS-3. Except for the tail-to-Sun attitude, the forward fuselage bondline temperatures appear to have reached equilibrium in each of the attitudes. Table 2-IX presents the initial and final temperatures as well as the average response rate for each attitude. For the cold tail-to-Sun attitude, bondline temperatures range from +4° F to -62° F for the port hatch and the starboard side of the cabin, respectively. Except for the upper canopy, bondline temperature were 5 to 20° F warmer than predictions. The upper canopy, however was approximately 80° F warmer than predicted. This large disagreement between actual and predicted values probably resulted from the cabin heat leak through the escape hatch structure. From this initial flight data, it appears that the expected attitude constraint (which would cause the TPS (thermal protection system) bondline glassy transition temperature to exceed -170° F) at high beta angle orbits might not exist for this area of the vehicle.

Mid fuselage bondline and structural temperatures were generally warmer than predicted with a wide range of deviations from predicted values (Table 2-X). Sill longeron temperatures were both warmer and colder than predicted. The mid-longeron exhibited the largest deviation from predicted values by approximately 40° F at the end of the nose-to-Sun attitude. The maximum temperature of 110° F was reached at the end of the top-to-Sun attitude on the port aft, and the starboard mid- and aft sill longerons. The 582 ring frame and 1307 bulkhead exhibited slower thermal response and were warmer than predicted.

Payload bay insulation surface measurements were warmer than predicted. In the tail-to-Sun and nose-to-Sun attitudes, temperatures dropped to a minimum of -140° F as compared to a predicted of -180° F and the temperature at X-670 near the centerline of the bay reached 260° F as compared to a predicted value of approximately 200° F in the top-to-Sun attitude.

The payload-retention fittings minimum temperatures ranged from -62° F on the DFI pallet longeron trunnion to -45° F on the DFI pallet keel trunnion. Flight temperatures were 10 to 15° F warmer than predicted in the tail-to-Sun and nose-to-Sun attitudes and up to 30° F warmer during the top-to-Sun phase. Review of the data also shows that for a given location, the latch, rail, and trunnion temperatures were generally within 10° F of each other.

Wing bondlines were generally warmer than predicted for hot and cold vehicle orientations with the largest deviation of 25° F occurring on the top of the wings in the top-to-Sun, nose-to-Sun and tail-to-Sun attitudes. The maximum temperature of 70° F was reached on the top of the wings near the end of the top-to-Sun attitude and the coldest temperature of -60° F was reached on the top of the wings at the end of the tail-to-Sun attitude.

The aft fuselage in general ran 20 to 35° F warmer than predicted for the tail-to-Sun and nose-to-Sun attitudes. It is expected that the up position of the elevon surfaces during the on-orbit phase contributed to the warm structural bias. During the tail-to-Sun attitude, the base heat shield warmed more rapidly than expected, 60° F as compared to 30° F. The aft fuselage did not reach equilibrium during the 80 hours of nose-to-Sun and was continuing to cool gradually. Table 2-XI shows representative temperatures at the end of each attitude and the rate of temperature change.

TABLE 2-IX.- ATTITUDE HOLD THERMAL RESPONSE
FORWARD FUSELAGE

Description	Tail-to-Sun		Nose-to-Sun		Top-to-Sun				
	Initial temperature, °F	Transient rate, °F/hr	Equilibrium temperature, °F	Initial temperature, °F	Transient rate, °F/hr	Equilibrium temperature, °F			
Upper cabin X530 bondline	30	-3.1	-44 ^a	12	N/A	18	26	2.6	72
Upper FRCS bondline	44	-3.7	-18	50	1.5	81	81	1.7	106
Lower X560 bondline	62	-1.1	36	50	-1.6	-4	-2	0.8	20
Starboard cabin bondline	12	-3.6	-62	-2	N/A	-4	4	2.3	31
Starboard FRCS bondline	54	-4.1	0	60	1.0	72	72	N/A	76
Port X480 hatch bondline	56	-2.6	4	40	-0.06	32	30	-1.3	18
Port cabin bondline	20	-3.2	-44	9	N/A	4	4	3.1	36

^a Gradually decreasing temperature

Comments: Tail-to-Sun structural transients (average) are a result of gravity gradient to tail-to-Sun attitude change
Nose-to-Sun structural transients (average) are a result of PTC to nose-to-Sun attitude change
Top-to-Sun structural transients (average) are a result of nose-to-Sun to top-to-Sun attitude change

TABLE 2-X.-- MID-FUSELAGE BONDLINE TEMPERATURES AT END OF MAJOR STS-3 MISSION PHASES

Measurement location	Bondline temperature, °F ^a				
	End of tail-to-Sun orbital rate	End of PTC	End of nose-to-Sun	End of top-to-Sun	End of day 7 PTC
Bottom fuselage bondline X=777, Y=-42 X=1030, Y=0 X=1214, Y=-2	5 (10) -3 (-22) 3 (-19)	30 (23) 17 (10) 24 (6)	-39 (-25) -29 (-46) -24 (-47)	-2 (3) 23 (14) 42 (21)	39 (27) 40 (26) 39 (23)
Wing bottom bondline P,X=1280, Y=-240 S,X=1280, Y=+240	-40 (-43) -42 (-44)	3 (-1) 3 (1)	-44 (-50) -42 (-48)	37 (37) 45 (43)	28 (19) 33 (20)
Wing top bondline P,X=1280, Y=-240 S,X=1280, Y=+240	-61 (-80) -63 (-53)	-3 (-11) 0 (2)	-47 (-54) -44 (-44)	48 (51) 48 (56)	23 (10) 28 (19)
Fuselage side bondline P,X=850, Y=-150 S,X=850, Y=+150 P,X=1215, Y=-105 S,X=1215, Y=+105	-32 (-45) -40 (-47) -34 (-42) -33 (-40)	13 (20) 18 (23) 18 (20) 17 (23)	-27 (-36) -29 (-34) -19 (-32) -17 (-31)	12 (9) 15 (11) 62 (55) 67 (56)	28 (26) 34 (29) 40 (32) 47 (36)
Payload bay door bondline P,X=613, Y=-50 S,X=613, Y=+50 P,X=929, Y=-50 S,X=929, Y=+50 P,X=1215, Y=-50 ^b S,X=1215, Y=+50	-40 (-42) -42 (-43) -43 (-40) -45 (-43) -22 (-42) -21 (-15)	-2 (-20) 3 (-19) 28 (-10) 26 (-8) 22 (-2) 20 (48)	-65 (-75) -55 (-75) -20 (-58) -19 (-58) -2 (-58) 2 (11)	-40 (-41) -36 (-47) -18 (-29) 0 (-33) 52 (-18) 60 (72)	-7 (-18) 10 (-19) 25 (-8) 33 (-10) 38 (-2) 45 (48)

^aPre-mission predictions are shown in parentheses.

^bFlight data/prediction agreement is poor because data from the wrong TMM node was used in the TMS.

TABLE 2-XI.- AFT-FUSELAGE BONDLINE/BASE HEAT SHIELD TEMPERATURES

Sensor description	Temperature, °F at end of indicated period ¹					
	End of PTC	End of tail-to-Sun	End of PTC	End of nose-to-Sun	End of top-to-Sun	End of PTC
Base heat shield structure	-9	55 (+0.9° F/hr)	26	-3 (0.7° F/hr)	26 (Small rise)	26 (Small decay)
Bondline bottom floor centerline	27	21 (+0.5° F/hr)	34 (+1.36° F/hr)	-7 Unable to get decay rate due to bottom Sun variations	19 (Small rise)	34 (Small decay)
Bondline - starboard sidewall	5	-20 (+2.0° F/hr)	20 (+0.4° F/hr)	-17 (Small decay)	29 (+0.6° F/hr)	21 (Small rise)
Bondline - port sidewall	15	-12 (Small rise)	24 (Small rise)	-14 (Small decay)	26 (+0.9° F/hr)	21 (Small rise)

¹Parentheses () indicate rate of change.

Table 2-XII summarizes the OMS pod bondline temperature histories for the three primary test attitudes. In general, the temperature profiles and transients were similar to predicted but warmer as shown in the table. The pod housing generally reached cyclic equilibrium in approximately 10 hours after an attitude change. However, the RCS housing required 15 to 20 hours, except for PTC which took 10 hours. The minimum temperature of -93° F occurred on the port pod at the end of the tail-to-Sun attitude. By analysis, the coldest location is expected to have reached approximately -113° F, however, no instrumentation was located at that point to provide an exact temperature measurement. The maximum temperature, 125° F, occurred on the port RCS housing aft surface at the end of the tail-to-Sun attitude.

STS-3 was the first mission to return data relative to the main landing gear (MLG) response to cold attitudes that is required to define an expected time/attitude constraint to prevent exceeding -35° F on the strut actuators and the hydraulic dump valve. The minimum temperature experienced (-17° F) occurred on the brake lines during the nose-to-Sun attitude. The predicted temperature was -33° F. A break in the attitude which resulted in a bottom sun attitude lasted between 2 and 3 hours and caused a warming in the MLG area that required approximately 20 hours to recover to the temperature levels prior to the break.

A summary of pertinent temperatures for each payload bay door closure test is provided in Table 2-VII. All payload bay door closures were successful except for the tail-to-Sun orbital rate attitude test. The vehicle was oriented to a top-to-Sun attitude for approximately 15 minutes followed by a PTC mode. After approximately 42 minutes of PTC, the doors were successfully closed and latched. It is interesting to note that only the seals (as compared to motors and latches) experienced large temperature increases as a result of the attitude changes.

Prior to STS-1, it was determined analytically that a constraint in a tail-to-Sun attitude would be necessary to prevent overheating of OMS engine feed lines and vernier RCS feedlines. At the end of the tail-to-Sun orbital rate attitude during STS-3, the port OMS engine inlet line temperature reached 110° F and was estimated to reach 120° F equilibrium if the attitude had been continued. Postflight data evaluation indicates the potential of approaching 200° F (145° F engine start limit) for a high beta angle mission.

The ability of the GPC-software thermostat-control logic for operation of the hydraulic systems circulation pumps and the adequacy of this mode to maintain acceptable temperatures was demonstrated.

As a result of the left outboard elevon being in a down position and the other three being in an up position during the long nose-to-Sun attitude, the left outboard elevon actuator was the only actuator that was subjected to as cold an environment as expected. The up elevons resulted in a warmer radiation environment for the other three actuators. Circulation pump duty cycles were 2 to 3 percent except for system 3 which was running at approximately 15 percent in the nose-to-Sun attitude. The software set points were changed from 0° F "on" and 20° F "off" to -10° F "on" and 0° F "off" (at approximately 85:05:00 G.m.t.) to reduce the duty cycle. The actuator cooled when the system 3 line limits were reduced. This points out the close thermal coupling that exists between the actuator and the hydraulic fluid temperature. The immediate effect of the return line temperature limit change was to reduce the pump duty cycle, however, as the actuator cooled to the new return line temperatures the duty cycle again increased to approximately 15 percent until the elevon actuator heaters were activated to obtain thermal response data at 86:13:41 G.m.t. It is expected that if the other three elevons had not been in an up position, similar duty cycles would have been experienced by the active systems for those actuators.

Forward RCS compartment heaters did not reach cyclic equilibrium during the cold tail-to-Sun orbital rate attitude as a result of a heater configuration change. Internal forward RCS compartment temperatures were within 10° F of predictions. The bulk RCS propellant temperature reached a minimum of 63° F (60° F limit). The port fuel relief line exceeded its 20° F limit when it reached 14° F at 82:17:00 G.m.t. This line is normally empty and did not pose a problem. Efforts to establish a more realistic limit are underway.

The forward thruster heater duty cycles were consistently higher (up to 2 1/2 times) than those predicted during the nose-to-Sun and top-to-Sun attitudes except for the forward-firing thrusters. During the tail-to-Sun attitude, the thruster heater duty cycles were above predicted values except for the left thruster heater (F1L). The duty cycles ranged from 53 to 87 percent.

In general, OMS pod heater duty cycles ran slightly lower than the predicted duty cycles except the following:

- a. Port and starboard keel web heaters during the tail-to-Sun attitude (actual 45 percent vs predicted 40 percent).
- b. OMS engine cover heaters during the nose-to-sun attitude (actual 55 percent vs predicted 49 percent [port] and 41 percent [starboard]) and during the top-to-Sun attitude (actual 32 percent [port] vs predicted 0 percent).
- c. RCS drain panel during the nose-to-Sun attitude (actual 39 percent [port] and 28 percent [starboard] vs predicted 25 percent).
- d. Starboard GSE service panel heaters during the nose-to-Sun attitude (actual 38 percent vs predicted 31 percent).

The nose-to-Sun attitude was the coldest of the three primary attitudes during STS-3. Only the keel web heaters operated during the tail-to-Sun attitude and only the port OME cover heaters were activated during the top-to-Sun attitude.

The aft port and starboard minimum RCS bulk propellant temperatures were 70° F and 72° F, respectively, for the oxidizer. This occurred at the end of the top-to-Sun attitude just prior to the keel web heaters being activated during the PTC period, thereby preventing a violation of the 70° F entry propellant temperature limit.

The aft RCS thruster heater duty cycles were in general 1 1/2 to 2 times higher than predicted. In general, the primary thruster heater duty cycles were highest (in the high 30 percent range) during the nose-to-Sun attitude. As predicted, the aft-firing thruster (L1A, L3A, R1A, R3A) heaters did not operate during the tail-to-Sun attitude.

The APU system fuel feed lines, service lines, and seal cavity drain line duty cycles were approximately the same or lower than predicted. However, the lubrication oil line heaters were slightly higher than predicted (25 percent vs. 17 percent). No APU fuel tank heater operations occurred until the last day when the APU 3 fuel tank heater turned "on". The monitoring measurement for the thermostat apparently has a 6° F bias from the thermostat turn "on" limit since the temperature was reading 61° F and the thermostat set point is 55° F. At 89:09:16 G.m.t., an IMU alignment occurred apparently collapsing the stratification which was being setup in the fluid as a result of the heating of the tank. The maneuver to top-to-Sun at 89:12:18 G.m.t. also appears to have caused another stratification collapse, however, the data also indicate the heater turned off during this same time frame.

The aft primary RCS thruster, L2U, 100-second continuous firing thermal soakback test (FTO 312-02) was initiated at approximately 85:14:35 G.m.t. As had been determined prior to the flight, the firing resulted in several steady-state and pulse-mode firings rather than a 100-second continuous firing.

The aft vernier RCS thruster pulse-mode firing test (FTO 312-03) was initiated at approximately 86:14:41 G.m.t. The oxidizer valve and the fuel injector maximum temperatures in the first pulse mode were 142° F and 176° F, respectively, as compared to predicted values of 180° F and 480° F. Comparison of temperatures from the second pulse-mode series is not available to make a flight versus prediction comparison at this time.

The star tracker hot environment thermal response test (FTO 312-04) was initiated at 87:01:08 G.m.t. in the top-to-Sun attitude. The star tracker was turned "off" for 12 hours as planned and turned "on" at 87:13:00 G.m.t. as planned for 12 hours. A self-test following the 12-hour "on" period showed no unacceptable errors. All temperature were within acceptable limits.

The FES (flash evaporator system) feed water line heater thermal response test (FTO 312-05) was accomplished during the nose-to-Sun attitude. Approximately 8.5 hours of no flash evaporator operation was provided to allow operation of the FES line heaters without any water flow influence. The required data were obtained for determination of the heater design adequacy.

The maximum entry bondline temperatures of 228° F and 190° F occurred on the starboard OMS pod and the bottom midfuselage, respectively. The maximum forward RCS compartment bondline temperature of 185° F occurred on the starboard side. Initial temperatures at entry interface were 15, 23, and 55° F, respectively. These maximum entry temperatures are comparable to those experienced on STS-2.

Hydraulic temperatures were well above entry interface requirements. No circulation pump operation occurred post-landing based on the fact that none of the bondline temperatures defined for post-landing circulation pump operation exceeded the defined 240° F limit.

2.8.2 Thermal Protection Subsystem and Leading Edge Structural Subsystem

2.8.2.1. Nose Cap.- The nose-cap radiometer temperature measurements of the RCC (reinforced-carbon-carbon) shell were suspiciously low for the STS-1 and 2 flights. Since the nose-cap radiometer data are considered questionable, a peak nose-cap RCC temperature of approximately 2200° F has been estimated using the STS-3 flight data and postflight predictions. The lack of available radiometer data has hindered the verification of nose-cap entry aerodynamic heating; however, comparisons of the nose-cap interior structural DFI temperature data indicate good thermal performance of the insulation and attachments. Figure 2-10 presents a comparison of measured STS-1, 2, and 3 flight temperatures for the lower side forward support link bolt which attaches the nose-cap shell and T-seal. The lower centerline aluminum bulkhead temperature response for the first three STS flights is shown in figure 2-11. These data, along with other data, indicate no degradation in the nose-cap performance for these initial flights. Table 2-XIII presents a summary of the STS-3 nose-cap DFI temperature measurements. The exterior surface of the nose-cap was examined visually for evidence of cracks, chips, etc., and no evidence of any of those conditions was found except for two small RTV (room temperature vulcanizing) 566 contamination spots previously noted after the STS-2 mission.

Several HRSI (high temperature reusable surface insulation) interface panel tiles were removed from the lower centerline 416 carrier plate to inspect slumped tiles aft of this interface panel. Inspection of the interface cavity indicated some scorching/burning of the filler bar at the base of the slumped tiles. The interface panel thermal barrier was brittle at the tile OML (outer moldline) similar to the results obtained from the nose-cap thermal certification test.

TABLE 2-XIII.- COMPARISON OF STS-1, -2, and -3 LEADING EDGE
STRUCTURAL SYSTEM NOSE CAP DATA

Sensor no. VO9T(P)XXXXA	Location component	STS-1 maximum temperature, °F	STS-2 maximum temperature, °F	STS-3 maximum temperature, °F
9941	Lower centerline support link	a	725	720
9942	Lower centerline bulkhead	185	178	162
9943	Lower centerline HRSI/RCC TB	a	a	a
9944	120 centerline bulkhead	170	170	155
9945	120 centerline support link	700	710	640
9946	120 centerline HRSI/RCC TB	a	a	a
9947	Upper centerline support link	350	360	353
9948	Upper centerline HRSI/RCC TB	a	a	a
9949	Bulkhead HRSI	1750 ^b	2000	1980
9950	Bulkhead HRSI bondline	180	172	160
9951	Bulkhead HRSI	1150 ^b	1190	1180
9952	Lower centerline support link	610	610	570
9953	Deleted			
9954	Deleted			
9955	Stagnation RCC	1360 ^b	1450 ^a	1450 ^a
9956	Lower centerline RCC	a	a	a
9957	Lower tang. RCC	1075 ^b	1200 ^a	1190 ^a
9958	Upper centerline RCC	700 ^b	860 ^a	860 ^a

^aSensor inoperative or data questionable.

^bMaximum value at start of data.

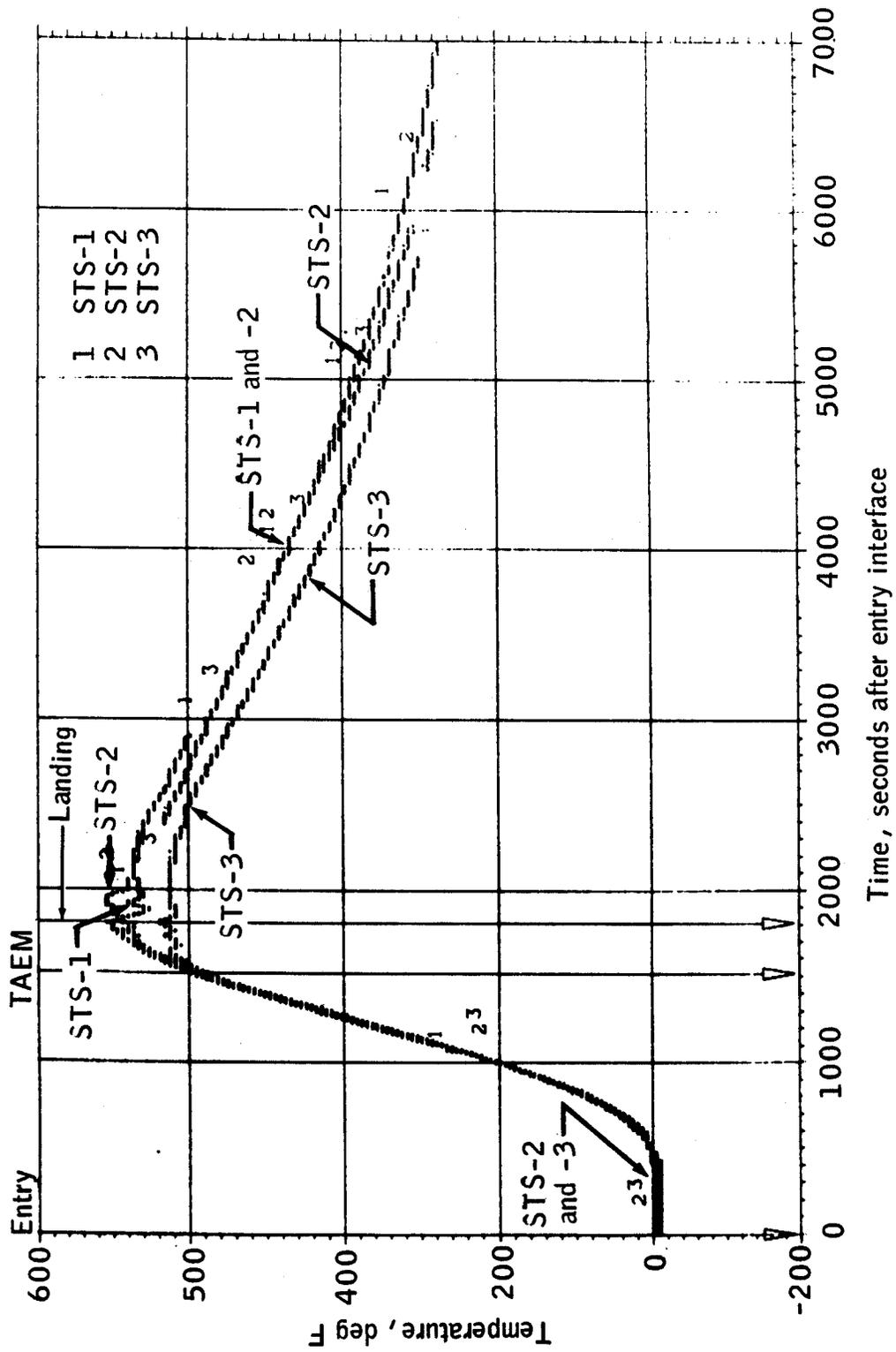


Figure 2-10.- Nose cap side support link temperatures.

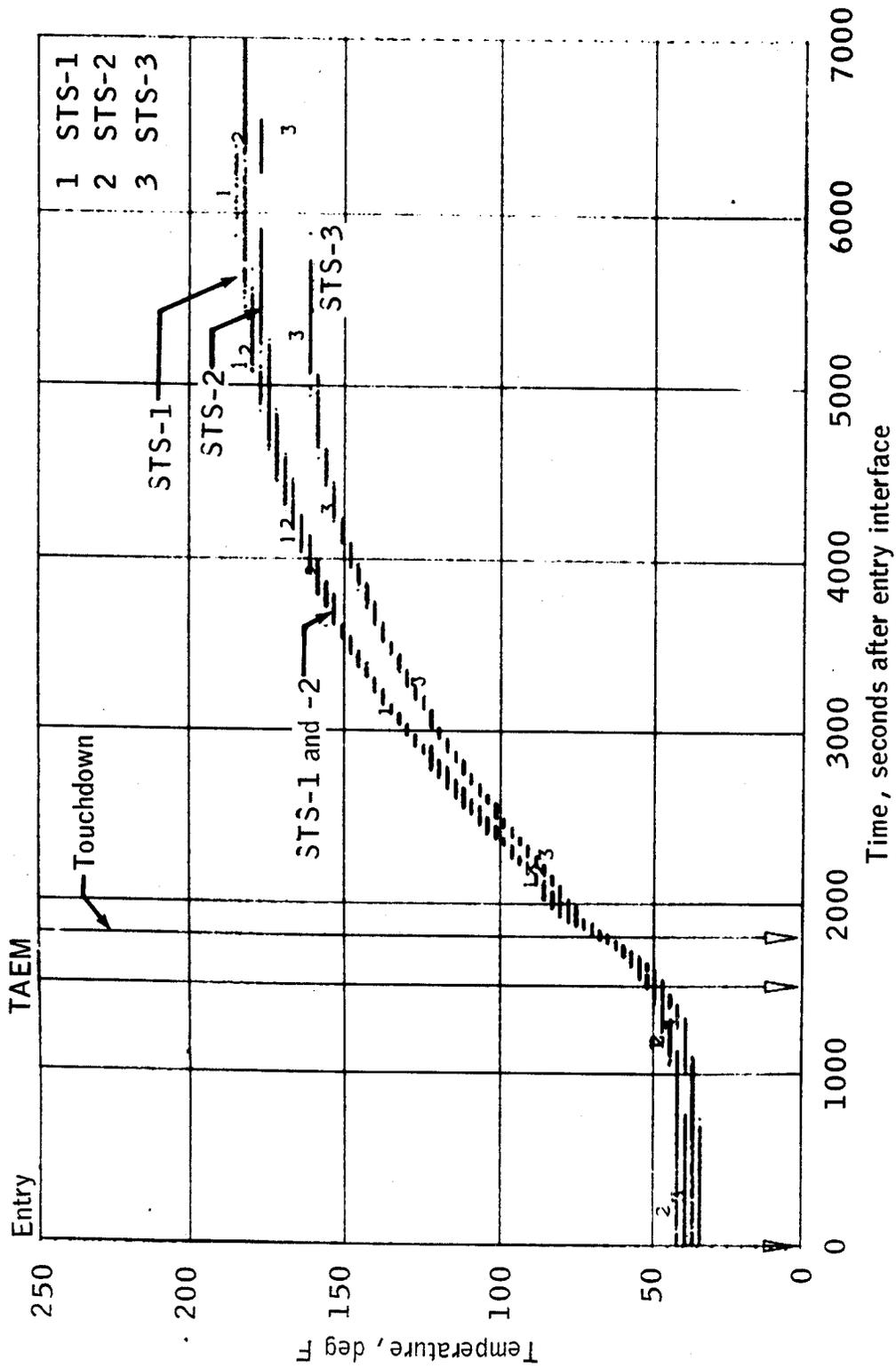


Figure 2-11.- Nose cap lower bulkhead temperatures.

2.8.2.2 Wing Leading Edge.- Figures 2-12 and 2-13 present time histories of the RCC IML (inner moldline) temperatures for the STS-2 and 3 flights from the 55- and 80-percent semi-span leading-edge radiometers. Shock interaction effects in the 55-percent area produced higher localized heating as shown in these figures. Figure 2-14 compares the measured flight temperature response of lower panel 16 (80 percent) Inconel clevis. As can be seen from these figures, no degradation in the thermal performance has occurred from the STS-1, 2, and 3 flights. Table 2-XIV presents a summary of the wing leading edge DFI temperature measurements for the STS-3 flight. The wing leading panels were all examined externally for evidence of anomalies, chips, cracks, etc., from the STS-3 mission. Three small coating chips on panels 3 left-hand, 11 right-hand, and 18 right-hand were repaired with the launch site coating repair. Areas of discoloration as a result of gas flow-through at the RCC/HRSI interface panels were significantly reduced from the STS-2 flight results.

The post-STS-2 design modification (addition of large diameter (1/2") end-panel stuffers, i.e., cigarette stuffer, to both panel ends) to the lower access panel 10 left-hand was successful. No evidence of flow through and aluminum tube heating as experienced on STS-2 was found. In general, while some flow-through is still occurring, there appears to be a significant reduction in flow-through where the redesigned lower access panels were installed for the STS-3 flight. Prior to the STS-3 flight, eight left-hand panels (7 through 14) and nine right-hand panels (6 through 11, 16 through 18) were modified to reflect the redesigned gap filler.

The STS-3 flight data indicate that the thermal performance of the LESS (leading edge structural subsystem) nose cap and wing leading edge was outstanding with no degradation in thermal performance for the three flights. In addition, a detailed inspection indicates that a significant improvement in reducing flow-through at the RCC/interface panel area was achieved with the modified thermal barrier design.

2.8.2.3 RCC-Forward External Tank Attachment.- The corners of the aft RCC plate at the bearing plate hub interface had significant chipping with the carbon substrate exposed on one of the chipped corners. Evidence of flow and heating near these chipped corners on the IML of the aft plate was evident. The attachment holes show evidence of wear after three flights. These plates will be replaced for the STS-4 flight.

No DFI data measurements exist for the external tank attachment plates; however, an examination of available DFI near this location would indicate a peak RCC temperature of 1600 to 1800° F.

2.8.2.4 Windows (Thermal Panes).- The six windshield thermal panes and the two overhead thermal panes were examined for evidence of the surface haze that appeared during the STS-1 and 2 flights. Windows 2 and 3 appeared to have the same amount of haze as observed after STS-2. Window 4 had more haze than noted after STS-2. Windows 1, 5, and 6 had a minimal amount of hazing. Impact deposits were observed on windows 3, 4, and 5, but these impacts did not scratch or flaw the window surface. The overhead windows appeared clear.

The contamination (hazing and impact deposit) could not be removed with the standard solvent. Disposition of this condition for STS-4 was determined as no constraint.

Several RSI (reusable surface insulation) overhang tiles on the right-hand forward, and middle windows and left-hand forward window had to be removed and replaced as a result of impact damage.

2.8.2.5 Elevon/Elevon Ablators.- The ablaters and elevon structure were examined for STS-3 heating effects. In general, the ablator appears to be comparable to the STS-2 results. Both the right-hand and left-hand inboard ablaters look better than STS-2; however, the outboard rib ablaters appear to have greater surface recession (increased heating) than

TABLE 2-XIV.- COMPARISON OF STS-1, STS-2, AND STS-3 WING LEADING
EDGE SUBSYSTEM FLIGHT DATA

VO9T(P)XXXXA	Wing leading edge		STS-1	STS-2	STS-3
	Location	Component	maximum temperature, °F	maximum temperature, °F	maximum temperature, °F
9901	Panel 4	Lower clevis	a	a	a
9902	Panel 4	Lower spar bracket	170	196	175
9903	Panel 4	Lower IF panel	162	187	175
9904	Panel 4	Upper clevis	425	445	a
9905	Panel 4	Upper spar bracket	81	104	88
9906	Panel 4	Insulation surface	1260 ^b	1300	1285
9907	Panel 4	Insulation bondline	122	137	125
9908		Deleted			
9909	Panel 4	Lower RCC temperature	1625 ^b	1890	a
9910	Panel 9	Lower clevis	915	875	a
9911	Panel 9	Lower spar bracket	295	305	320
9912	Panel 9	HRSI facing RCC	1220	1205	1135
9913	Panel 9	Lower IF panel	300	a	290
9914	Panel 9	Upper clevis	--	--	--
9915	Panel 9	Upper spar bracket	267	250	220
9916	Panel 9	Upper HRSI facing RCC	1300 ^b	1650	1640
9917	Panel 9	Upper IF panel	270	a	317
9918	Panel 9	Insulation surface	1975 ^b	a	1810
9919	Panel 9	Insulation in-depth	1165 ^b	1155	1125
9920	Panel 9	Insulation in-depth	889	900	835
9921	Panel 9	Insulation bondline	400	411	383
9922	Panel 9	Insulation surface	1840 ^b	2025	1975
9923	Panel 9	Insulation surface	1675 ^b	1750	1750
9924	Panel 9	Cavity pressure	--	--	--
9925		Deleted			
9926	Panel 9	Lower RCC temperature	2450 ^b	2470	2460
9927	Panel 9	Upper RCC temperature	1390 ^b	1920	1920
9928	Panel 16	Lower clevis	580	570	550
9929	Panel 16	Lower bracket	257	270	265
9930	Panel 16	Upper clevis	425	435	430
9931	Panel 16	Insulation surface	2400 ^a	2400 ^a	2400 ^a
9932	Panel 16	Insulation bondline	215	197	192
9933		Deleted			
9934	Panel 16	Lower RCC temperature	1890 ^b	2170	2100
9935	Panel 22	Lower clevis	565	570	510
9936	Panel 22	Lower bracket	169	177	175
9937	Panel 22	Insulation surface	1110 ^b	1150	1140
9938	Panel 22	Insulation bondline	122	122	127
9939	Panel 22	Cavity pressure	--	--	--
9940	Panel 22	Lower RCC temperature	1750 ^b	1850	1840

^aSensor inoperative or data questionable.

^bMaximum value at start of available data.

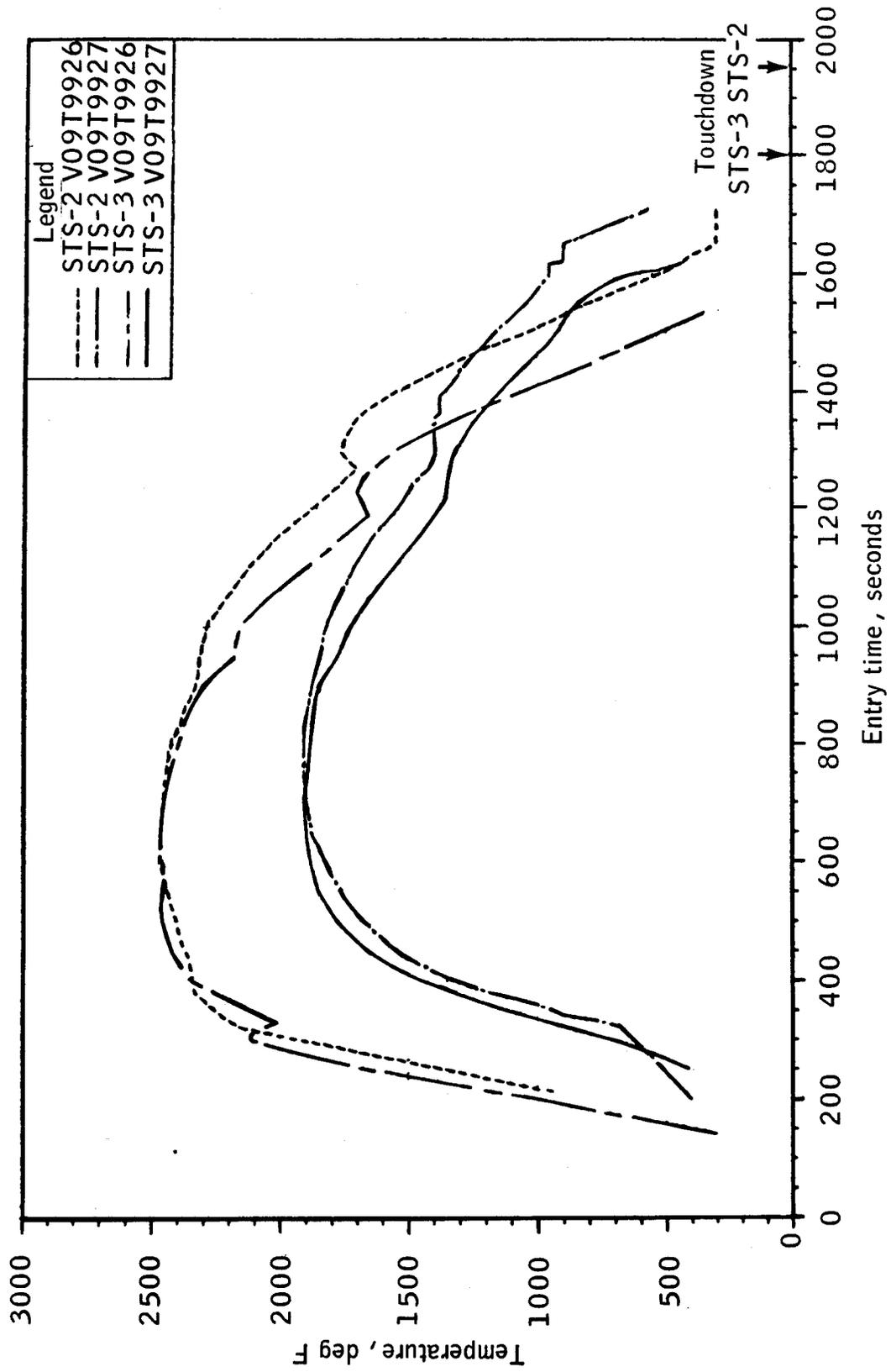


Figure 2-12.- STS-2 and STS-3 Comparison of LESS and RCC panel 9 temperatures.

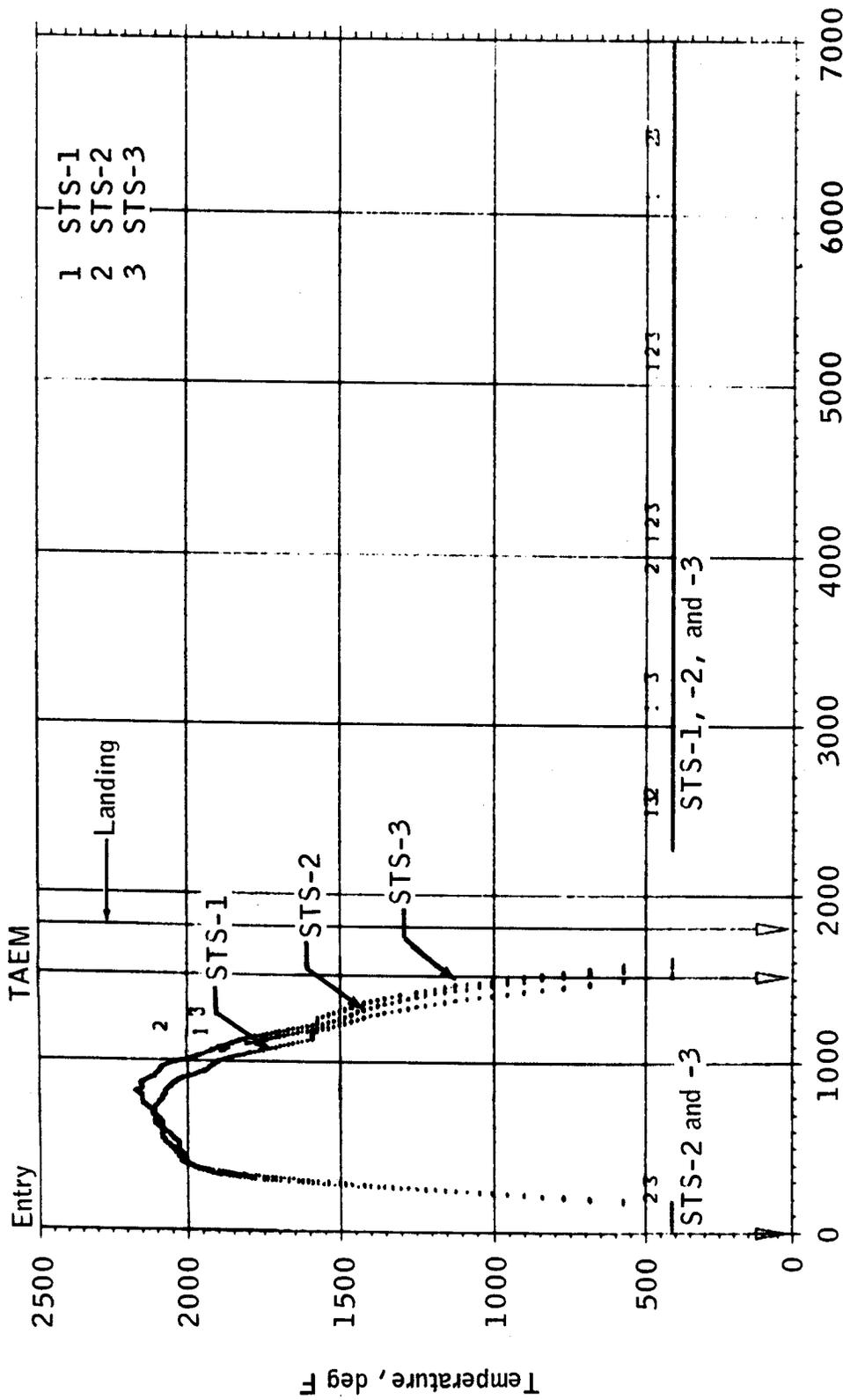


Figure 2-13.- Wing leading edge 80-percent RCC temperature .

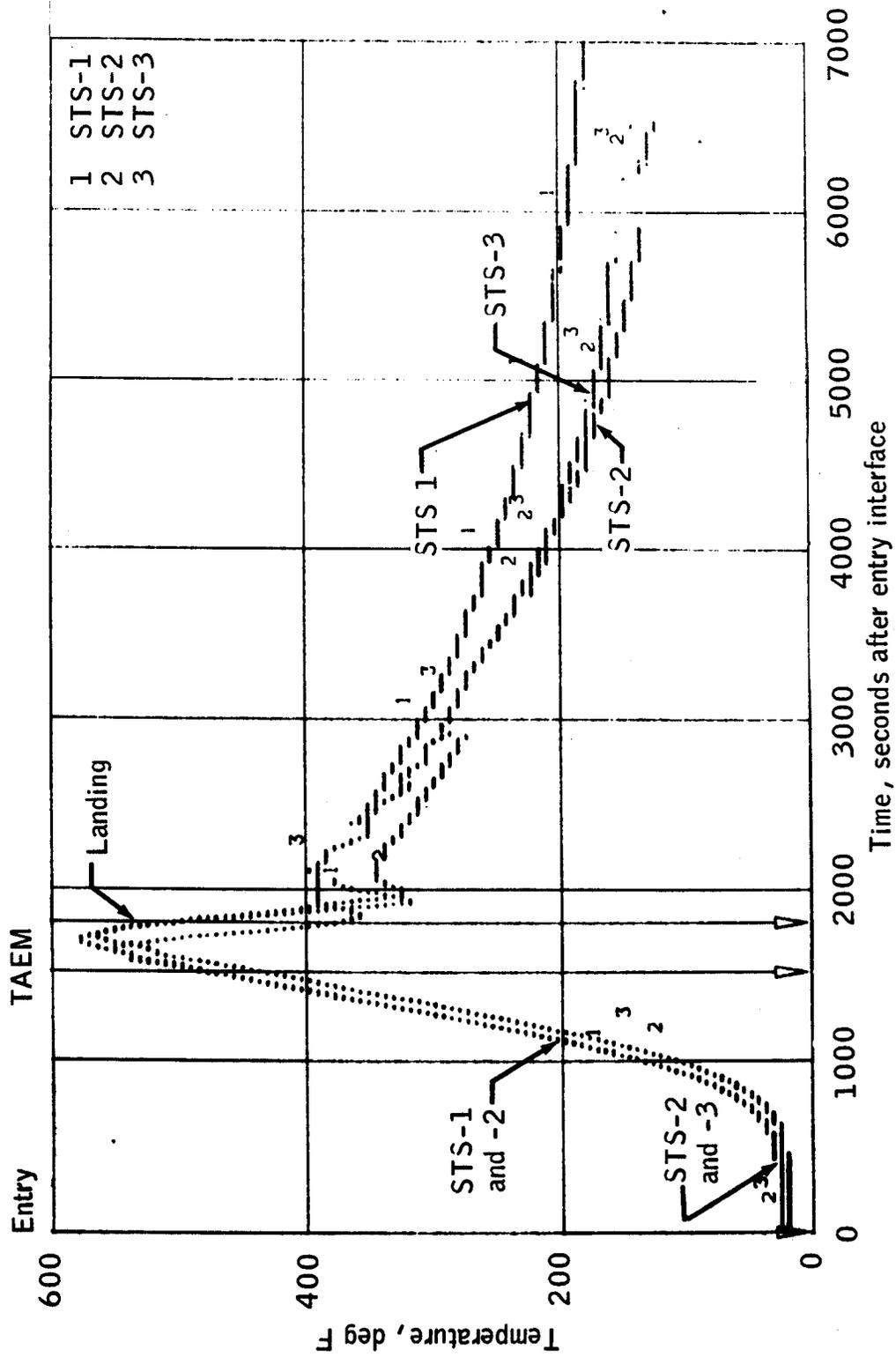


Figure 2-14. - Wing leading edge 80-percent lower attachment clevis temperature.

experienced on the STS-2 entry. Both the right-hand and left-hand outboard ablators had surface recession depths of 0.4 to 0.6 in. along the miter joint and additional longitudinal (approximately 10 in.) erosion starting six in. aft of the hub seal and 1 to 2 in. above the miter joint. Temperature labels on the left-hand outboard aluminum carrier plate IML indicated a peak temperature of 210° F.

The STS-3 ablator hardware (4 ribs, 2 spills, 4 tips) have been removed for detailed sectioning to aid in evaluating STS-3 heating/thermal performance evaluations.

2.8.2.6 Thermal Protection Reusable Surface Insulation: Good DFI data were obtained during entry and these data showed in general that STS-3 was slightly cooler than STS-2. Figures 2-15 and 2-16 compare the STS-2 and STS-3 peak measured surface and structural temperatures, respectively, during entry. Figures 2-17, 2-18, and 2-19 show some typical comparisons of flight data with the heating predictions at three locations on the lower fuselage.

Shortly after the Orbiter reached orbit, the crew reported several tiles missing from the upper forward fuselage. In addition, a review of the launch pad camera films and an inspection of the pad after launch resulted in identifying the loss of several tiles from the upper body flap region. (Section 7.0, flight test problem report 16), but detailed analyses showed no safety of flight issue. The results of the detailed post-flight inspections of the vehicle are as follows:

- a. Minor damage to the surface of the RSI (reusable surface insulation) tiles was again found, but was less than that noted on STS-2. Inspection showed 193 surface dings, gouges and chips of the tile coating had occurred. Surface impact damage to the lower body flap tiles was considerably less than that found on STS-2 and STS-3.
- b. Surface contamination continues to occur but to a lesser extent, and contamination was significantly reduced from STS-1 and 2 on the body flap. Figure 2-20 shows the lower body flap surface after STS-3. The darker tiles were replaced prior to STS-3 and show very minor surface contamination as compared to the adjacent light gray tiles which were contaminated during the STS-1 and STS-2 launches.
- c. Continued excessive tile-to-tile gap heating occurred but to a lesser degree than during STS-1 and STS-2 (614 locations during STS-1, 360 locations during STS-2 and 219 locations during STS-3). The most significant local heating occurred between a tile just aft of the nose cap and the nose landing gear door forward thermal barrier. A thermal barrier repair appeared to have allowed some flow intrusion between the tile and the thermal barrier, resulting in local slumping of the tile; however, no overtemperature of the structure or door seals occurred.
- d. The FRSI (felt reusable surface insulation) on the forward region of the OMS pods and on the aft section of the payload bay doors again experienced higher heating. This resulted in scorching of the FRSI coating. Measured surface temperatures in both locations were in excess of 900° F with the multi-mission re-use limit for FRSI being 700° F.
- e. Two locations on the right OMS pod experienced minor overtemperatures of the graphite epoxy structure (figure 2-21). The damage at location A resulted from excessive shrinkage of the FRSI at the FRSI/LRSI (low temperature reusable surface insulation) interface. This resulted in a 0.5-inch diameter delamination of the graphite epoxy facesheet. The damage at location B was the result of the loss of a small corner of the LRSI tile. The graphite epoxy damage was similar to location A. Both locations were repaired.

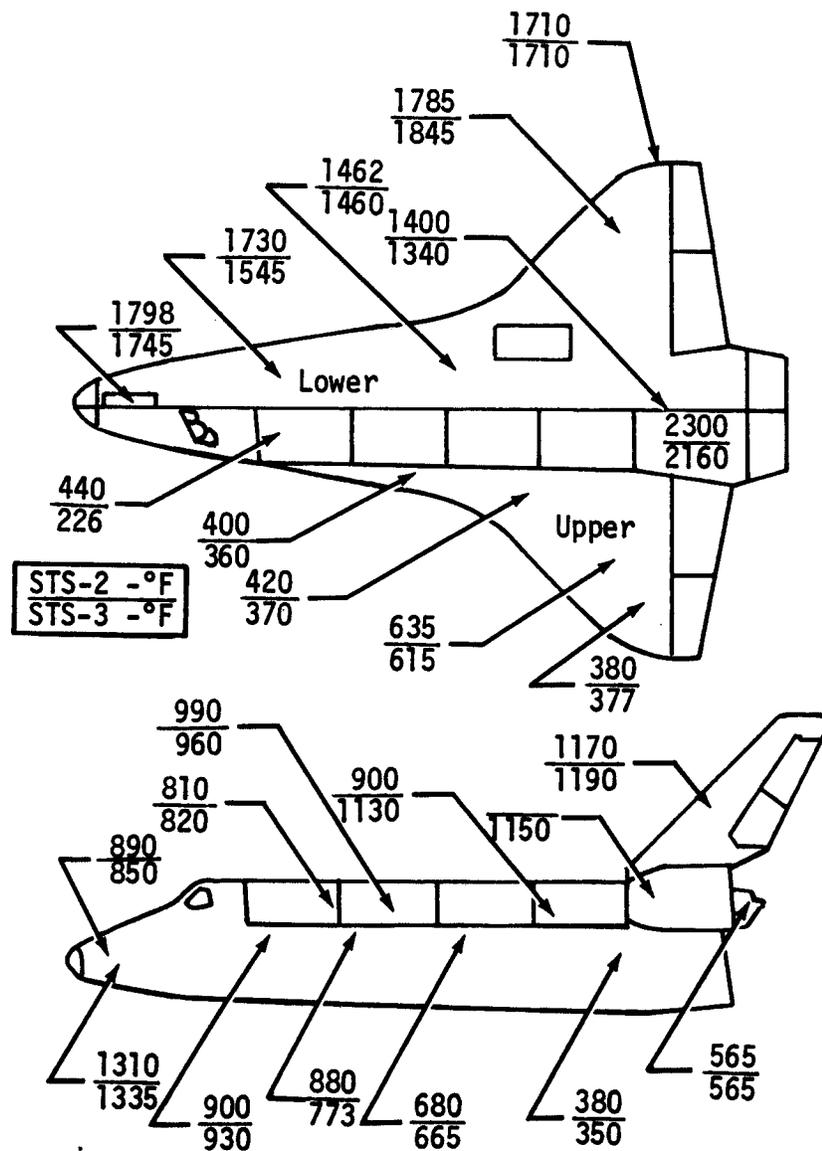


Figure 2-15.- STS-2 and STS-3 peak entry surface temperature comparison.

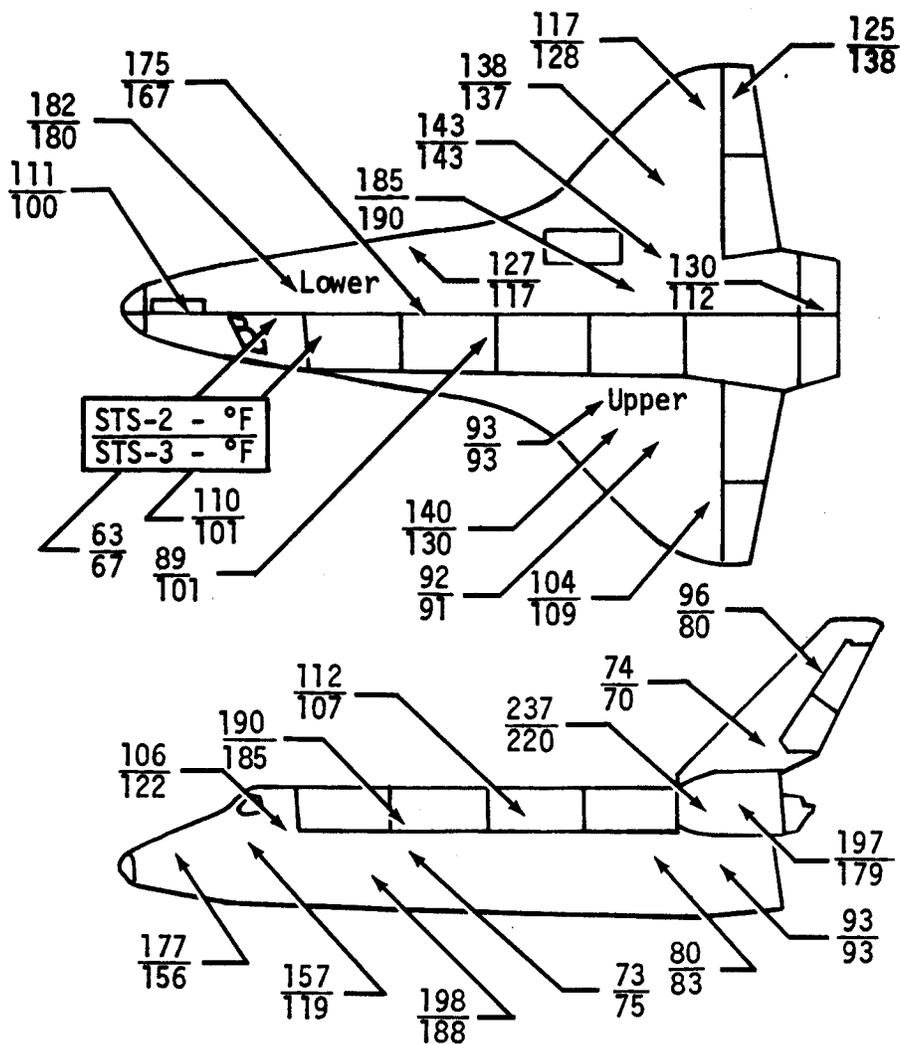
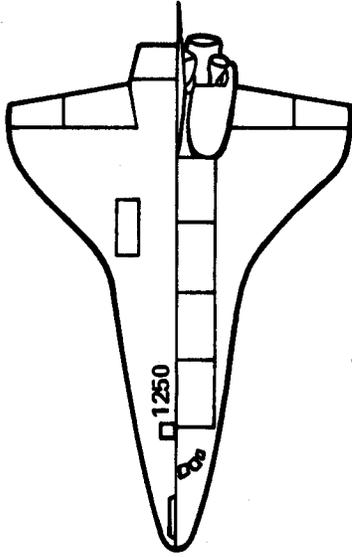
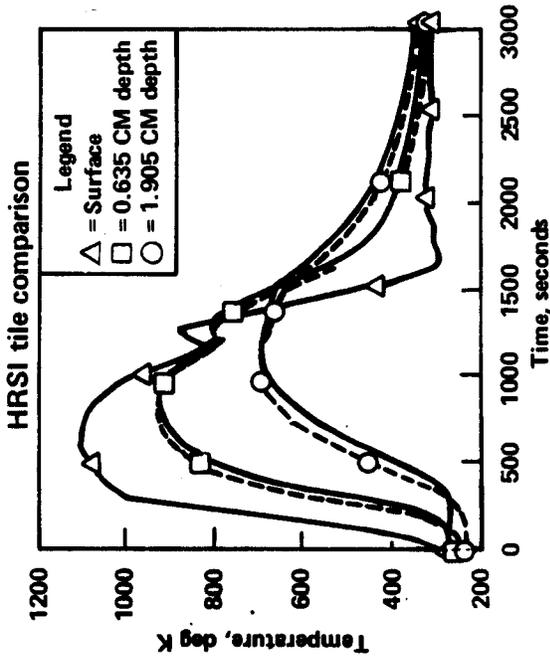


Figure 2-16.- STS-2 and STS-3 peak entry structure temperature comparison.



— Analysis
- - - Data

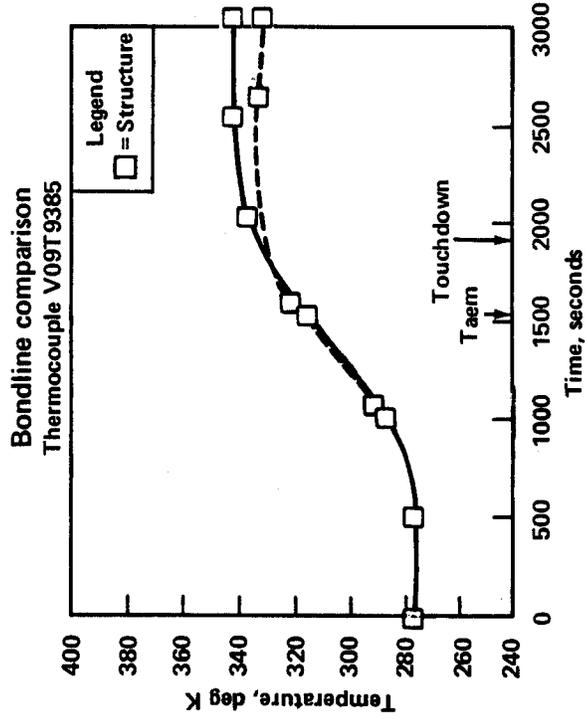
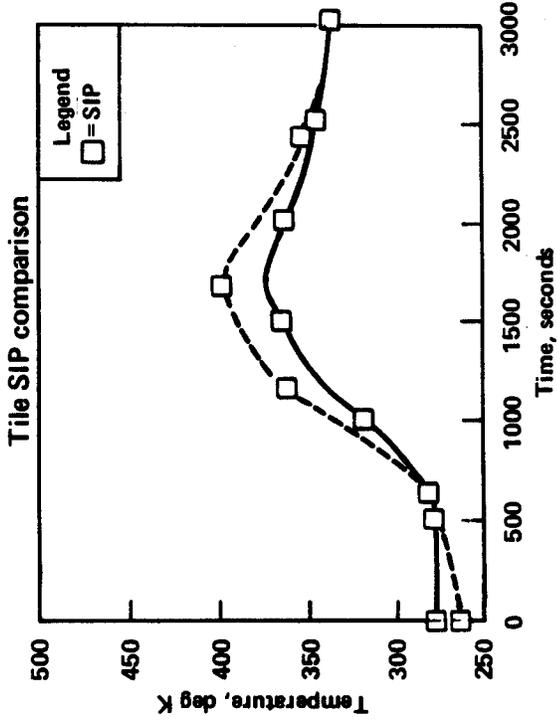


Figure 2-17.- Comparison of analysis and actual temperature data at station 1250.

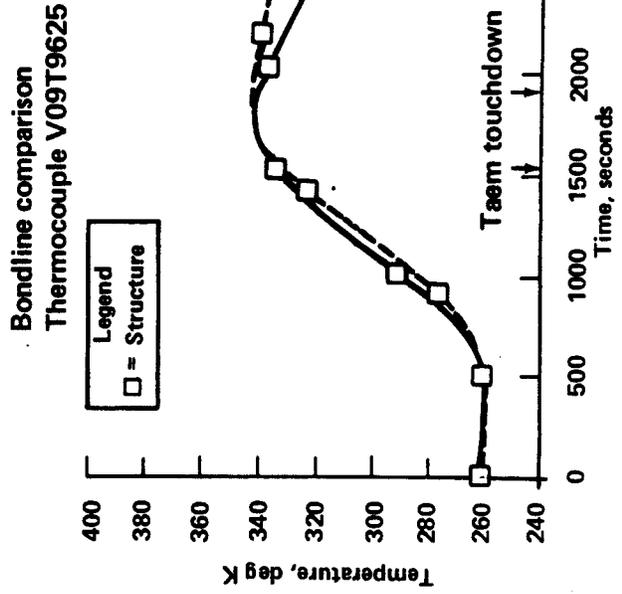
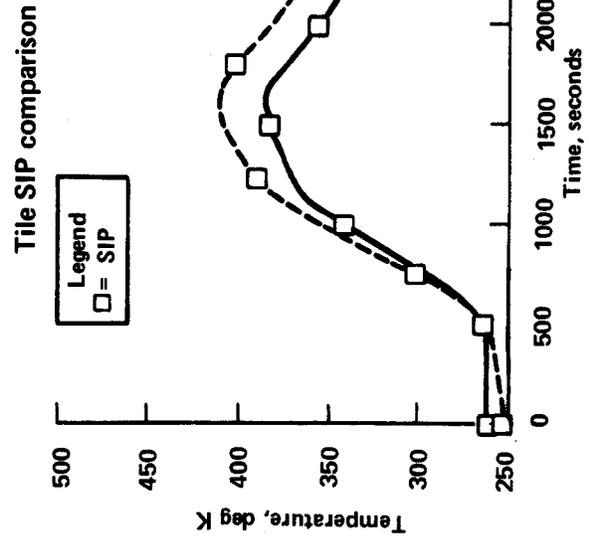
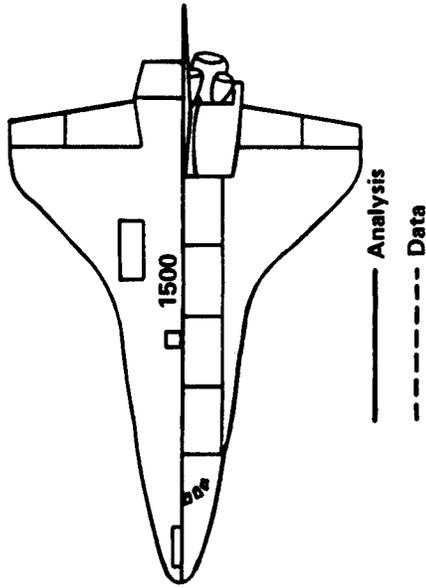
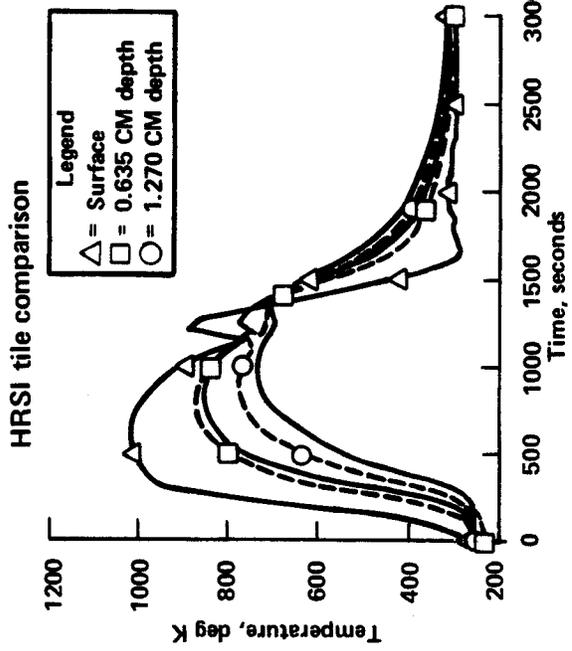


Figure 2-18.- Comparison of analysis and actual temperature data at station 1500.

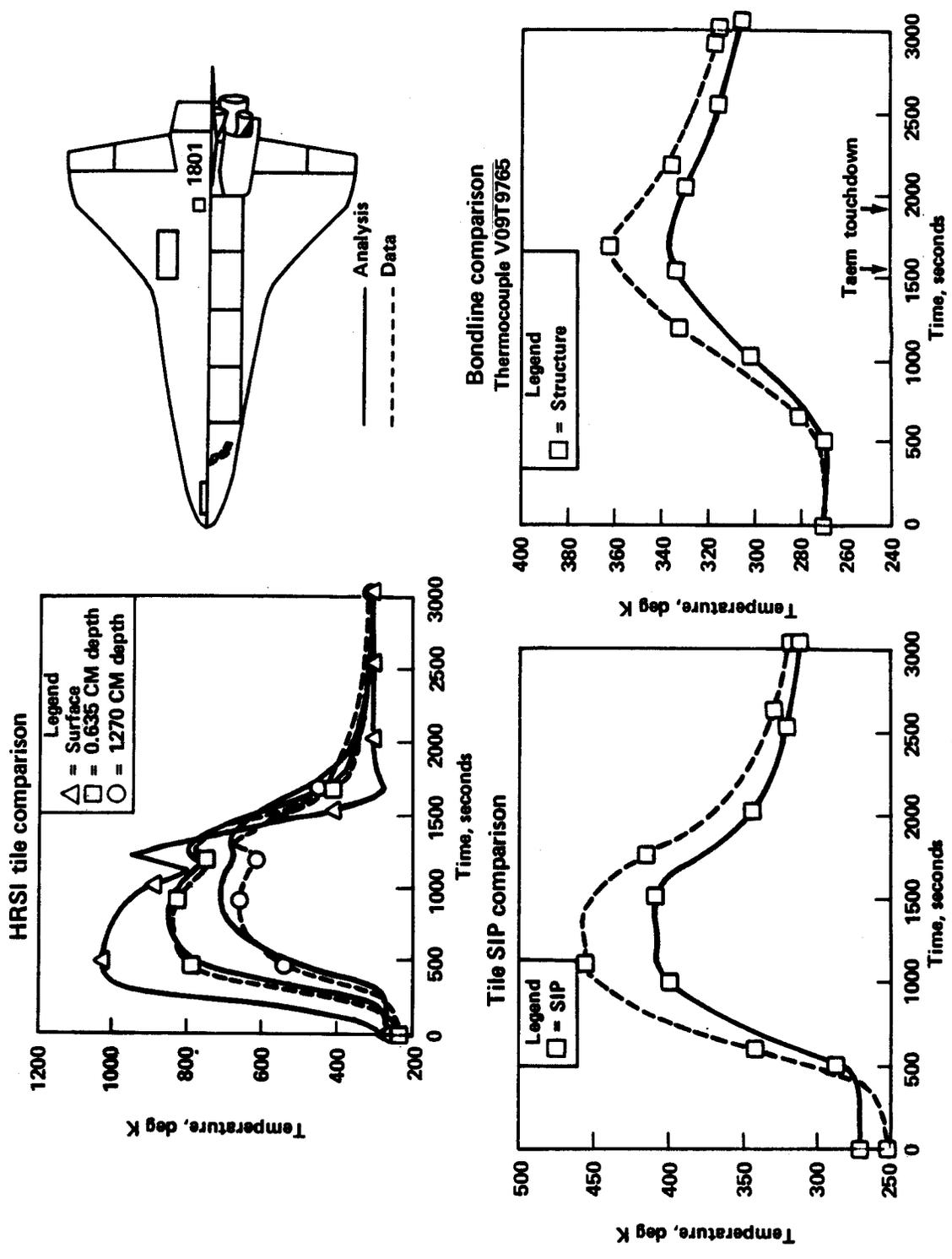


Figure 2-19.- Comparison of analysis and actual temperature data at station 1801.



Figure 2-20.- Damage to the lower surface of body flap after STS-3.

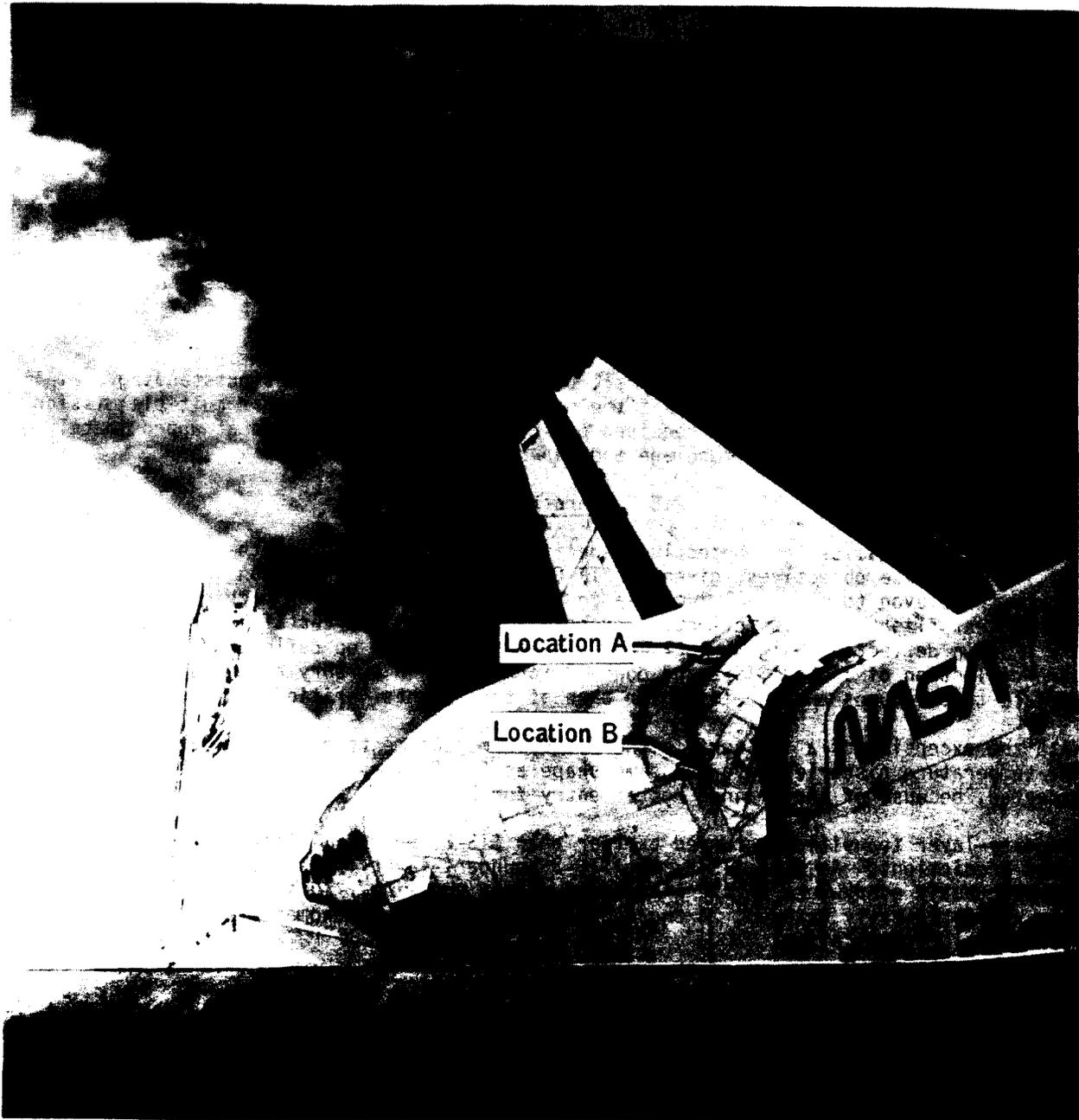


Figure 2-21.- Over temperature locations on right OMS pod.

f. Twenty-one full tiles and 16 pieces of tiles were found missing on the upper forward fuselage (figures 2-22 and 2-23). Very little evidence of entry heating was noted in the tile cavities (minor scorching of the exposed SIP (strain isolator pad) edges in two locations). No structural damage occurred in the regions of the lost tiles. In each case, the tile or piece of tile was undensified and was a tile requiring a bond to the filler bar for positive structural attachment margin. The specific causes are under investigation (Section 7.0, flight test problem report 16). All of the tiles on the upper forward fuselage are being densified prior to STS-4.

g. Sixteen undensified tiles were lost from the upper body flap. Post-flight inspection showed significant scorching of the SIP pads and filler bars in two of the four areas of the tile loss (figure 2-24). This overheating occurred during ascent and detailed inspection of the body flap showed no evidence of structural damage. The specific causes of this condition are being investigated (Section 7.0, flight test problem report 16). All of the tiles on the upper body flap are being densified prior to STS-4.

In spite of these conditions, the overall performance of the RSI was outstanding. Minimal turnaround refurbishment will correct the problems and tile damage for multiple mission usage. Total tile replacements required prior to STS-4 are estimated at approximately 150 plus 53 on the upper forward fuselage and upper body flap.

2.8.3 Aerothermodynamics

The aerothermodynamics test objectives specified for STS-3 entry of the Orbiter were satisfied. These objectives, given in FTR 07VV024, included an elevon deflection plan to expose the elevon to a gradual increase in heating environment for each of the orbital flight test flights. This deflection program assisted in the certification environment for a 5° - down deflected design value. The program involved an up deflection of 1° for STS-1, down deflection of 1° for STS-2 and down 3° for STS-3. A cursory analysis of the measured surface temperatures on the elevon for each of the flights verified the increase expected.

With the exception of a few surface thermocouple measurements around the Orbiter, most of the temperature histories have the same shape as STS-2 entry and a little higher values, as expected, because of the planned hotter entry for STS-3.

Boundary-layer transition occurred earlier than STS-2, but still later than wind-tunnel-based predictions. As with STS-2, the STS-3 boundary-layer transition occurred almost simultaneously everywhere on the windward fuselage and wing. This behavior suggests the flow was artificially tripped. A thorough analysis of this phenomenon will be done for the operational certification.

2.9 ENTRY AERODYNAMICS

2.9.1 Performance and Trim

Comparison of the predicted lift-to-drag (L/D) ratio with the post flight STS-3 L/D ratio is presented in figure 2-25. Likewise, the comparison of the pitching moment (Cm) predictions, including the ± uncertainties data, with the post-flight STS-3 Cm data is presented in figure 2-26. Overall, the L/D comparison appears to be in good agreement with the predicted data except for the subsonic region. This discrepancy, as seen on STS-1 and 2 (higher L/D than predicted) is primarily due to the subsonic drag increment which is not as yet reflected in the Aerodynamic Data Book predictions. Longitudinally, the pitching moment data (Cm) illustrated the same type of results as seen on STS-1 and STS-2 in the viscous



Figure 2-22.- Tile losses on forward fuselage.



Figure 2-23.- Detailed photograph of tile damage on right forward fuselage .



Figure 2-24.- Tile damage on upper body flap.

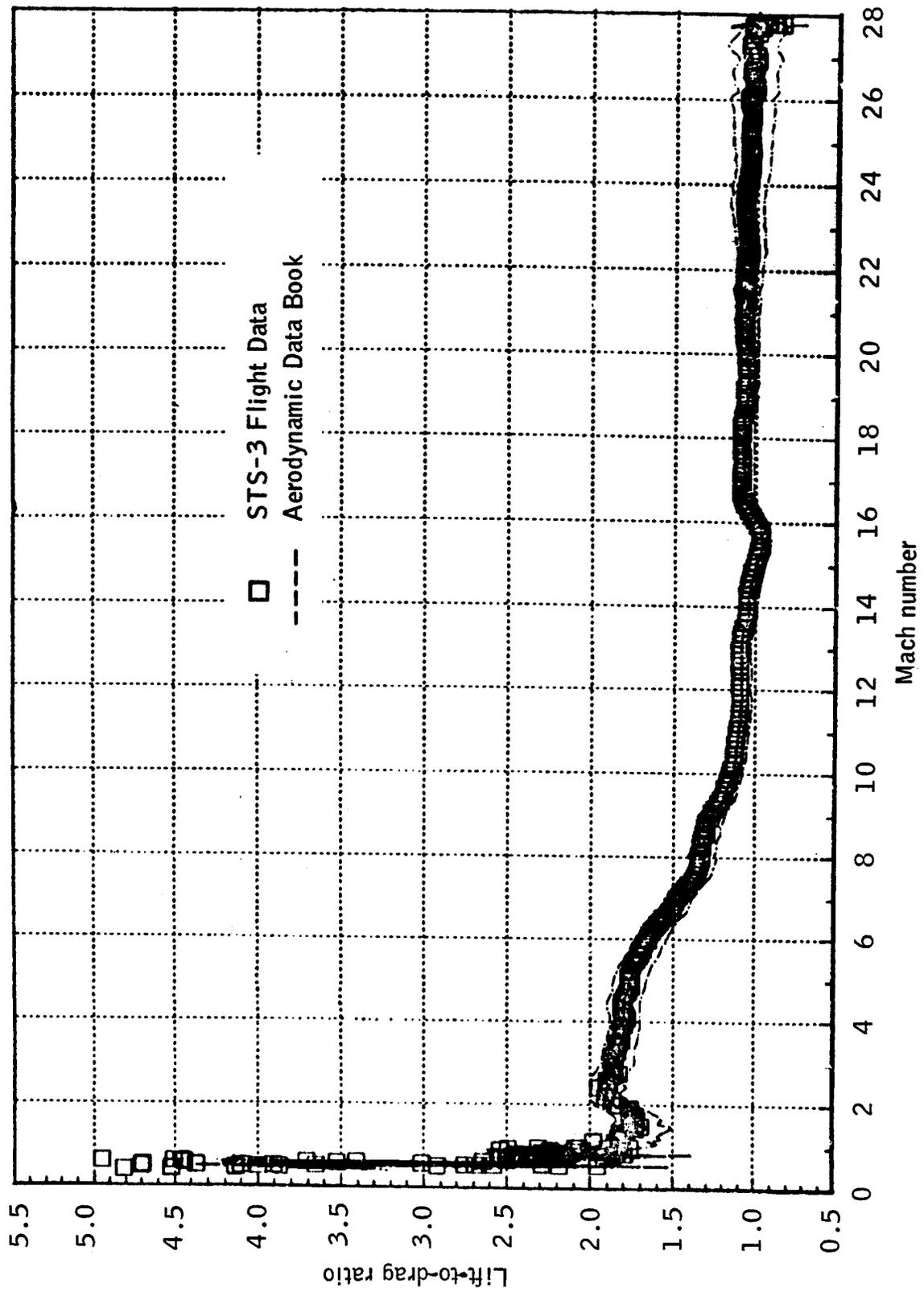


Figure 2-25.- Lift-to-drag ratio .

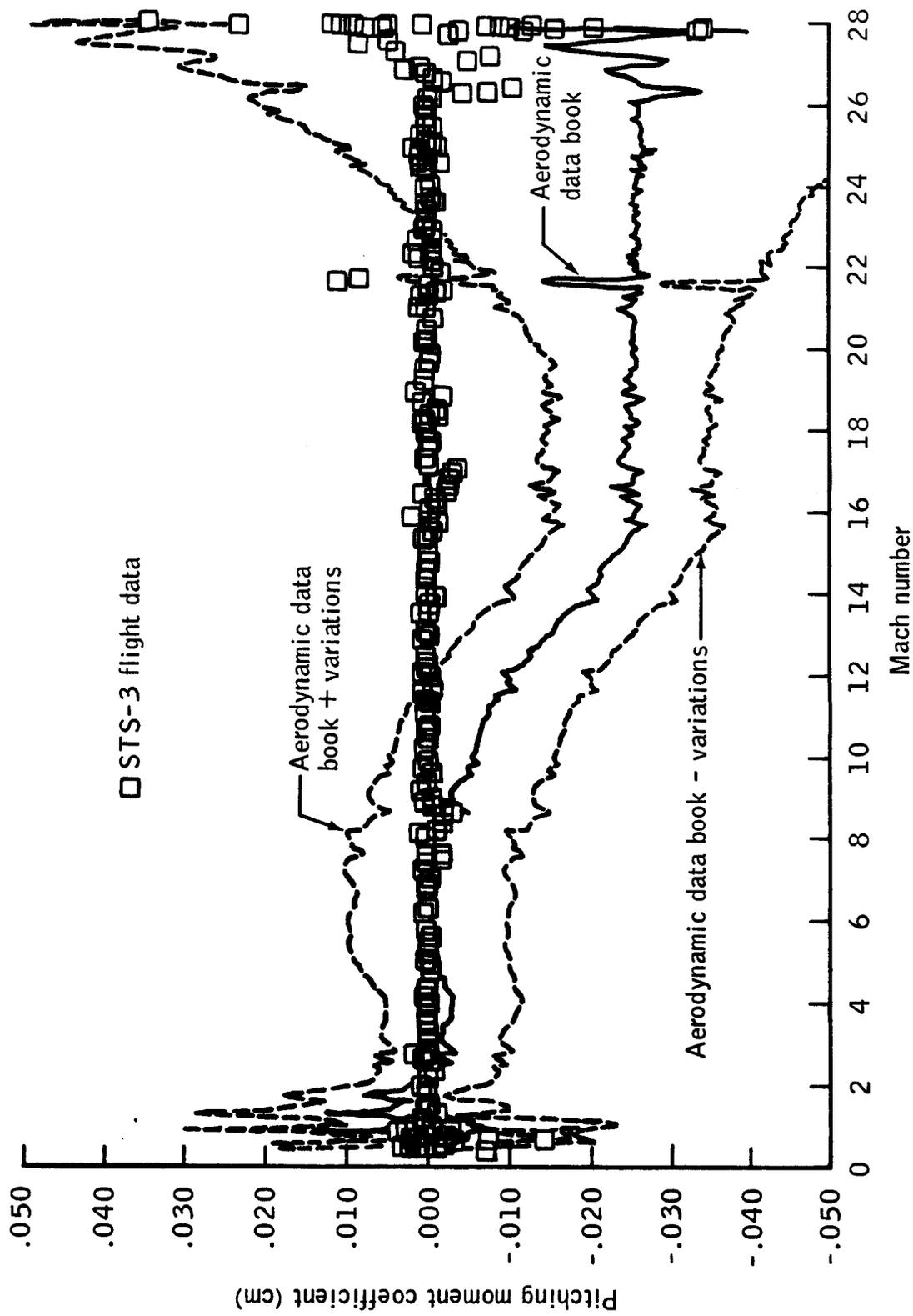


Figure 2-26.- Comparison of pitching moment predictions.

region (Mach 28 to 8) and good agreements thereafter except for a small variance between Mach 2 and 1. These aerodynamic differences more or less confirm the STS-2 Assessment Aerodynamic Package that is to update the Aerodynamic Data Book.

2.9.2 Flight Test Requirements Accomplished

Six out of seven aerodynamic maneuvers were performed on STS-3. One stick maneuver in the pitch axis was performed at a velocity of 21,500 ft/sec. Five lateral directional maneuvers were also performed at the following conditions:

$$q = 22 \text{ psf}$$

$$V = 21,500 \text{ ft/sec}$$

$$V = 18,000 \text{ ft/sec}$$

$$V = 14,000 \text{ ft/sec}$$

$$V = 8,400 \text{ ft/sec}$$

The maneuver at Mach 4 was deleted before entry, because of a potential conflict with a bank reversal. Further, the PTI 1 was not resequenced at the $V = 8,400$ ft/sec condition, and the software automatically sequenced to PTI 2. Thus, the maneuver designed for Mach 4 was executed at 8,400 ft/sec. All of the maneuvers performed will be analyzed to extract stability and control derivatives. Impacts because of lost test maneuvers will be assessed.

2.9.3 Stability and Control

Analysis of stability and control data from PTI's, ASI's, and bank reversals has just begun because of processing problem data. DFRF analyses show that PTI results support STS-2 findings. Preliminary results from the 0.25 Hz motion between Mach 2 and 1 indicate that small rudder deflections provide different responses than expected from wind-tunnel data and the larger deflections obtained from STS-2 PTI's. Further work will be done to identify the aerodynamic model that correctly defines this phenomenon.

2.9.4 Mass Properties Comparison Based on Deorbit Maneuver Data

The STS-3 Y-axis center-of-gravity can be determined assuming the OMS-4 firing is constant and non-rotating over 40 seconds. This is done by setting the sum of the moments produced by the 2 OMS engines to zero and solving for the moment arm which allows for equilibrium. This analysis requires OMS engine gimbal locations, gimbal angles, either thrust or acceleration values, and either an estimated Xcg or Zcg. Although extremely sensitive to any errors in these measured inputs, the final aerodynamic results strongly suggest a Y-axis center of gravity as 1-in. to the left (port) side.

Figure 2-27 shows a plot of the thrust vector (averaged over the entire firing) and the acceleration vector (averaged at 40 seconds) extrapolated from the OMS gimbal points. The longitudinal center of gravity shown was computed by linearly interpolating the predicted mass properties to 40 seconds into the firing. A 1-in. uncertainty is given on the X-axis and a 0.5-in. is given on the Z-axis. Due to the uncertainties involved in calculating the thrust and acceleration vectors, no conclusions should be made as to the accuracy of the predicted center of gravity.

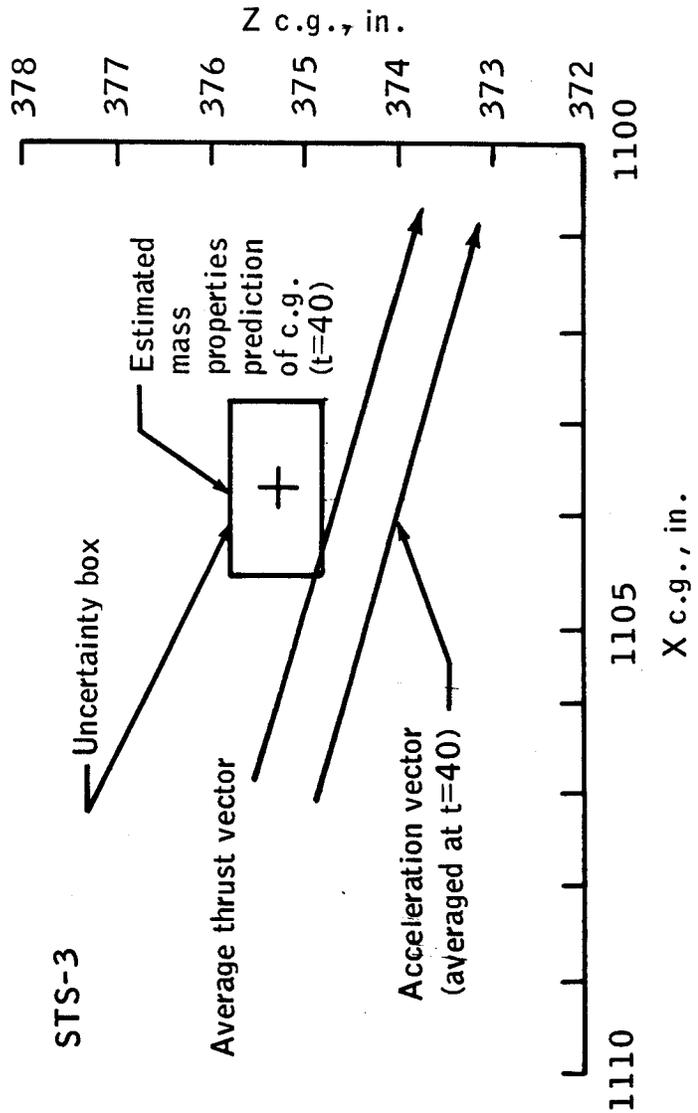


Figure 2-27.- Thrust vector and c.g. determination.

The weight of the Orbiter at the deorbit maneuver was estimated for STS-3 by dividing OMS thrust by the acceleration. At 40 seconds into the firing, the weight is 12063.7 lb/0.05689 g's = 212036 lbs, with an uncertainty of 2000 lb due to a 1 percent thrust error. This value is 1325 lb less than the linear interpolated value estimated from mass properties.

2.9.5 Quarter Hertz Oscillation

Six cycles of undamped low-frequency oscillations occurred in the Mach 2 to Mach 1 region. These oscillations produced a roll rate of approximately 2.5 deg/sec peak-to-peak and were better defined than those observed on STS-1 and STS-2. However, the PTI planned at Mach 1.6 on STS-4 has been replaced with the structural PTI, and therefore it is very probable that data from PTI's on STS-5 and STS-6 will be required before the aerodynamic data can be accurately defined in this Mach region.

2.9.6 Aerodynamic Coefficient Instrument Package

The ACIP (Aerodynamic Coefficient Instrument Package) operated for all commanded data takes on STS-3.

The data takes for which ACIP acquired data are:

- 1) Launch - 81:15:59:30 to 81:16:15:00 G.m.t.
- 2) Gravity gradient - 81:23:46:40 to 81:23:48:41 G.m.t.
- 3) Vernier RCS plume impingement - 87:21:27:00 to 87:21:39:00 G.m.t.
- 4) Deorbit maneuver to wheels stop

Preliminary analysis of the ACIP data quality show the rate gyros to be operating as expected. With respect to the IMU and as on STS-1, biases are present in the linear and angular accelerometers. High-frequency vibrations are also present suggesting structural vibrations.

2.10 REMOTE MANIPULATOR SYSTEM

The RMS (remote manipulator system) handling characteristics were very good, especially regarding the deployment and berthing of the PDP (plasma dynamics package) payload, although it was almost 450 lb lighter than the planned IECM (induced environment contamination monitor) payload. Experience with grappling a grapple fixture at a known location (i.e., the PDP when latched) showed that the changes from preflight were very small (RMS joint angles were within 1° and the position of end effector was within 1.5 in.). This provides great confidence in the digital displays over a wide range of Orbiter thermal conditions.

Other real-time non-planned activities included an inspection of the nose tiles with the RMS elbow television camera, an impromptu manual search during the PDP electron beam tests, and an evaluation of the RMS light. All of these activities were completed with good results.

2.10.1 FTO Accomplishment

The intent of all of the scheduled FTO's was accomplished to some degree; however, the dynamic data effectiveness was reduced by the loss of the DAC (data acquisition cameras) cameras. The FTO accomplishments are discussed in the following paragraphs.

The FTO 351-01 (Hot Case Arm Evaluation) was delayed until STS-4, however, since the Orbiter was in the top-to-Sun orientation for 28 hours, some data were obtained with the RMS in the cradled configuration. The temperature rise rate during this period was lower than expected and the arm temperatures were still increasing towards the predicted temperatures when the top-to-Sun attitude was terminated.

Although FTO 351-02 (Cold Case Evaluation) was conducted in several separated parts, the data obtained were very good and the intent of the FTO was accomplished.

A calibration error was found in the computer temperature look-up table and the RMS temperatures ranged from 5 to 25° F warmer than indicated during the mission. This error was also present on STS-2. However, the temperatures have been converted to actual temperatures for both missions. The RMS temperatures stabilized as expected for the various Orbiter conditions. The RMS heaters were able to maintain the temperatures within the specified operating range of the RMS, although the duty cycle was about 50 percent instead of the 20 percent predicted.

The initial step 1 of FTO 352-01 (Payload Deployment and Berthing Performance) required deployment and berthing of the IECM. This could not be accomplished because of the failure of the starboard-aft CCTV camera because of the concern that a subsequent failure of the port aft CCTV camera after deployment would have prevented berthing, thus causing the IECM to be jettisoned. Evaluation of the STS-3 flight data has allowed modification of this flight rule for STS-4, and the activity is scheduled for STS-4.

The final step required deployment and berthing of the PDP. This step was accomplished 3 times with no difficulty (each time took less than 10 minutes). The portion of the FTO is considered complete and no further tests are required.

The FTO 352-02 (End Effector Grapple/Rigidization Performance) required grapple/rigidization in both the manual and auto modes and these activities were accomplished. The wrist CCTV camera was lost prior to the first grapple, making the task more difficult and the FTO data more valuable. This experience showed that a grapple could be executed without the wrist camera, provided that adequate alternate views are available. Another plus in this test was a demonstration of complete capture and rigidization with a wrist roll error of 9° (10° is the limit and 1° to 2° is normal with a wrist camera). Power transfer from the RMS to the PDP was accomplished with no difficulty via the special purpose electrical grapple fixture. This FTO is complete.

The FTO 352-04 (RMS Control System Evaluation (Loaded)) required the IECM to be used as a load (weight - 830 lb), but the CCTV problem caused the PDP to be used as a substitute (weight - 383 lb). The DAC cameras were also required and these did not operate. This FTO is not complete.

The FTO 352-05 (RMS/RCS Interactions) tests were also scheduled using the IECM, but the PDP was substituted. The DAC cameras were to be used here also and have even more importance since the CCTV camera brackets are known to respond significantly to the PRCS pulses. Data from these tests will be processed by the NRCC photogrammetric group, but little useful information is expected. This FTO is not complete.

All four steps of this FTO 355-01 (RMS Back-up Mode Control Performance (Unloaded)) were successfully accomplished.

The third step required actuation of three joints in turn for 5 seconds each to perform a dynamic validation run. The DAC cameras and the CCTV were to collect the only data for this FTO. Since the CCTV is satisfactory, the switch profile will be run in a simulator and compared to the position when the arm joint data were restored. The CCTV data from the final step will also be used to support this step.

The final required cradling the arm in back-up mode and this was accomplished with no difficulty by both crewmen.

The FTO 355-02 (RMS Software Stop Performance (Unloaded)) was successfully accomplished. The elbow-pitch joint stopped in 1.49 deg or 1.2 deg before the hardstop was reached (coarse speed). The shoulder yaw joint stopped in 0.17 degree or 2.43 degree before the hardstop was reached (vernier mode). This FTO is complete.

The FTO 355-04 (Unloaded Arm Response to PRCS Activity) was accomplished with good data results except for the DAC camera. Again, since the CCTV camera is subject to vibrations from the PRCS and is unreliable for dynamics, this FTO will be planned for STS-4 with the understanding that DAC camera operation is mandatory.

2.10.2 RMS Problems

The DAC Camera problem is discussed in section 2.5.3 (Section 7.0, flight test problem report 18).

Three times (during the arm cold soak tests) when the arm was powered up, a wrist-pitch encoder check alarm was encountered. (Section 7.0, flight test problem report 22). The alarms were reset by the crew and normal activities were continued. Following a power-on sequence (to investigate the encoder alarms), a wrist yaw communications scanner bite was encountered. This bite signal remained on until the arm was deselected. (Section 7.0, flight test problem report 30). The arm was powered up 3 times after this occurrence and the bite signal never reappeared.

The RMS ready for latch indicator switch failed to transfer at the end of second day operations when the arm was stowed. (Section 7.0, flight test problem report 12). The indication was received on later operations.

3.0 ENVIRONMENTS ANALYSIS

3.1 ACOUSTIC ENVIRONMENT

Acoustic data were reviewed for compliance to the design specification criteria and the previous flight tests. Data were reviewed for the time period from main engine ignition through the supersonic phase of ascent. The measured data were found to be within specification and are compatible with the measured data from the previous two flights.

The plots in figures 3-1, 3-2, and 3-3 are representative of the acoustic level that the aft fuselage section of the vehicle has experienced through the first three flights. A new measurement (V08Y9774) was active on the tip of the vertical stabilizer on STS-3. The level measured is within the predicted level and should remain consistent from flight to flight. The aft heat shield measurement (V08Y9686) and the inboard elevon actuator measurement (V08Y9733) gave evidence that the acoustic level remains constant from flight to flight for the lift-off phase.

Review of the remaining microphones located both internally and externally to the Orbiter are in agreement with the levels measured on the previous two flights.

The analysis of the acoustic environment for the payload bay is based on six microphones: four located in the payload bay and one each located forward aft on the payload bay doors exteriors. Figure 3-4 is a spatial average of the measurements from the three flights that have been completed. The levels are well below the predicted criteria in all frequency ranges. The microphone (V08Y9405) located on bulkhead X-582 is still indicating a slightly higher level of noise than the other three internal microphones. Further study of this condition is underway to determine the cause. One other internal microphone (V08Y9219) shows a loud whistle sound at a frequency of 315 Hz during the transonic phase of ascent and this is attributed to a vent door in close proximity to the measurement.

No acoustic data were obtained during entry because of a recorder failure.

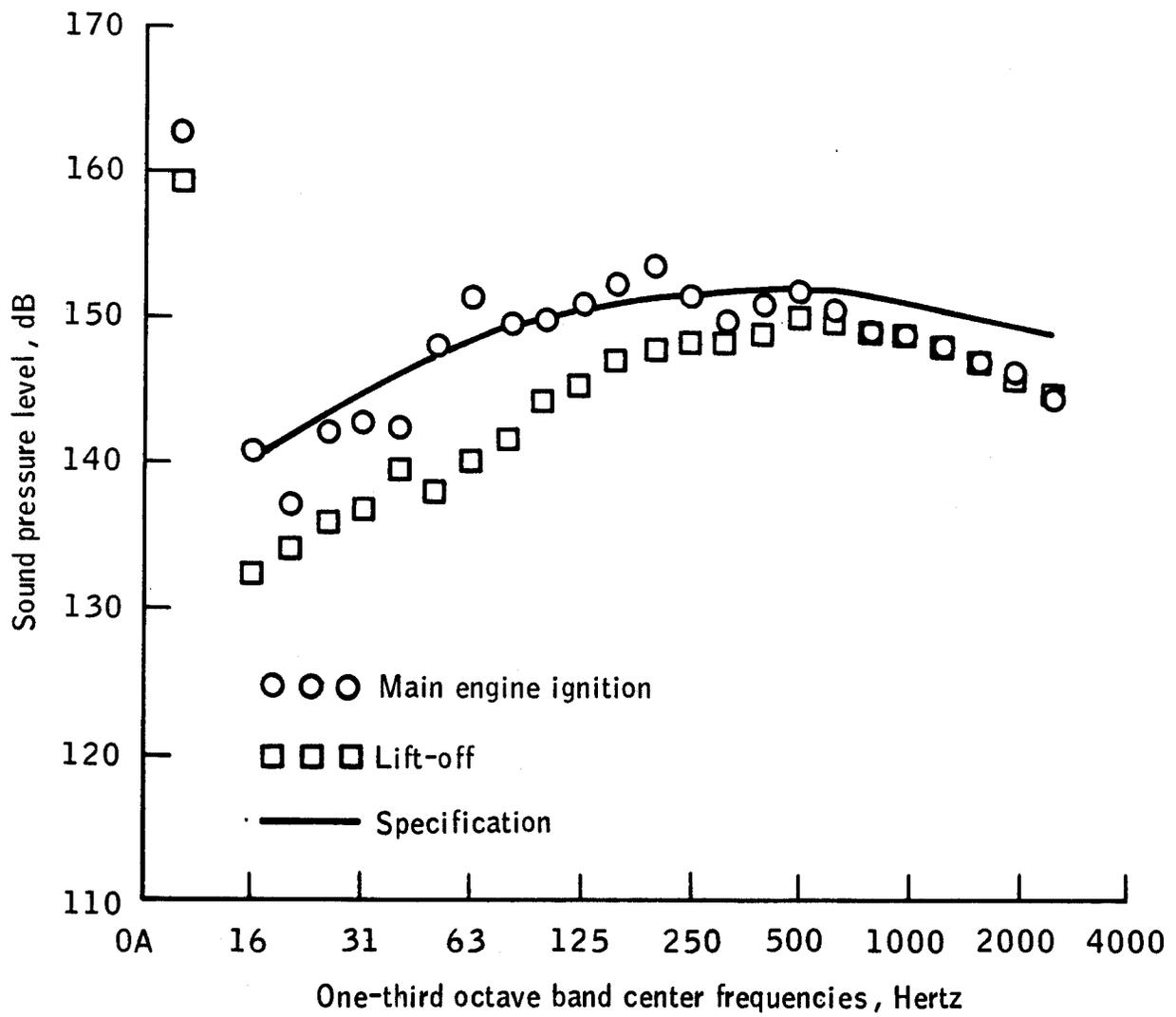


Figure 3-1.- Acoustic measurement of tip of vertical stabilizer station.

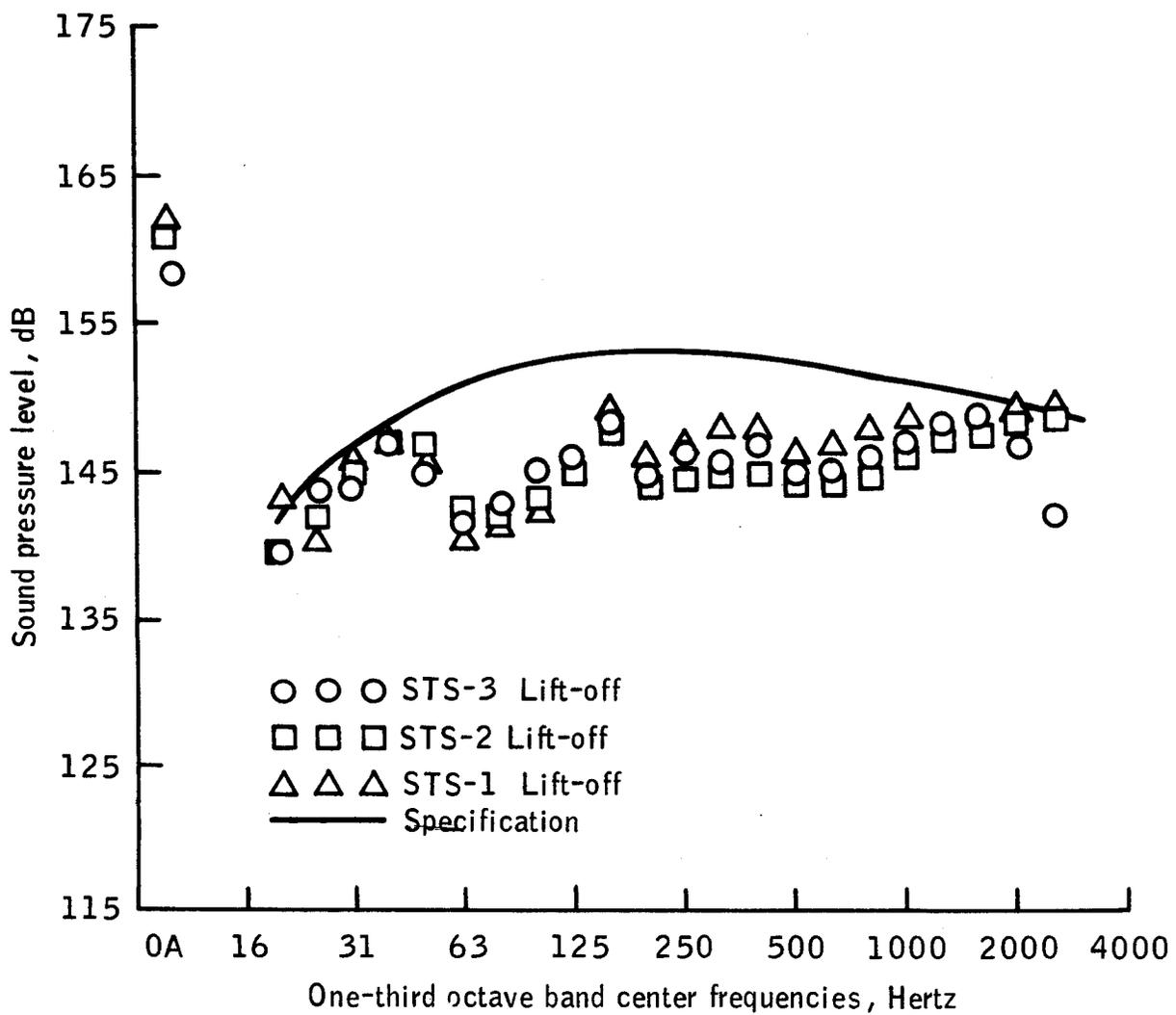


Figure 3-2.- Acoustic measurement on aft fuselage heat shield.

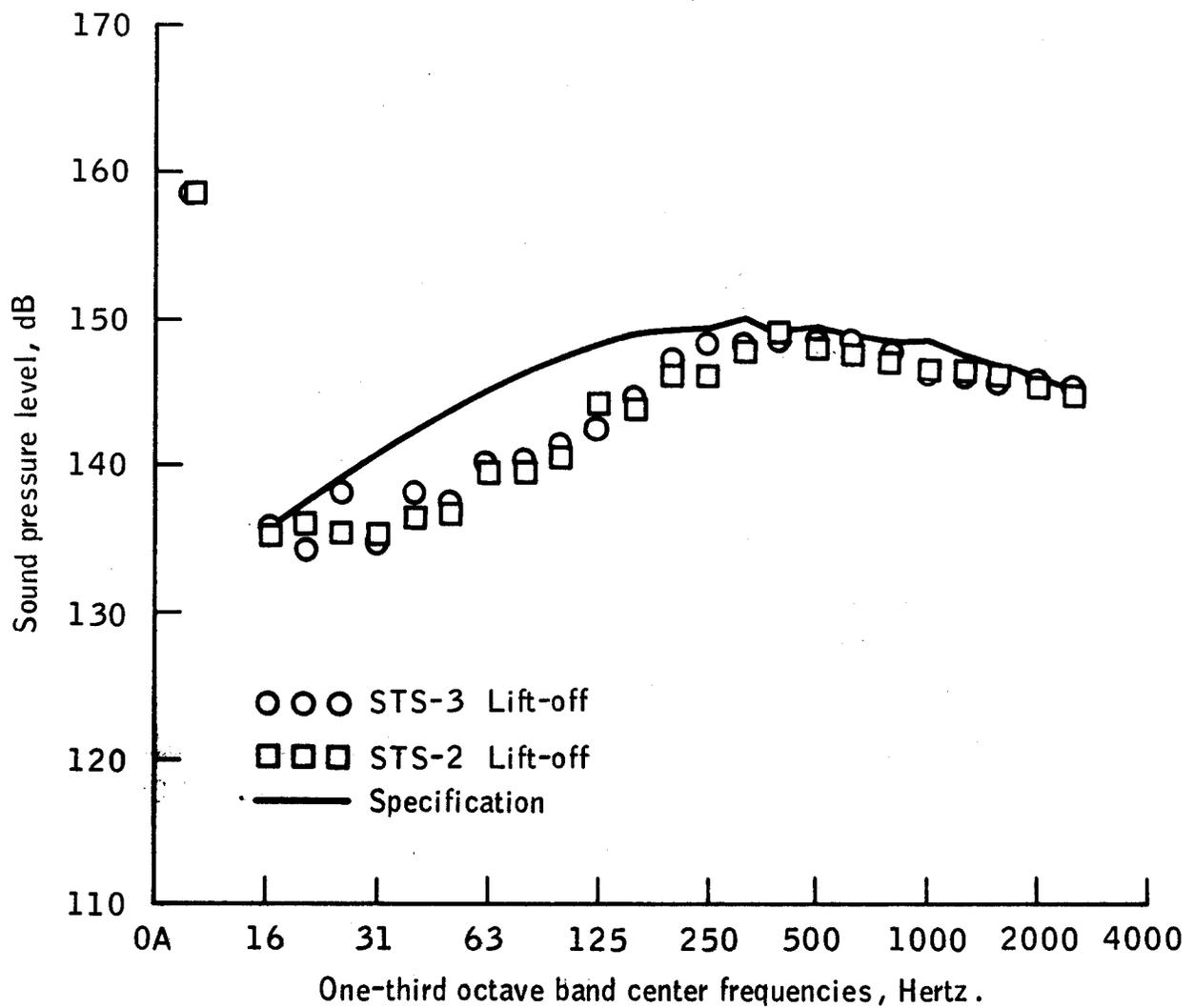


Figure 3-3.- Acoustic measurement on inboard elevon actuator.

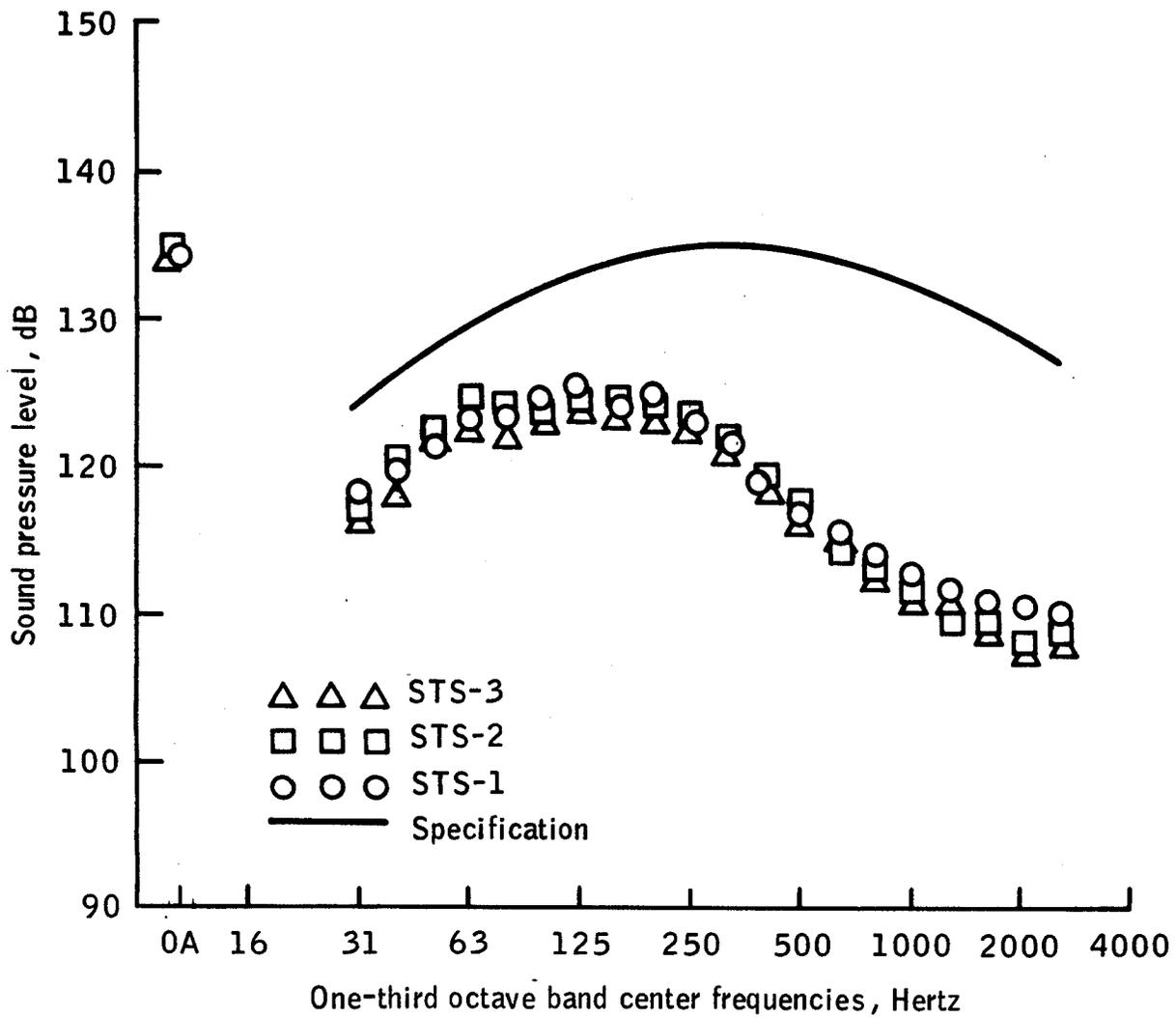


Figure 3-4.- Spatial average of internal acoustic measurements from payload bay at lift-off.

3.2 VIBRATION ENVIRONMENT

Vibration data analyzed from the STS-3 mission is in compliance with the design criteria and agrees with the previous flights. The areas that have had anomalies on previous flights are still showing the same anomalies. The anomalies will not affect the remaining OFT flights, but may impact mission life certification.

The APU are continuing to show vibration levels that are not compatible with each other. APU's 1 and 2 have vibration levels that are consistent from flight to flight with APU 2 slightly higher than APU 1. The levels are attributable to the loading of the systems. APU 3 had the same high level of vibration that has appeared on the previous flights and is considerably higher than APU 1 and 2. This high level of vibration has been observed during the qualification testing with no adverse affect on the operation of the unit. The phenomenon that was observed on APU 3 on this flight was that the level started to increase approximately 2 minutes into the flight as compared to 8 minutes into the flight on previous flights. This is coincident with the increase in temperature that eventually caused the APU to be shutdown at approximately 8 minutes into the flight.

The measured vibration level within the payload bay area is in compliance with the design criteria for the X and Z axis. The measured data in the Y-axis shows a slightly higher response than the predicted level and further data analysis will have to be performed to determine the validity of the data.

In summary, the vibration environment measured during STS-3 is representative of the previous flights.

4.0 BIOMEDICAL EVALUATION

The primary objective of the medical operations effort is to assure the health and well-being of flight personnel during all phases of the mission.

4.1 EVALUATION OF CREW HEALTH

The preflight evaluation of crew health occurred at launch minus 30 days (February 22), launch minus 10 days (March 10), and launch morning. The crew was in excellent health throughout the preflight period. Inflight, the crew reported symptoms of space sickness. Symptoms persisted for three days in each crewman. Each crewman took medication for space sickness while symptomatic. The Commander's sleep periods were interrupted by radio static on occasion the first 3 nights. The crew rested well the latter part of the mission. Postflight analysis showed that the Commander and Pilot maintained hydration. They each consumed approximately 1900 calories of food per day. A salt-fluid cardiovascular countermeasure was not completed because of the delayed mission termination. Postflight evaluation of crew health on landing day and three days later found them to be in excellent health.

4.2 HEALTH STABILIZATION

The health stabilization program has been in effect since Apollo 14. No infectious diseases have been reported in prime crewmembers in the preflight and inflight period from Apollo 14 through STS-3. The number of authorized primary contacts for STS-3 was 236. This is compared to 927 primary contacts for STS-1 and 164 primary contacts for STS-2.

4.3 RADIATION

There were no unexpected exposures to radiation on STS-3. The projected radiation dose to the STS-3 crew was 67 millirad. TLD's (thermoluminescent dosimeters) worn by the crew measured 46.1 ± 2.6 mrem (average). Pocket dosimeters placed at various locations in the spacecraft to verify shielding calculations averaged 55 ± 3 millirad. TLD's in these same locations averaged 46.7 mrem. The average HZE flux was 22 ± 3 particles/cm² for the entire

8-day mission. The spacecraft passed through the South Atlantic Anomaly 48 times during the mission. Solar flare activity was minimal throughout the mission.

4.4 MOTION SICKNESS PROPHYLAXIS (DSO S343)

The motion sickness prophylaxis (DSO S343) objective was to provide motion sickness prophylaxis and to collect relevant data. Crew motion sickness experience was evaluated preflight and the motion sickness susceptibility test was completed. Side effects and the efficacy of anti-motion sickness drugs were evaluated. A microcassette recorder was used inflight to record crew data. Both crewmen used scopolamine/dexedrine in flight on days 1, 2, and 3. Both crewmen reported some space motion sickness symptoms; however, these symptoms were not severe and did not interfere with the completion of the mission objectives. Postflight testing has not been completed.

4.5 TOXICOLOGY

The trace contaminant gas composition of the Orbiter cabin atmosphere was determined by utilizing two different sampling methods. Four gas sampling cylinders were used by the STS-3 crew to collect atmospheric samples at evenly spaced periods during the mission. Ground-based analyses of these samples yielded both qualitative and quantitative information.

The second sampling method was to examine small quantities of the activated carbon contained in the cartridges used to remove carbon dioxide from the cabin atmosphere. Analyses of the activated carbon samples yielded qualitative information only.

Similar to the STS-1 analytical results but quite different than the STS-2 results, the STS-3 cabin atmosphere was relatively free of contaminant gases. A total of 20 compounds were identified, all of which were well below the spacecraft maximum allowable concentration limits. The fire extinguishant Halon 1301, used in the Orbiter, was detected at a high value of 3 parts per million (ppm). One of the Orbiter fire extinguishers had been discharged as a precautionary measure because of faulty smoke detectors.

4.6 MICROBIOLOGY

The prime crewmen were microbiologically evaluated at launch minus 30 days, launch minus 10 days, launch minus 2 days, and landing +3 days. The backup crew was evaluated at launch minus 30 days, launch minus 10 days, and launch minus 3 days. All crewmen exhibited normal microbial flora in ears, nose, throat, urine, and feces cultures. No detectable microbial exchange between crewmen occurred during the mission.

Microbial evaluation of the Orbiter consisted of collecting 21 surface samples and 4 air samples at designated sites throughout the mid-deck and flight deck at launch minus 30 days, launch minus 2 days, and landing day. Most of the surface sites exhibited a significant buildup of total bacteria and total fungi. Increases of 100 to 1000-fold in total bacteria per 25 cm² were not unusual. Thirteen different species of the potential fungal genus, Aspergillus, were isolated during the sampling periods, whereas, no pathogenic bacteria were isolated. No significant buildup in total numbers of airborne bacteria and fungi occurred during the mission. No pathogenic bacterial were isolated from the air; two genera of potentially pathogenic fungi were isolated, Alternaria and Dreschlera hawaiiensis. Typical water-borne bacteria were isolated from the on-board potable water supply. No pathogens were isolated from the potable water.

4.7 BIOINSTRUMENTATION

Heart rate data were within expected limits. The signal for one crewmember was not received. This is under investigation since the crewmember followed prescribed procedures.

5.0 DESCENT TRAJECTORY

The entry trajectory for STS-3 was completely normal. Entry interface (400000-ft altitude) was reached at a range-to-runway distance of 4136 nmi. The preflight prediction was 4135 nmi. The ranging errors remained negligible during all of entry, as shown in figures 5-1 and 5-2. The drag acceleration profile flown compares well with preflight predictions (figure 5-3), and roll magnitudes and reversals also show close agreement (figure 5-4). Figure 5-5 shows that the vehicle angle of attack deviated slightly from the prediction, but only during the alpha modulation associated with roll reversals, and then by less than 1 degree. Figure 5-6 indicates once again excellent trajectory control with predicted and actual altitudes nearly identical.

The TAEM phase (2500 ft/sec velocity) was achieved at a range of 59.13 nmi., and an altitude of 80469 ft. The preflight predictions were 58.5 nmi., and 78308 ft. The TAEM groundtrack and altitude profiles are shown in figures 5-7 and 5-8, respectively. The CSS mode was engaged at an altitude of 41811 ft. for the turn around the heading alignment circle and the auto mode was reestablished at 11824 ft.

At this point, all necessary parameters were within autoland engage limits, and transition to this mode occurred at 10000 ft. The autoland system flew the vehicle to an altitude of 143 ft. where manual control was selected. Main gear touchdown was 1092 ft. past the threshold at 220 knots estimated air speed. The nose gear touched down 6261 ft. past the threshold, with wheels stop at 14824 ft. (figure 5-9).

6.0 ORBITER ANOMALY SUMMARY

This section contains a summary listing (Table 6-I) of each anomaly defined during the flight, post-flight testing, and data analysis. Also included are the problem closeout reports with the status of each problem at the time of publication of this report.

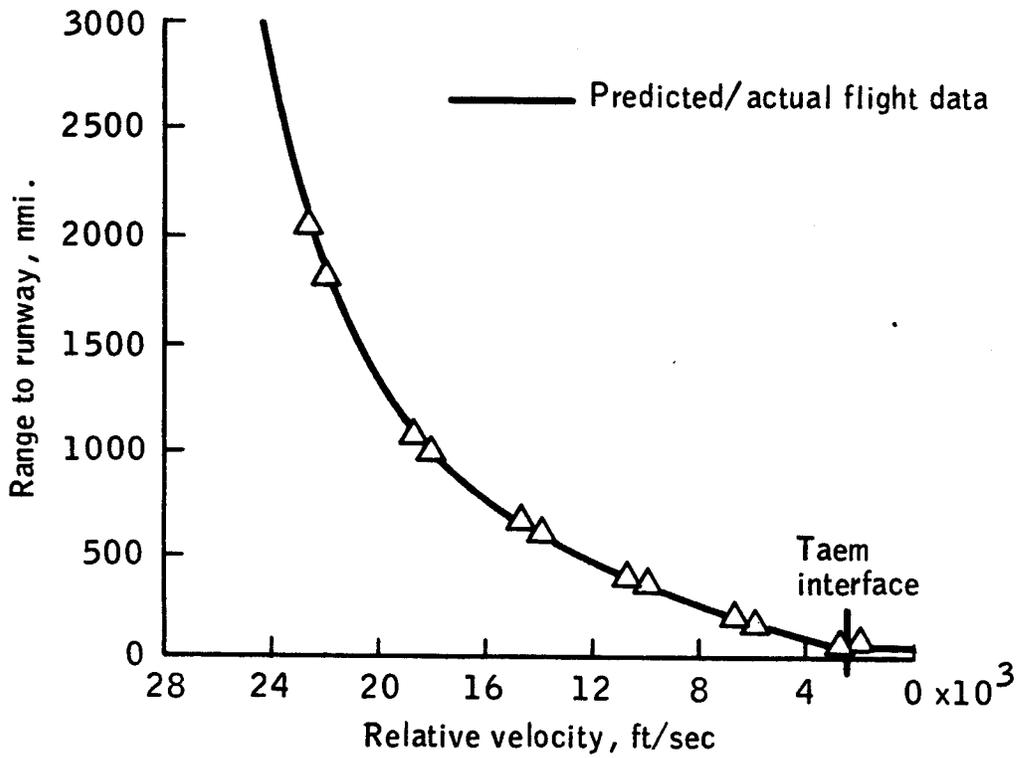


Figure 5-1.- Entry ranging performance.

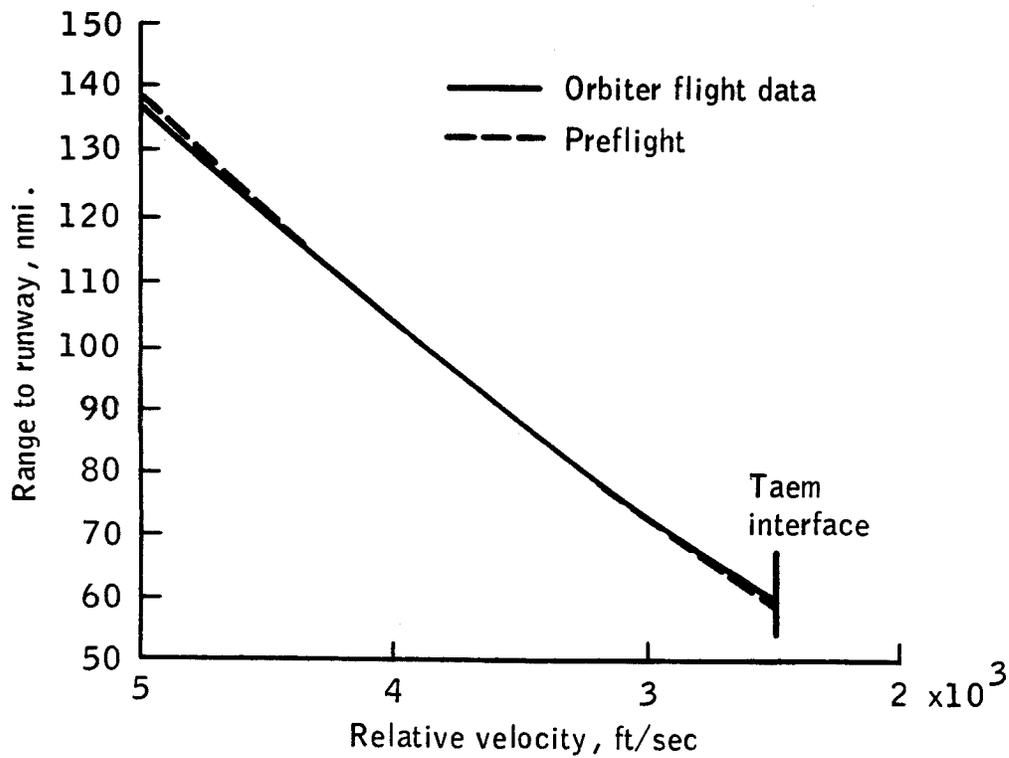


Figure 5-2.- Late entry energy management.

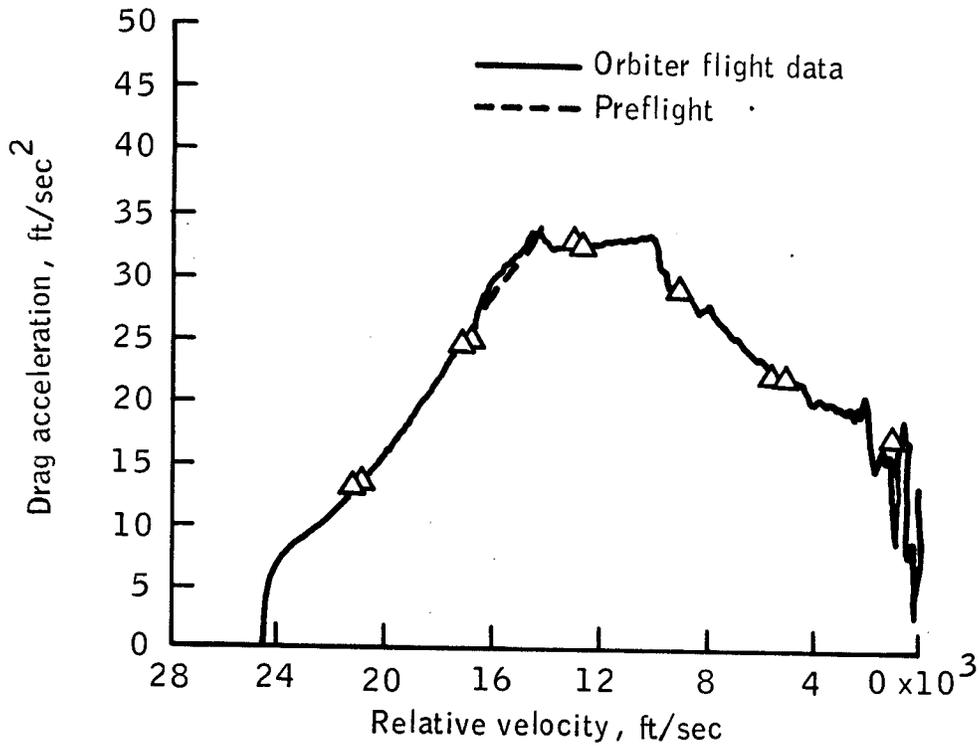


Figure 5-3.- Entry drag acceleration.

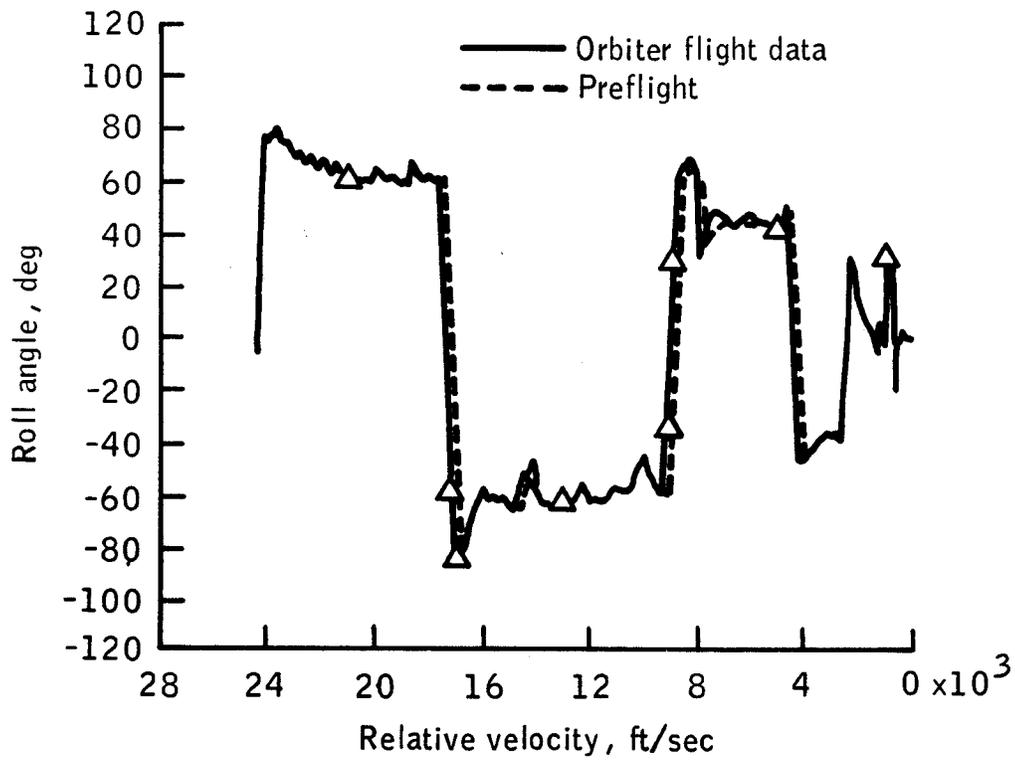


Figure 5-4.- Entry roll performance.

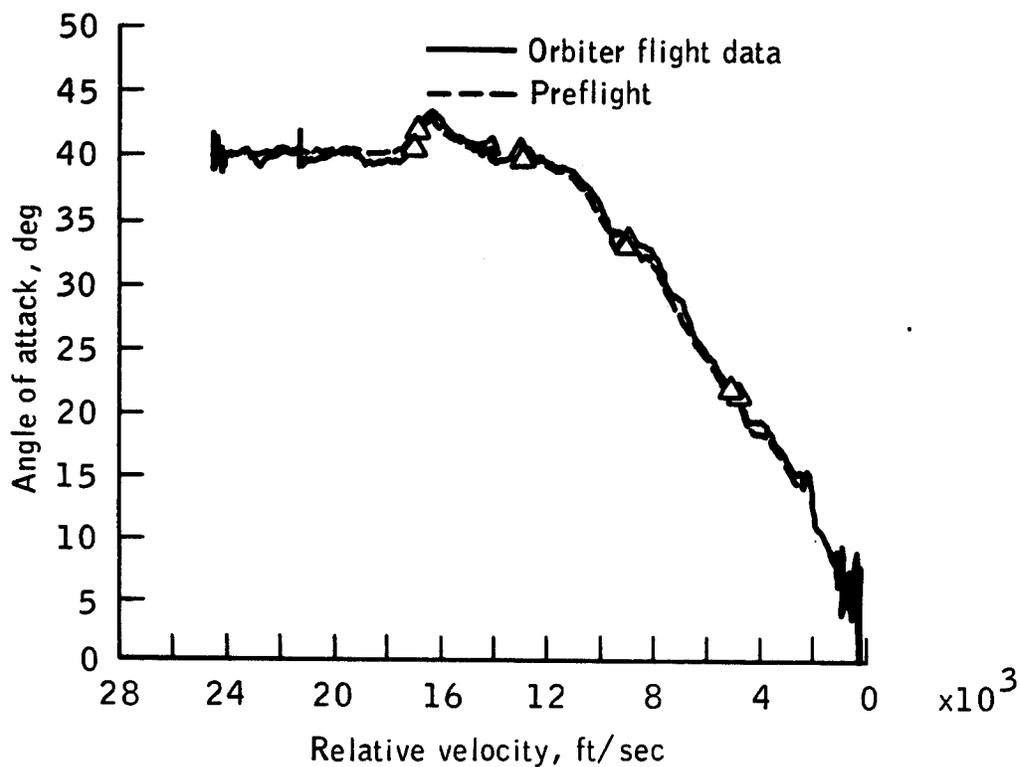


Figure 5-5.- Entry angle of attack performance.

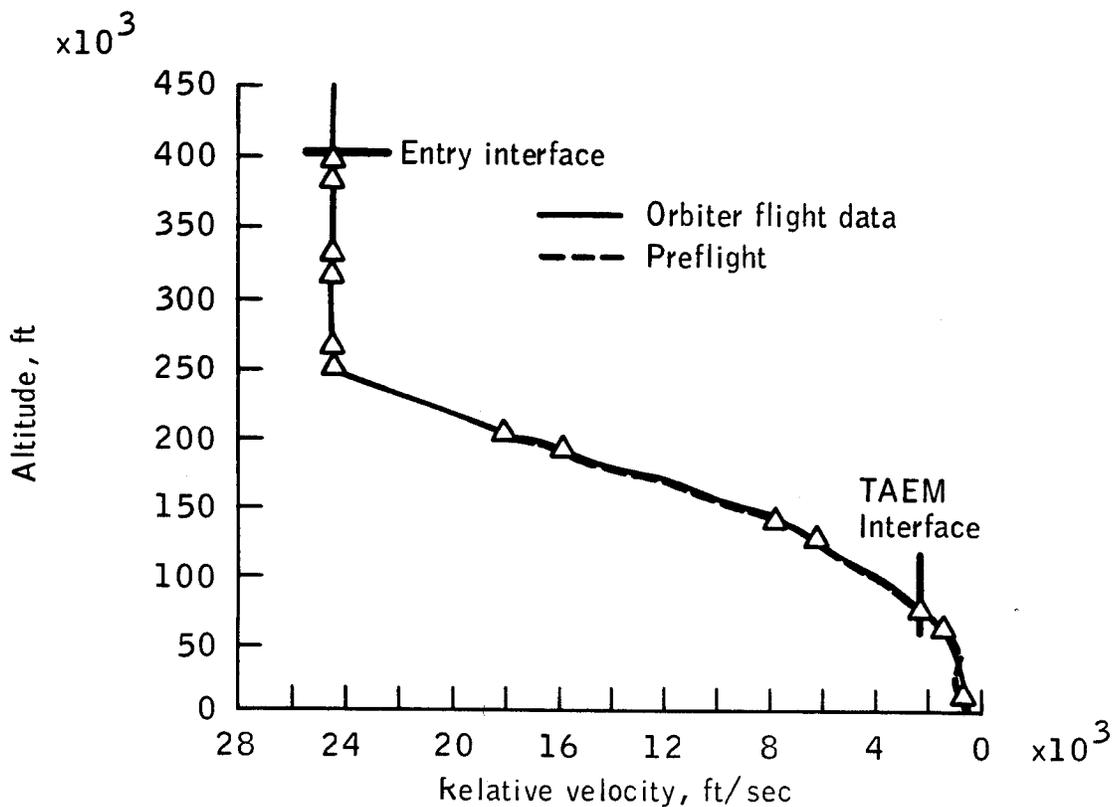


Figure 5-6.- Entry altitude profile

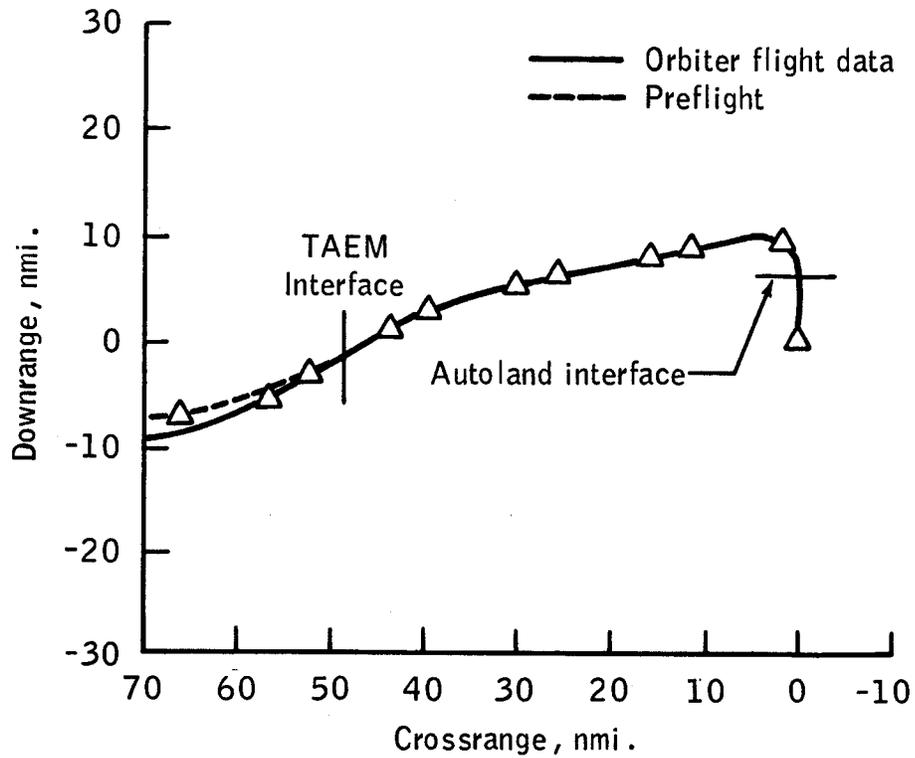


Figure 5-7.- TAEM groundtrack.

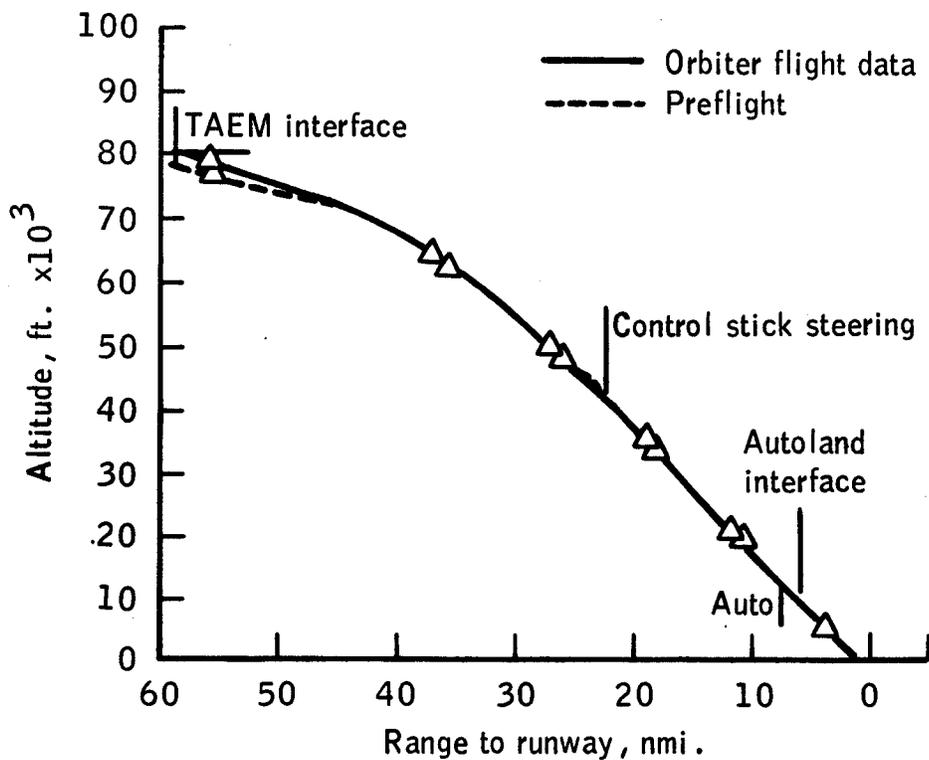


Figure 5-8.- TAEM altitude profile.

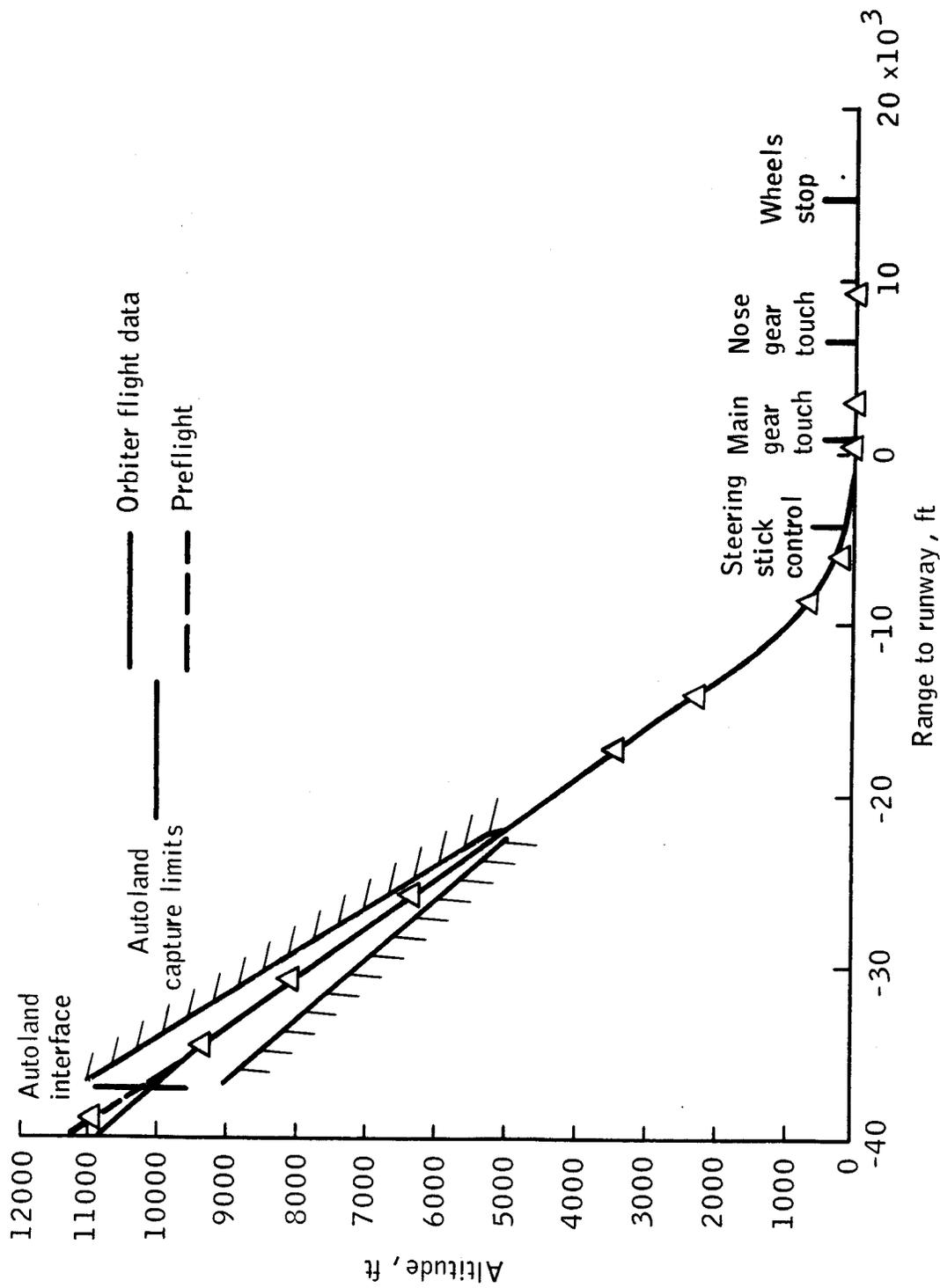


Figure 5-9.- Autoland altitude profile.

TABLE 6-I.- SUMMARY STATUS OF STS-3 PROBLEMS

No.	Title	Status
1	RPC 2 in H ₂ O ₂ control box 1 failed on.	CLOSED
2	MCA power AC3 three-phase forward circuit breaker failed to latch.	CLOSED
3	Heater A trip indication on oxygen tank 3.	CLOSED
4	Water spray boiler 3 allowed APU 3 lubrication oil to exceed high limit.	CLOSED for STS-4
5	APU 2 fuel cavity drain system leaked.	CLOSED
6	Vernier RCS engine F5L oxidizer temperature not tracking injector temperatures.	CLOSED
7	Smoke detector 3A and the detector on the left-hand flight deck failed self-test.	CLOSED
8	RMS wrist TV camera inoperative.	CLOSED
9-1	Forward bulkhead PLBD switch modules showed ready to latch after port door opened.	CLOSED for STS-4
9-2	Aft bulkhead actuator on port cargo bay door stalled during latch closure.	CLOSED
10	Seven operational instrumentation sensors failed.	CLOSED
11	IMU 3 accelerometer shifted abruptly on first day.	CLOSED for STS-4
12	RMS ready-for-latch indicator switch failed to transfer.	CLOSED
13	Video tape recorder (VTR) cassette jammed.	CLOSED
14	Waste Management System (WMS) slinger speed erratic and circuit breaker opened.	CLOSED for STS-4
15	CCTV Camera C failed.	CLOSED
16	Missing tiles.	CLOSED
17	ARPCS GN ₂ usage excessive.	CLOSED for STS-4
18	All PDRS DAC (16 mm) cameras inoperative except mid-aft.	CLOSED
19	WCS vacuum vent valve knob came off.	CLOSED
20	CCTV camera B zoom did not operate.	CLOSED for STS-4
21	S-band transponder 2 failed in low power mode (downlink).	CLOSED
22	RMS encoder check false alarms.	CLOSED
23	S-band transponder 1 failed in both high and low power modes (downlink).	CLOSED
24	APU 3 fuel tank pressure decay.	CLOSED
25	Wireless crew communications unit A transmitter not functioning.	CLOSED
26	Cap seal on several drink bags failed.	CLOSED
27	DFI wideband mission recorder bite 1 indicated fail.	CLOSED

TABLE 6-1.- SUMMARY STATUS OF STS-3 PROBLEMS - CONCLUDED

No.	Title	Status
28	S-band power amplifier 1 indicated a slight downward trend in RF power out.	CLOSED for STS-4
29	CRT 1 did not respond to entries using keyboard 1.	CLOSED
30	RMS servo power amplifier (SPA) commutator failure message on wrist-yaw joint after port arm select.	CLOSED
31	Flash evaporator primary B controller exhibited erratic control and then shutdown.	CLOSED for STS-4
32	Payload main bus C talkback (DS17 PNL R1A1) not operating.	CLOSED
33	Smoke detector 2A gave an erroneous alarm.	CLOSED for STS-4
34	Mid MCA 3 operations status 4 at zero - should be one.	CLOSED
35	Ammonia boiler controllers A and B failed.	CLOSED for STS-4
36	Lost radiator "latched" indication.	CLOSED
37	MDM FF4 input/output error.	CLOSED
38	MPS 3 LOX prevalve RPC leak.	CLOSED for STS-4
39	MPS 3 LOX internal leak.	CLOSED
40	VTR run light inoperative.	CLOSED
41	Landing evaluation.	CLOSED
42	Debris camera failed.	CLOSED
43	Cracked rotor on RH outboard brake.	CLOSED for STS-4
44	Water in air lock drag in duct.	CLOSED for STS-4
45	Crew compartment DAC camera end of film light inoperative.	CLOSED
46	Broken tabs on rudder speed brake seal blocks.	CLOSED
47	Urine nozzle seal leak.	CLOSED

Statement of problem: RPC (Remote Power Controller) 2 in cryo control box 1 failed "on"

Discussion: Prior to launch, the hydrogen tank 1 heater B was failed "on" in the automatic mode. The problem was isolated to RPC 2 in cryo control box 1. Heater B was de-energized by placing the panel control switch in the off position. Only one heater per tank is required for flight, and the crew procedures were changed to use only heater A in hydrogen tank 1.

Postflight tests and inspection isolated the problem within the RPC to a "short" between lead wire 4 and terminal pad 3 of the current regulator LID (leadless inverted device). This allowed the RPC to remain on without the presence of a control signal.

Improper routing of the lead wire caused an electrical breakdown between the lead wire and the terminal pad. This is the fourth RPC failure due to LID lead wire "shorts" experienced in the shuttle program with 1000 series RPC's.

Production and inspection has included longer leads and a visual clearance check to ensure proper lead routing before potting for all LID's in the 2000 series RPC's. This series RPC is used for replacement on OV-102 and for all the RPC's on OV-099 and subsequent vehicles.

Conclusions: The RPC failure was due to a "short" between the lead wire and a terminal pad of the current regulator leadless inverted device. This failure has occurred on other OV-102 RPC's and corrective actions to eliminate the short have been taken for OV-102 replacement RPC's, and the RPC's for OV-099 and subsequent vehicles.

Corrective action: The failed RPC in cryo control box 1 has been replaced with an upgraded design and the cryo box has been retested for STS-4.

APPROVED

A. Cohen
A. Cohen

6/7/82
Date

Effect on subsequent missions:

None

L. J. Moon *R. J. Rice* *Robert J. Ward*

Personnel assigned: L. J. Moon/EH5; R. Rice/EP5; R. J. Ward/WA3

Resolution: CLOSED 6/7/82

Statement of problem: MCA Power AC3 Three-Phase Circuit Breaker (CB 11 Pnl. MA 73C) Failed to Latch When Actuated.

Discussion: Prior to the STS-3 flight, the MCA power AC3 circuit breaker could not be actuated. This breaker MCA Power AC3 protected the -Z star tracker door, motor No.1, the right ADTA motor no. 1 and the left vent door 1/2 motor no. 1. An external mechanical latch was installed which closed the breaker. This configuration remained intact throughout the flight. The circuit breaker has been removed and replaced by another similar breaker. Analysis indicated the clevis pulled out of the socket, thus preventing the breaker from operating properly. This specific failure mode can inhibit the "trip free" (CB operates with handle held in place) feature of the circuit breaker. This possibility was recognized prior to flight and accepted based on the fact that if a short would have developed, the current limiting circuit in the inverter would have opened prior to overloading the wire.

Conclusions: The most likely cause of the failure was excessive force being used to open the breaker.

Corrective action: Users will be cautioned not to use excessive force when pulling breakers.

APPROVED

A. Cohen
A. Cohen

5/24/82
Date

REB

Effective on subsequent missions: None

Personnel assigned: *A. J. Harkas*
A. J. Harkas/EH5

Resolution: CLOSED 5/17/82

Statement of problem: Heater "A" trip indication on O₂ tank 3.

Discussion: At t+35 seconds after lift off O₂ tank 3 heater "A" current level detector (CLD) tripped inhibiting the heater. Since the heaters were "off" and the pressures were too high to activate the heaters in "auto", this was a nuisance trip. The CLD was reset and did not trip again. If the CLD does not reset, the detector can be inhibited by opening the circuit breaker. The O₂ tanks have redundant heater systems and each system has redundant CLDs.

Postflight testing has verified proper operation of the CLD and did not identify a cause for the trip indication. Analysis indicates that the driver latches in 3 to 4 milliseconds and a 70 millisecond glitch can trip the CLD.

Conclusions: The cause of the trip indication is unknown. The CLD is operational.

Corrective action: Postflight testing verified proper current level detector operation.

APPROVED *Caron Cohen* *5/12/82*
 A. Cohen Date
SCJ 5/13/82

Effective on subsequent missions: None

Robert R. Rice 5-18-82

Personnel assigned: R. Rice/EP5; R. J. Ward/WA3

Resolution: CLOSED 5/3/82

Statement of problem: Water spray boiler (WSB) 3 allowed APU 3 lube oil to exceed the high limit.

Discussion: WSB3 froze during ascent of STS-3 resulting in APU shutdown. WSB3 worked properly during on-orbit checkout and entry.

Extensive tests were performed at the JSC test facility and the vendor. Neither test was able to exactly duplicate the failure under simulated flight-type conditions. One freezeup did occur at the vendor test facility when heat loads were reduced below flight levels (50 Btu/min). Tests did show that the present 1.25" steam vent orifice allows freezing conditions to be reached in the boiler during ascent. Tests also demonstrated that a 0.8" orifice keeps conditions above freezing on ascent and provides good performance for on-orbit and entry conditions.

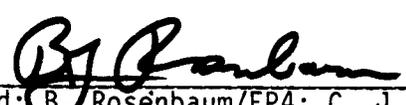
Conclusions: The freeze-up of WSB3 caused the APU lube oil to exceed the high limit. WSBs 1 and 2 have performed satisfactorily on the 3 previous flights. Ground tests with simulated flight conditions have not exactly duplicated the flight problem.

Corrective action: The existing 1.25" steam vent orifice on WSB3 will be replaced with a 0.8" orifice.

APPROVED


A. Cohen
Date

Effect on subsequent missions: None

 6/11/82  6/11/82
Personnel assigned: B. Rosenbaum/EP4; C. J. Walsh/WC-6

Resolution: CLOSED for STS-4 6/11/82

Statement of problem: APU 2 Fuel Cavity Drain System Leaking

Discussion: The APU 2 fuel cavity drain line leaked to 0 psia approximately 21 hours after launch. During entry, the leak rate was much greater, approximating the increasing atmospheric pressure.

A post-flight system leak check at KSC isolated the problem to the drain relief valve. When the valve was disassembled at the vendor, the soft seat was found to have a depression across the sealing surface. The valve was also heavily contaminated. The majority of the contamination was cotton fiber, gold, brush fiber, glass granules, clear nylon, and polymer. These materials are not present in the APU. The remaining particles (iron oxide, 303 stainless, and aluminum), while they may have come from the APU, were not of sufficient magnitude to cause concern for APU performance or wear. The contamination was most probably introduced during APU replacement after STS-1. A possible source of the contamination is the catch bottles on OV-102 which were used in a mockup prior to vehicle installation. They were not ultrasonically cleaned (as now required), but were flushed after installation.

Conclusions: The leak was caused by a depression in the soft valve seat of the drain relief valve due to contamination. The contamination was most probably introduced during APU replacement after STS-1.

Corrective action: The APU 2 drain relief valve has been refurbished and reinstalled. The drain lines on all three APU's have been flushed with alcohol for cleaning.

APPROVED
RCB
5/25/82

Aaron Cohen
A. Cohen

5/25/82
Date

Effective on subsequent missions: None

Rennie J. Lance 5/25/82 Robert J. Ward 5/25/82
Personnel assigned: R. J. Lance/EP4; R. J. Ward/WAS

Resolution: CLOSED 5/26/82

Statement of problem: VRCS Engine F5L OX Injector Temperature Not Accurately Tracking

Discussion: Vernier reaction control system (VRCS) engine F5L oxidizer injector temperature, a leak detector, was not accurately tracking the engine injector temperature as measured by the fuel injector temperature. Normally the oxidizer to fuel difference is less than 10° F, but the F5L Ox leak detector showed a non-linear bias ranging up to 60° F depending on temperature. The Ox temperature read low and the higher the temperature, the greater the difference. The detector did work adequately to prevent an erroneous leak indication.

KSC electrical troubleshooting of the sensor and signal conditioner shows normal operation. Both the OX and Fuel injector temperatures tracked properly during a heat test at KSC to a temperature of 190° F. Additional troubleshooting would require removal of the engine. This action is not considered necessary since the sensor locations in the thruster can detect either an OX or a fuel leak.

Conclusions: The cause of this problem is unknown.

Corrective action: A memory read/write software change will be verified for STS-4 allowing the FRCS vernier thrusters to use only the fuel leak detectors. Fly "as is" with this software backup.

APPROVED

*Aaron Cohen*5/18/82
DateSCJ 5/18/82
A. Cohen

Effective on subsequent missions: None

C. W. Hohmann *Robert J. Ward* 5/17/82

Personnel assigned: C. W. Hohmann/EP4; R. J. Ward/WA3

Resolution: CLOSED 5/14/82

Statement of problem: Smoke detectors, 3A and left flight deck, failed to self-test

Discussion: Smoke detector 3A failed to self-test twice (Day 1 and 3). On day 3, the left flight-deck smoke detector also failed to self-test. Both smoke detectors passed the postflight self-test. These detectors had already exceeded their age life (800 hours) and were waived for flight use. The air pump bearings are the life limiting items. Bearing torque increases with operation and this causes the motor current to exceed the self-test limit.

Replacing the smoke detectors with the new design units requires removal of avionics boxes for access which impacts the turnaround timeline. Should access become available, the failed units will be replaced.

Conclusions: The problem was caused by the air pump bearings that bind occasionally because of the lubricant used. Repeating the self-test will normally clear the fault.

Corrective action: Replace units, if access become available.

Henry A. Rotter Jr.

APPROVED

A. Cohen
A. Cohen

5/11/82
Date

Effective on subsequent missions: The remaining six smoke detectors of the old design may fail the inflight self-test.

Personnel assigned: H. A. Rotter/

Resolution: CLOSED 5/12/82

Statement of problem: RMS Wrist TV Camera Inoperative

Discussion: The RMS TV circuit breaker number 55 on panel R15 tripped when the camera was activated. A component failure inside the power supply resulted in an electrical short. The RMS elbow camera on STS-2 and the aft bulkhead camera "C" on STS-3 also experienced the same failure mode. In each case, the camera temperatures were approximately 0° C.

Failure analysis and tests have determined that the failures were caused by the erratic operation of the camera dc/dc power converter at cold camera temperatures near 0° C.

Cold temperatures decreased the base-emitter voltage of the switching transistor resulting in a decreased voltage at the collector. This created an unstable frequency within the dc/dc converter that generated high voltage spikes shorting the output transistor.

Conclusions: Cold temperatures decreased the base-emitter voltage of the switching transistor resulting in a decreased voltage at the collector. This created an unstable frequency within the dc/dc converter that generated high voltage spikes shorting the output transistor.

Corrective action: The input resistor has been changed to provide more voltage drive to the switching transistor. Diodes have also been added to protect the output transistor from high voltage spikes. All flight TV cameras were modified and tested prior to STS-4.

APPROVED

[Signature]
A. Cohen

5/10/82
Date

Effective on subsequent missions: None

RE *RJSK*

Personnel assigned: R. Edmiston/EE2; R. J. Ward/WA3

Resolution: CLOSED 05/07/82

Statement of problem: Forward bulkhead PLBD (payload bay door) switch module showed "ready to latch" after port door was opened.

Discussion: During the port door opening sequence after a long cold soak with the orbiter tail to the sun, the 3 forward ready-to-latch (R-T-L) switches failed to transfer to the OFF position, indicating the R-T-L switch actuating arm failed to rotate enough to allow switch transfer. After 15 minutes of the top-to-sun attitude, the port door was closed and the forward and aft bulkhead latches operated normally. The port door was then opened and the forward R-T-L indicator returned to its normal "off" position. All components operated normally for the remainder of the mission.

Tests and analysis of the switch module verified that a combination of low temperatures and minimal arm-to-housing clearances caused binding that prevented the actuating arm from rotating to the point required for switch transfer. This condition does not prevent successful door operation as long as the "door closed" switch transfers and recovery is always possible by a vehicle attitude change to warm the module.

Conclusions: The combination of low temperatures and minimal arm-to-housing clearances caused binding that restricted rotation of the actuating arm in the PLBD switch module. Failure of the "ready to latch" switch to transfer does not prevent successful door operation and the switch will operate properly if the vehicle attitude is changed to warm the module.

Corrective action: Fly as is for STS-4. A design change has been recommended to increase the spring force and open the clearance between the arm and the housing.

APPROVED

A. Cohen
A. Cohen

6/11/82
Date

Effective on subsequent missions: Possible recurrence of the problem under cold conditions.

Robert Langley 6-11-82

Personnel assigned: N. Jervas/EW3, R. D. Langley/WTB

Resolution: CLOSED for STS-4 6/11/82

Statement of problem: Aft bulkhead actuator on port door stalled during latch closure.

Discussion: During the door closure test following approximately 23 hours with the orbiter in the tail-sun attitude, the aft port bulkhead latch gang failed to reach the fully latched position. The forward and aft latches were then unlatched, the door was opened, and the vehicle was reoriented to a top-sun attitude for 15 minutes followed by 42 minutes of PTC. The door was then successfully closed and latched. Subsequent door operations were normal.

Tests/Analysis/Inspection revealed the following:

1. Actuator - Two of four mounting lugs were cracked, the torque limiter slipped at the proper limit, there was no internal damage or wear that could have caused the problem, and the actuator housing was not properly heat treated. The reduced strength housing still resulted in a high margin of safety for the design case and would have failed due to an external jam if properly heat treated.
2. Thermal - The forward bulkhead seals were significantly colder than the aft bulkhead seals and were warmer than design limits. Door-to-vehicle distortions were eliminated as possible causes because the door was essentially latched and theodolite data did not indicate any significant door deflection.
3. Motor Current Data - The motor current began to increase after 15 seconds operation and stalled at 20 seconds. Nominal runtime is 24 seconds.
4. Other damage - None.
5. Analysis - The load required to cause the failure was isolated to the #1 latch.

Conclusions: Based on observed damage, motor current data, and elimination of other failure modes, the anomaly was most probably caused by jamming due to an external object. Based on motor current the object was compressible and began to load the motor at 15 seconds (3/8 inch object) and went to stall at 20 seconds (1/16 inch object).

Corrective action: The hardware has been fully inspected and all damaged components have been replaced. The payload bay has been inspected and cleaned. Functional and structural margins have been reverified.

APPROVED

Aaron Cohen
A. Cohen

6/11/82
Date

WIK

Effective on subsequent missions: None

Released by / 6-11-82

Personnel assigned: N. Jervas/EW3

Resolution: CLOSED 6/11/82

Statement of problem: Operational Instrumentation Failures

- Discussion: The following operational instrumentation failures occurred during STS-3:
- a. V41T1261A - Main engine 2 GH₂ outlet temperature measurement went off-scale high for 8 minutes and then returned to normal.
 - b. V41P1300C - Main engine 3 LH₂ inlet pressure failed off-scale high.
 - c. V45Q2205A - PRSD Hydrogen tank 2 quantity failed off-scale high.
 - d. V58T1143 - Hydraulic system 1 mid-body return line temperature was biased low.
 - e. V58T0198 - Hydraulic system 1 rudder speed brake return line temperature response lagged another measurement on same line.
 - f. V58T0398 - Hydraulic system 1 rudder speed brake line return temperature failed off-scale high.
 - g. V62Q9150 - Water tank quantity shifted from 85 percent to zero and back to normal. This was the result of a known contamination problem with the sensor wiper.

Conclusions: Measurement failures as noted.

Corrective action: a. Transducer replaced with improved design; b. Transducer replaced; c. Defective signal conditioner replaced; d. Measurement in tolerance, so fly as is; e. Attributed to sensor location, so fly as is; f. Transducer replaced; g. Tank removed for STS-4 so no action required. The PRSD H₂ Tank Quantity measurement is the only measurement used for Launch Commit Criteria on Ground Launch Sequence operations and it has been repaired.

APPROVED

A. Cohen
A. Cohen

5/19/82
Date

Effective on subsequent missions: *scf 5/18/82* None

Personnel assigned: *Watch for* F. Rotramel/6411; R. Ward/WA3

Resolution: CLOSED 5/11/82

Statement of problem: IMU 3 accelerometer bias shifted abruptly on the first day.

Discussion: At approximately 82:05:45 G.m.t. IMU 3's null bias acceleration shifted by $-321 \mu\text{g}$ in the x axis and $+383 \mu\text{g}$ in the y axis. The bias was updated and has since remained stable. The probable cause of the shift was particle contamination within the accelerometer which has disturbed the pendulum and caused a small shift from the nominal null position, thus affecting both the x and y outputs. Other possible causes for the problem are the accelerometer flex tape or transformer intermittency. IMU 3, serial number (S/N) 012, was removed from the vehicle and shipped to the JSC Inertial Systems Laboratory (ISL) for special testing.

The bias shift was verified and the up axis drift noted preflight was confirmed at 0.075 deg/hr. The IMU is now at the vendor for analysis, testing and repair.

The cause of the bias shift is unknown at this time.

Conclusions: The IMU x and y accelerometer bias shifted and then remained stable. The cause of the bias shift is unknown pending results of analysis and tests at the vendor.

Corrective action: IMU 3 S/N 012 was removed and replaced by S/N 016.

APPROVED

A. Cohen
A. Cohen

6/11/82
Date

Effect on subsequent missions: None

S. L. Bachman 6/11/82 *R. J. Ward 6/11/82*
Personnel assigned: S. L. Bachman/EH6; R. J. Ward/WA3

Resolution: CLOSED for STS-4 6/11/82

Statement of problem: RMS Ready-for-latch Indicator Switch Failed to Transfer.

Discussion: The RMS aft pedestal system 1 ready-for-latch switch failed to transfer when the RMS was cradled at 82:20:52 G.m.t. Only the system 2 switch transferred when the wrist section of the arm contacted the pedestal. The aft pedestal temperature was -40° F when the switch failed to transfer. In subsequent manipulator arm operations at higher temperatures, both system 1 and 2 switches operated properly.

Actuation of either redundant switch gives the required indication on the display and control panel. Failure of both indicator switches would not affect arm stowage or subsequent manipulator operations. The crew can determine that the RMS has been properly cradled by a comparison of visual marks on the RMS and the pedestal.

All indicator switches operated properly during RMS post-flight tests and checkout. In systems qualification tests, the ready-for-latch switches operated satisfactorily at -100° F. The failure may have been caused by friction in the switch actuation mechanism.

Conclusions: The RMS aft pedestal system 1 ready-for-latch switch failed to transfer on only one occasion when the wrist section contacted the aft pedestal during cold system operations. The failure was not repeated during subsequent in-flight operations or post-flight testing. The cause of the single failure to transfer has not been determined.

Corrective action: No discrepancies were found during post-flight inspections, test or checkout. No work-around is required if an RMS indicator switch fails to transfer again.

OK 5/25/82
APPROVED
RJB
5/25/82

A. Cohen
A. Cohen

5/25/82
Date

Effective on subsequent missions: None

R. B. West 5/26/82 *R. J. Ward* 5/26/82
Personnel assigned: R. B. West/EW3; R. J. Ward/WA3

Resolution: CLOSED 5/26/82

Statement of problem: The video tape recorder (VTR) cassette jammed in the recorder while the crew was removing the cassette.

Discussion: After the cassette jammed the crew was instructed to cut the tape to remove the cassette. The VTR requires approximately 10 seconds to release the tape from the recorder drum mechanism when the recorder is stopped. If the "eject" or power "off" buttons are pushed before the tape is released, the tape will be stripped from the cassette and jam the VTR. The STS-3 flight crew confirmed they had removed the cassette too soon. A hardware modification has been designed to physically prevent cassette ejection before tape release and if approved could be added at a later date.

Conclusions: The VTR cassette jammed because the crew removed the cassette before it had been released by the recorder mechanism.

Corrective action: The ten-second delay requirement will be emphasized to each flight crew during pre-mission training.

APPROVED

Aaron Cohen
A. Cohen

5/19/82
Date

Effective on subsequent missions: *5/15/82* None

Personnel assigned: C. Hyman/EE2 x6154; C. R. *Ryan Edmiston* /EE2 x4191 *5/17/82*

Resolution: CLOSED 5/13/82

Statement of problem: Waste Management System (WMS) Slinger Speed Erratic and Circuit Breaker Opened

Discussion: A used emesis bag was placed in the WCS per crew procedures prior to first commode use. The slinger circuit breaker opened at the end of the first commode use. The crew closed the circuit breaker (CB) and tried the slinger in the high speed mode (feces). The CB reopened. The slinger was then tried in the low speed mode (emesis). It ran at about 60 rpm instead of the normal 200 to 300 rpm and continued to degrade until day 6 when the slinger completely stopped.

Post-flight inspection found the emesis bag caught on a bottom tine with the bag velcro closure and plastic insert completely torn off. The bag was wrapped around the slinger filter 1-1/2 times causing the excessive motor current which eventually stopped the slinger.

Since air flow through the commode started prior to slinger activation, the bag could have been drawn to the slinger prior to its start. The commode design is not compatible with the present emesis bag.

Conclusions: The commode design is not compatible with the present emesis bag as the current bag material does not lend itself to shredding by the slinger.

Corrective action: A modification has been incorporated which results in the slinger being activated after commode repress but before the air flow starts. The purpose of this modification is to prevent materials from impeding the slinger while it is rotating.

A procedure change has been implemented which will have the emesis bag stowed in the wet trash. These changes are incorporated for STS-4 and subs.

Additional changes to improve WCS operations for STS-5 are being considered.

APPROVED

7515
5-10-82

Aaron Cohen
A. Cohen

5/10/82
Date

Effective on subsequent missions: None

Henry A. Rotter
Personnel assigned: H. A. Rotter/EC3; R. J. Ward/WA3

Resolution: CLOSED for STS-4 5/10/82

Statement of problem: CCTV Camera "C" Failed

Discussion: The crew compartment TV circuit breaker number 34 on panel R15 tripped when the aft bulkhead TV camera "C" was activated. A component failure inside the power supply resulted in an electrical short. The RMS elbow camera on STS-2 and the RMS wrist camera on STS-3 experienced the same failure mode. In each case, the camera temperatures were approximately 0° C.

Failure analysis and tests have determined that the failures were caused by the erratic operation of the camera dc/dc power converter at cold camera temperatures near 0° C.

Cold temperatures decreased the base-emitter voltage of the switching transistor resulting in a decreased voltage at the collector. This created an unstable frequency within the dc/dc converter that generated high voltage spikes shorting the output transistor.

Conclusions: Cold temperatures decreased the base-emitter voltage of the switching transistor resulting in a decreased voltage at the collector. This created an unstable frequency within the dc/dc converter that generated high voltage spikes shorting the output transistor.

Corrective action: The input resistor has been changed to provide more voltage drive to the switching transistor. Diodes have also been added to protect the output transistor from high voltage spikes. All flight TV cameras were modified and tested prior to STS-4.

APPROVED

A. Cohen
A. Cohen

5/10/82

Date

Effective on subsequent missions: None

Personnel assigned: *RE* R. Edmiston/EE2; *RJW* R. J. Ward/WA3

Resolution: CLOSED 05/07/82

Statement of problem: Tiles and pieces of tiles were lost from the upper forward fuselage and upper body flap surface during STS-3.

Discussion: Twenty-one (21) tiles and 16 pieces of tiles were lost from the upper forward fuselage and 16 tiles were lost from the upper body flap surface.

Based on the launch photography, on-orbit TV and upon retrieval of many tiles and tile pieces in areas surrounding the launch pad, it has been concluded that the tiles were lost during ascent. No densified tiles or pieces of densified tiles were lost.

Pre-entry thermal and structural analysis showed no issue with safety of flight, but did show some potential for structural overtemperature. Post-flight inspection showed no structural overtemperature in the areas where tiles were lost.

Conclusions: Extensive post-flight testing and vehicle tile chemical analysis indicate that the probable cause of tile loss is a combination of poor bond to the filler bar, damage due to high traffic, taping damage, and possible excessive application of water-proofing agent. Excessive downhand applications (i.e., spraying in a downward direction) can affect the undensified tile-to-filler bar and "SIP" bonds.

Corrective action: All undensified tiles on the upper forward fuselage and the upper body flap will be densified prior to STS-4. Specific procedures for tape application and removal, to be performed by experienced personnel only, have been implemented. Downhand spraying will be minimized and procedures to minimize puddling have been developed.

APPROVED

A. Cohen
A. Cohen

6/11/82
Date

Effect on subsequent missions: None

R. L. Dotts 6/11/82
Personnel assigned: R. L. Dotts/ES3; C. Schomburg/ES5

Resolution: CLOSED 6/11/82.

Statement of problem: ARPCS GN₂ usage excessive.

Discussion: During STS-3, a GN₂ leak in the Atmospheric Revitalization Pressure Control System (ARPCS) was isolated to System 2 downstream of the regulator inlet valve but outside the cabin upstream of the X₀ 576 bulkhead penetration. The leak was stopped in flight by closing the System 2 regulator inlet valve. The leak was temperature dependant occurring only in a cold attitude. The first leak started at a structural temperature of -45° F stopping as the vehicle warmed during PTC. The leak repeated at -25° F in conjunction with GN₂ flow induced by a potable water dump but stopped again after the vehicle warmed in a top-to-sun attitude. Tests at KSC at ambient temperatures did not show a leak.

Thermal predictions of the STS-4 tail-to-sun attitude indicate that the supply panel structural mounting temperatures will be about 15° F warmer than experienced on STS-3 which is about the temperature (-45° F) at which the leak started on STS-3. By closing the System 2 regulator inlet valve during the cold soak attitudes the leak can be stopped and there will be no effect on the mission.

Conclusions: The GN₂ leak occurred only during cold soak attitudes. And is between the System 2 regulator inlet valve and the cabin bulkhead penetration.

Corrective action: Crew procedures will be used on STS-4 to isolate the System 2 leak during the 65 hour tail-to-sun attitude. The OV-103 panel will be cold soak tested and then exchanged with the OV-102 panel after STS-4 or 5. The leaking GN₂ panel will be returned to the vendor for test and rework.

APPROVED

A. Cohen
A. Cohen

5/18/82
Date

Effective on subsequent missions: The GN₂ System 2 regulator inlet valve will be closed during cold soak attitudes on STS-4.

for Personnel assigned: *for* N. Prince/EC3; R. J. Ward/WA3

Resolution: CLOSED for STS-4 4/28/82

Statement of problem: All PDRS DAC (16mm) cameras inoperative except mid-aft

Discussion: Power to the cameras was not turned on until first required usage. Therefore, the cameras were exposed to a long cold soak during the tail-to-sun thermal tests without camera heaters being activated. Five of the six cameras did not operate when they were turned on. The sixth camera operated but popped a circuit breaker shortly after activation. The camera operated after the circuit breaker was reset and pulled the full roll of film, however 90 percent of the film was run off before the camera stopped. Post-flight examination of the cameras revealed blown fuses in the five cameras that did not operate. All the cameras were operational post-flight.

Post flight tests at KSC and JSC evaluation of the camera circuitry have not revealed a cause for the sixth cameras failure to stop when commanded.

Conclusions:

1. A cold start caused motor overloads in five camera, blowing fuses.
2. The camera which operated had a fuse which was apparently of higher current capability than the other five (operating on high side of specified tolerance).
3. The failure of the sixth camera to stop when shutoff by the crew is unexplained.

Corrective action:

1. Procedures have been initiated to turn camera power on early after attaining orbit to assure that camera heaters are active.
2. Fuses in the cameras have been by-passed. Circuit protection is provided by circuit breakers on the camera control panel.
3. A procedure to stop a camera by opening its circuit breaker if it does not stop normally has been proposed for addition to the PDRS operations checklist.

APPROVED

R. Machell
R. Machell

5/18/82
Date

Effective on subsequent missions: None

Personnel assigned: H. D. Yeates/ED24

h

C.H. Selach

5/18/82

Resolution: CLOSED 5/17/82

Statement of problem: WCS Vacuum Vent Valve knob came off.

Discussion: The knob on the vacuum vent valve came off during STS-3 when the valve was moved. The valve could not be closed for entry. The knob was retained by a flat tip set screw which does not exert enough force. The set screw could not be reset by the crew.

Conclusions: The set screw did not exert sufficient force to retain the knob.

Corrective action: Rollpins have been added to retain control knobs on the WCS.

APPROVED

A. Cohen
A. Cohen

5/11/82
Date

Effective on subsequent missions: None

Henry A. Rotter, Jr.
Personnel assigned: H. Rotter

Resolution: CLOSED 5/12/82

Statement of problem: CCTV camera "B" lens stuck in a zoom "out" mode during flight.

Discussion: Attempts by ground command and by the crew failed to correct the zoom anomaly. The camera (S/N 006) and lens (S/N 004) were removed from the spacecraft and returned to the CCTV contractor for failure analysis. The lens was demated from the TV camera and operated on a set of special test equipment. The zoom function was still stuck as previously reported. After approximately ten zoom "in" commands, the lens became unstuck and operated properly.

Since the initial flight failure occurred during a "cold" case environment, the lens was then exposed to 0° C, -20° C, and -40° C test environments. In each environment, the lens operated properly without any tendency to stick. After removal from the test chamber, the lens assembly cover was removed, A motor torque test was run, and lens drag torque was measured. Each function was with acceptable limits.

The lens and motor were removed from the assembly and returned to the vendor for inspection. Both units were disassembled and inspected for potential flaws. The vendors found no problem within the lens or the motor. The motor was temperature cycled at temperatures from -20° C to +60° C and checked for any electrical discontinuities. No problems were found.

This is the only zoom failure in a total of 10 camera positions flown during the past 3 missions. No failures of this type have occurred during ground testing. The use of the zoom feature for the cameras on STS-4 is not mandatory for RMS operations.

Conclusions: At this time, no specific flaw has been found to account for the flight anomaly.

Corrective action: The lens assembly has been replaced for STS-4. Testing at the vendor continues.

APPROVED

Aaron Cohen
A. Cohen

6/7/82
Date

Effect on subsequent missions: None

Ryan Edmiston 6/3/82
Personnel assigned: R. C. Edmiston/EE2; C. J. Walsh/WC3

Resolution: CLOSED for STS-4 6/2/82

Statement of problem: S-band transponder 2 (SN 304) failed in the low power mode (STDN/SGLS) in both the high- and low-frequencies.

Discussion: Transponder 2, along with power amplifier 1, was configured in the STDN-high power mode for the PDP (plasma dynamics package) EMI test at 84:21:46:20 G.m.t. When transponder 2 was configured back into the STDN low-power high-frequency mode at 84:21:46:20 G.m.t., the downlink was lost and a drop in received signal strength was observed. At 85:01:33:30 G.m.t., a test was conducted with transponder 2 in SGLS, high-frequency mode and downlink was lost. A test at 85:02:13:20 G.m.t. with transponder 2 in STDN low power low-frequency also provided no downlink.

After landing at White Sands, a test was run to exercise transponder 2 in the modes that it had failed in flight. As a result, the transponder returned to normal operation.

Transponder 1 (ref. problem 23) was removed, tested and inspected and found to have relays contaminated with epoxy and coating materials. The transponder 1 relays as well as those in transponder 2 are of the same lot, and therefore, it has been concluded that the problem is generic especially since cycling the relays cleared both units. Since the problem cleared with cycling and since the spare hardware has potentially the same problem, transponder 2 will not be replaced for STS-4.

Conclusions: The most likely causes are particles of epoxy and other contaminants in the RF control relays that interfered with the relay mechanism and fouled the relay contacts in zero gravity. Cycling the relays at White Sands in the 1g field cleared the problem by loosening and/or burning the contaminants on the contacts.

Corrective action: No action has been taken to replace suspect RF relays in transponder 2. If problem recurs in flight, repeated mode selections may clear fouled relays. In the future, the precapping inspection for contamination in the RF cavity of the relay will be conducted using higher resolution inspection optics.

 APPROVED

A. Cohen

5/20/82
Date

Effective on subsequent missions: None

 5-18-82
Personnel assigned: M. Luse/EE4; W. Seibert/EE3

Resolution: CLOSED 5/17/82

Statement of problem: RMS Encoder Check False Alarms

Discussion: When the RMS was powered up following the temperature monitoring mode during cold soak, the wrist pitch encoder issued false alarms at 83:15:21, 84:14:10 85:15:00 G.m.t. The encoder check is designed to catch discontinuities in encoder data of 1° or greater. The check is inhibited during power on initialization (POI), until the port arm select signal is received by the GPC. The manipulator controller interface unit (MCIU) delays the port arm select signal from the arm for 1 second after power is applied to the arm to allow for joint power conditioner (JPC) and servo power amplifier (SPA) power up and initialization. This is to insure that the GPC only processes valid data from the arm. The auto RMS mode is inhibited unless the encoder check is passed.

Prior to exiting the POI the encoder data read 0°. When POI is exited, the encoder data jump to actual angles. In the stowed position these are equivalent to joint angle biases. For the wrist pitch this is a value of 1.401° which exceeds the 1° limit in the encoder check. If the first valid encoder data are received after the port arm select flag is received by the GPC the encoder check is initialized to 0° and the next cycle exceeds the 1° limit.

At ambient temperature the JPC/SPA initialization occurs in about 900 milli seconds. Worse case analysis shows this initialization procedure could take up to 1.009 seconds at low temperatures.

All the false alarms occurred only when the arm was cold.

Conclusions: At low temperatures the POI sequence for wrist pitch exceeded the 1 second MCIU delay time for the port arm select signal. A false alarm was generated.

Corrective action: Fly as is for STS-4. If the wrist pitch encoder issues a false alarm after POI, the crew will release the auto mode inhibit by a reset of the encoder check through the aft orbiter keyboard.

Proposed modifications include changing the SPA timing, increasing the delay in the MCIU or delaying the port arm select signal in the GPC software.

APPROVED *[Signature]* 5/26/82
[Signature] A. Cohen Date

Effective on subsequent missions: False encoder alarms may be encountered if the arm is powered on at low temperatures. The crew will reset the encoder check through the aft orbiter keyboard.

Personnel assigned: *[Signature]* J. C. Peck/MP; *[Signature]* R. J. Ward/WA3 5/25/82

Resolution: CLOSED 5/25/82

Statement of problem: S-Band transponder 1 (SN 301) failed in STDN high- and low- power and high-frequency modes

Discussion: While troubleshooting the failed transponder 2 problem (problem No. 21), a ground command was executed at 85:03:43:55 G.m.t. to select transponder 1 in the STDN high-power, high-frequency mode and the downlink was lost. At 85:03:45:00 G.m.t., the crew was asked to configure transponder 1 in the STDN low-power, high-frequency mode and downlink was again lost. Degradation in the uplink received signal strength was also noted.

At 85:18:13:25 G.m.t., transponder 1 was again selected in the STDN high-power, high-frequency mode and cycled through the transmit/receive and receive-only modes, but the downlink was never recovered.

A postlanding test was conducted at White Sands and the transponder 1 failed modes were recovered. Transponder 1 was removed from OV-102 and returned to the vendor. Contamination in the form of epoxy and coating material was found in the RF cavity of all the RF control relays S1, S2, S3, and S4. Loss of these relays would have resulted in the failure symptoms observed during the flight.

Conclusions: The most likely cause is particles of epoxy and other contaminants in the RF control relays that interfered with the relay mechanism and fouled the relay contacts in zero gravity. Cycling the relays at White Sands cleared the problem by loosening and/or burning the contaminants off the contacts in a 1g field.

Corrective action: The defective relays were removed and replaced with relays from a different vendor. In the future, the pre-capping inspection for contamination in the RF cavity of the relay will be conducted using higher resolution inspection optics.


APPROVED

A. Cohen

5/20/82
Date

Effective on subsequent missions: None


Personnel assigned: M. Luse/EE4 x5351; W. Seibert/EE3 x5391

Resolution: CLOSED 5/18/82

Statement of problem: APU 3 fuel tank pressure decay.

Discussion: APU 3 fuel tank pressure decayed 12 psi during the STS-3 flight. A GN₂ QD (quick disconnect) was verified leaking at tank residual pressure. An analysis of the failed QD showed indentations and 2 flakes of rust on the poppet sealing surfaces. Also, there were indentations and smaller rust particles on the QD cap. A fine coating of rust was evident throughout the QD and a nylon-type fiber was found on the QD filter. The source of contamination is not known, but possibilities include moisture from the air, the ground half coupling, or freon cleaning GSE. Until recently, the GSE was cleaned with freon. Freon in the presence of hydrazine is known to attack stainless steels. This method of cleaning has been discontinued at KSC.

Conclusions: The pressure decay in APU 3 fuel tank was caused by a leaking GN₂ QD. The leakage was due to contamination and indentations on the QD sealing surface. The contamination could have come from humidity or an improper cleaning method.

Corrective action: The QD has been replaced and the new QD leak checked by mass spectrometer. The use of freon to clean the GSE has been discontinued.

APPROVED *Aaron Cohen* 6/11/82
 A. Cohen Date

Effect on subsequent missions: None

Personnel assigned: *R. J. Ward* 6/11/82
 Renee J. Lance/EP4; R. J. Ward/WA3

Resolution: CLOSED 6/11/82

Statement of problem: Loss of transmit function on CDR's Wireless Crew Comm Unit (WCCU Unit A)

Discussion: During the 5th day (086:14:16:00 G.m.t.) the CDR reported that he had lost the capability to transmit on his WCCU but could still receive. The battery was replaced but the problem was not cleared. The spare WCCU (Unit C) was deployed and communications were restored.

Post flight analysis showed that the failure was caused by a broken wire in the CDR's leg unit antenna. Several antennas were severely bent when stowed for landing during STS-2.

Conclusions: The antenna wire break was caused by repeated bending and straightening of the antenna which occurs when the crewman moves about the cabin and snags his antenna.

Corrective action: Antennas with a history of severe bends will be replaced and the remaining antennas will be reworked by moving the shrink-on ID label to an area where it will provide strain relief.

APPROVED

R M Machel
R. Machel*5/10/82*
Date

Effective on subsequent missions: None

R W Armstrong 5/17/82
Personnel assigned: R. W. Armstrong/EE3 x5435

Resolution: CLOSED 5/14/82

Statement of problem: Cap seal on several drink bags failed.

Discussion: During flight the crew reported the failure of the cap seal on several beverage packages. Packages sealed at the same time as those used in flight were rehydrated in the JSC Flight Food Processing Laboratory. The flight failure was repeated as the heat seal intended to secure the dispenser valve assembly to the beverage body yielded. The failure occurred when the package was slightly overfilled. The rupture in the cap seal was from 1/8 to 1/4 inches long and permitted about 1 ounce of liquid to escape. By pulling slightly on the cap it would easily be removed from the package. The complete beverage assembly was designed to withstand internal pressure in excess of 20 psi.

A checkout of the heat sealing equipment used in package fabrication revealed that the temperature controller was cutting out the heater element 50 degrees below the minimum required to produce the optimum heat seal.

Conclusions: The cause of the heat seal failures is a faulty temperature controller in the manufacturing process which failed to maintain the proper heat seal temperature.

Corrective action: The beverage sealing equipment was overhauled and the temperature controller cleaned, reset and tested. A safety temperature sensor/switch was installed in the unit to render the equipment inoperative below the minimum heat seal temperature. The optimum heat sealing range was highlighted on the temperature indicator to accentuate the proper sealing temperature to the operator.

APPROVED

R M Machell 4/28/82
R. M. Machell Date

Effective on subsequent missions: During beverage packaging operations for STS-4 inspections of the heat seal area will be increased to 100%. For STS-5 and subs a new packaging system similar to the skylab plastic food liner will replace the current skylab accordion type beverage container.

Personnel assigned: *RT* R. L. Sauer/SE3; *WJ* R. J. Ward/WA3

Resolution: CLOSED 04/26/82

Statement of problem: DFI wideband mission recorder bite no 1 failed at 86:14:40 G.m.t. during STS-3.

Discussion: The recorder was operated during all subsequent planned record cycles and all other telemetry was normal until AOS following blackout. At that time, the tape motion telemetry became intermittent, indicating that the recorder was operating at less than 7 1/2 ips. At approximately 3 minutes prior to touchdown, the bite 2 (record indicator) telemetry also indicated fail.

Data playback was done at JSC and the entry data was unrecoverable. The recorder was returned to the vendor and a failed lamp was found in the tape tension servo circuitry. This failure accounts for the observed telemetry indications. The failed lamp is rated by the manufacturer for dc operation at 500 hours. Operations time on this recorder was estimated to be 300 hours. A filament inspection showed that the filament failure was due to a complex grain notching phenomena which is common in lamp failures. Grain boundary notches form on the filament and cause localized heating which in turn aggravates the notching and ultimately results in failure.

The lamps in this recorder will be replaced with 2500 hour rated lamps selected for uniform current characteristics. The power transistors in the motor drive circuits will be replaced as a precautionary measure.

Conclusions: This problem was caused by the premature failure of a lamp in the tape tension servo control circuit.

Corrective action: For STS-4 the recorder has been replaced with the backup recorder, a refurbished qualification test unit. The lamps in the replacement recorder are the 500 hour rated lamps. The estimated operation time on this recorder is 120 hours well below the 500 hour rated lamp life time.

APPROVED

A. Cohen

Date

Effective on subsequent missions: None

Personnel assigned: J. F. Melugin/ED5; C. J. Walsh/WC6

Resolution: CLOSED 5/20/82

Statement of problem: The S-band power amplifier 1 telemetry measurement indicated that the power output degraded from 130 watts to 103 watts during the mission.

Discussion: A slow reduction in output power from power amplifier (PA) 1 was observed after it was selected for the plasma diagnostic package (PDP) EMI experiment and later when it was selected with transponder 2 in the STDN HI power mode. Most of the power drop (about 6 watts) occurred in the first 5 or 6 hours of operation and then settled down to a loss of 4 or 5 watts per day.

The PA assembly was removed post flight for failure analysis. The traveling wave tube (TWT) exhibited the same loss of power as observed during the flight but never degraded below 100 watts. The vendor has completed extensive tests on the TWT and will tear it down for a detailed metallurgical inspection. The TWT has a design life of 20,000 hours but degraded after only about 1000 hours of operation. The cause of the failure is unknown at this time.

If a TWT fails or degrades on STS-4, either the back up power amplifier or the low power mode can be selected.

Conclusions: The PA 1 TWT power output degraded below specified limits after several hours of operation. The cause of the failure is unknown at this time. Failure analysis continues at the vendor.

Corrective action: The defective PA assembly was replaced by the spare unit with about 550 hours of operating time on the TWT.

APPROVED

A. Cohen
A. Cohen

6/11/82
Date

Effective on subsequent missions: None

M. B. Luse 6-11-82 Robert J. Ward 6/11/82
Personnel assigned: M. Luse/EE4; W. Seibert/EE3; R. J. Ward/WA3

Resolution: CLOSED for STS-4 6/11/82

Statement of problem: Cathode Ray Tube 1 did not respond to entries

Discussion: Cathode ray tube 1 did not respond to entries. The crew performed a malfunction procedures which indicated a stuck keyboard switch might be the problem. A replacement switch from the aft keyboard was substituted and the problem cleared.

This problem has been seen in ground facilities when switches of that specific vintage have been used for several years. The fundamental problem is attributed to a "weak" spring which has chosen to minimize zero-g depression forces. An improved design is being phased in on an attrition, not mandatory basis.

Conclusions: The problem was caused by a weak spring in a keyboard switch.

Corrective action: The present attrition-based plan will continue.

APPROVED

Aaron Cohen
A. Cohen

5/11/82
Date

Effective on subsequent missions: None

Personnel assigned: P. Sollock

by Ed Hood

Resolution: CLOSED 5/12/82

Statement of problem: SPA (servo power amplifier) commutator failure message on RMS wrist yaw joint.

Discussion: A SPA commutator failure message for the wrist yaw joint was received 12.5 seconds after selecting RMS port arm at 87:16:09:25 G.m.t. The arm-based electronics (ABE) warning annunciator remained on until the arm was deselected and powered down 20 seconds later. The commutator scanner BITE alarm did not recur on subsequent power ups of the RMS during flight.

The commutator scanner detects motor position and transmits these data through a three-bit code to the motor control electronics. These signals are combined with motor speed commands to cause the necessary phase switching of the three phase motor. The commutator scanner BITE circuit is designed to detect invalid codes from the commutator scanner. If an error is detected, the BITE flag is latched within the SPA and can only be reset by cycling the power.

The commutator scanner BITE alarm did not repeat and proper motor operation was demonstrated during extensive post-flight testing. The cause of this anomaly is unknown. A signal transient probably upset the commutator scanner BITE circuit latching the BITE flag. A 4 to 5 microsecond signal transient can trip the commutator scanner BITE circuit.

A single false or transient BITE signal can be reset by cycling power to the RMS. A continuous false BITE signal maintains the ABE warning light on, but RMS operations can be continued. After a true commutator scanner failure the associated motor would fail to drive in the primary mode because of improper phase switching commands. However, the arm could be cradled in the backup mode.

Conclusions: The cause of this anomaly is unknown.

Corrective action: Turnaround testing has demonstrated proper operation of all RMS modes and functions. A proposed redesign would make the commutator scanner Bite circuit self-resetting for reduced sensitivity to signal transients

APPROVED

A. Cohen
A. Cohen

6/7/82
Date

Effect on subsequent missions:

None

Personnel assigned: *J. C. Peck* J. C. Peck/MP; *R. J. Ward* R. J. Ward/WA3

Resolution: CLOSED 6/4/82

Statement of problem: Erratic flash evaporator Primary B controller operation and shutdown during full-up (topping and high-load) operation.

Discussion: During the payload bay door closure test on day 6 (approximately 87:23:21:00 G.m.t.), the flash evaporator system (FES) Primary B controller exhibited erratic temperature control while operating in the full-up mode for approximately 30 minutes, resulting in a system shutdown. The evaporator outlet temperature cycled between 39° F and 56° F prior to shutdown.

The Primary B unit mid-point temperature sensor which provides "startup" and "standby" functions as well as an "anticipator" input to the controller was known to have a slower response time than the other sensors. This response time has been used in the Hamilton Standard analytical FES performance model resulting in a predicted erratic operation very similar to that observed on STS-3. The Primary B controller was removed and has completed checkout at the vendor with no indication of controller problems.

Conclusions: The erratic operation of the Primary B controller was caused by the slow response of the mid-point temperature sensor. The FES is fully operational for STS-4.

Corrective action: The Primary B mid-point temperature sensor has been replaced on OV-102 and the new installation has been verified to have a proper time constant.

APPROVED

A. Cohen
A. Cohen

5/24/82
Date

Effective on subsequent missions: None

J. D. Rosen 5/21/82
Personnel assigned: J. D. Rosen/EC3

Resolution: CLOSED for STS-4 *5/25/82*

Statement of problem: The payload main bus C talkback did not operate.

Discussion: During the STS-3 flight, the payload main bus C talkback (DS17 PNL R1A1) did not operate when the power was turned on. Power was on as indicated by the downlink measurement and the bus current. Troubleshooting after the flight confirmed the inflight data. Teardown of the talkback by the supplier disclosed an open at a coil lead solder joint. The cause of the failure was a bad (cold) solder joint.

Accumulative environmental effects including vibration and thermal cycling of the talkback probably caused the cold solder joint to open. This is the first occurrence of this type of problem with talkbacks. This failure is an isolated incident since almost 200 talkbacks have operated satisfactorily on OV-102.

Conclusions: The talkback did not operate because a cold solder joint failed at a lead to the coil due to accumulative environmental effects. This failure is an isolated incident in the soldering process.

Corrective action: The talkback has been replaced. The vendor will review his soldering processes to determine if any action can be taken to improve reliability.

APPROVED

705
6-4-82

A. Cohen

Aaron Cohen

6/7/82

Date

Effect on subsequent missions: None

Robert J. Ward *6/4/82*

Personnel assigned: A. J. Farkas/EH5; R. J. Ward/WA3

Resolution: CLOSED 6/4/82

Statement of problem: Smoke detector 2A had a continuous alarm.

Discussion: Smoke detector 2A in avionics bay 2 failed on with a continuous false alarm. It was reset by the crew twice with the same result. The circuit breaker for this detector was opened.

The false alarms occurred again at White Sands when the smoke detector was powered, but did not recur at KSC.

During troubleshooting at the vendor, the problem could not be repeated. Additional tests are in process.

Conclusions: At this time, no cause for this problem has been found during tests at the vendor. Testing will continue.

Corrective action: The smoke detector has been replaced for STS-4. Should this problem recur, the redundant smoke detector in the bay would detect a fire and should it fail a fire bottle would be discharged in the bay prior to entry.

APPROVED

A. Cohen
A. Cohen

6/11/82
Date

Effect on subsequent missions: None

4/11/82
FOR HAR
J. R. Walsh
Personnel assigned: H. Rotter/EC3; C. J. Walsh/WC6

Resolution: CLOSED for STS-4 6/11/82

Statement of problem: Midbody motor control assembly (MMCA) 3 operational status 4 was "0" but should have been "1".

Discussion: The problem occurred after the last vent door sequence prior to deorbit, indicative of a latched "on" hybrid relay. Post-flight data review disclosed that coincidental with this problem, left vent door 3 motor 1 continued to run even after the motor 1 close inhibit signal had been received by the MMCA 3 series hybrid relays, K4 and K18. Hybrid relay K4 is in the MMCA 3 operational status 4 "Daisy Chain". K4 remained latched even after the MDM control signal had been removed. Left vent door 3 troubleshooting at KSC did reproduce the problem one time. The box was removed from OV-102, and troubleshooting was performed on the box at Rockwell with no success in duplicating the problem. Relays K4 and K18 were replaced as a precautionary measure and for additional investigation. Extensive detailed failure analysis on both relays could not identify any anomalies.

The problem could not be reproduced at the box level or the relay level. The failure symptoms indicate that the most probable cause was an intermittent breakdown on the hybrid module substrate between the inhibit input and the relay control coil causing the relay to remain latched without the presence of a control signal. A component breakdown and failure analysis of the hybrid module substrate is continuing at the vendor for the specific details of this unique substrate failure mode.

There are over 550 of these hybrid relays in use on the vehicle and this the first failure of this type seen in the flight test program. This problem was a unique failure of a hybrid module substrate. The faulty operation of a hybrid relay is detectable in flight.

Conclusions: A latched "on" hybrid relay caused left vent door 3 motor 1 to continue running even after the motor 1 close inhibit signal was received. The most probable cause was an intermittent breakdown on the hybrid module substrate between the inhibit input and the relay control output.

Corrective action: Suspect hybrid relays K4 and K18 were replaced and the box was retested satisfactorily on OV-102.

APPROVED

Aaron Cohen
A. Cohen

6/11/82
Date

Effect on subsequent missions: None

Sped. White *Robert J. Ward* *6/11/82*
Personnel assigned: L. Moon/EH5; R. J. Ward/WA3

Resolution: CLOSED 6/11/82

Statement of problem: Ammonia boiler controllers A and B failed.

Discussion: ABS (ammonia boiler system) "A secondary" controller failed to control the temperature when activated at 89:16:21 G.m.t. Then the "B secondary" controller failed to activate. The problem was reproduced during troubleshooting at White Sands.

Disassembly of the system A isolation valve at the vendor revealed a large amount of ammonium chloride contamination under the seat and this reduced the valve stroke and increased the pressure drop. The source of this contamination is unknown, but it may have been induced during valve assembly.

The system B isolation valve was contaminated with iron rust on the side of the sliding poppet and on the wall of the barrel in which the poppet slides. The rust probably prevented the valve from opening when first activated and caused "sticky" valve operation during subsequent troubleshooting. The source of the rust is unknown. The valve had not been operated for about 7 months before flight while the ABS was fully serviced. Ammonia has a high affinity for water and should inhibit rust formation after servicing.

Conclusions: Ammonium chloride contamination in the system A isolation valve pushed the poppet seat partially out of the groove in which it rests. This resulted in a shortened valve stroke and an increased differential pressure.

Iron rust in the system B isolation valve most probably prevented the valve from opening when activated and caused "sticky" valve operation during troubleshooting. The source of the ammonium chloride contamination and the rust is unknown.

Corrective action: The isolation valves have been replaced and the ammonia has been checked to verify the absence of contaminants for STS-4. Valve assembly, storage, and shipping procedures are being reevaluated to minimize the possibility of contamination or rust.

APPROVED *A. Cohen*
 121642 A. Cohen

6/7/82
 Date

Effect on subsequent missions: None

J.D. Rosen 6/4/82 Robert J. Ward 6/4/82

Personnel assigned: J. D. Rosen/FC3; R. J. Ward/WA3

Resolution: CLOSED for STS-4 6/4/82

Statement of problem: Loss of radiator "latch" indication.

Discussion: During orbiter descent at GMT 89:16:01:23, 3 minutes and 23 seconds prior to touchdown, the port radiator forward panel latches 1 through 6 systems 1 and 2 together with the starboard radiator aft panel latches 7 through 12 system 2 did not indicate "latch" or "release". After landing, latch operation returned all the "latch" indications in 0.7 seconds compared to a normal latch operating time of 25 to 30 seconds. This indicated that all the latches had been locked during entry.

Motor current data and the crew debriefing indicate that the Control and Display (C&D) Panel control switch was moved from the "latch" position, through the "off" position and momentarily (0.5 - 0.7 seconds) to release, then to "off". At this point, the talkback for the starboard radiators indicated a not latched posture with the down list data showing loss of "latch" position on the starboard radiator latch actuator forward panel (latches 1-6). The panel control switch was then returned to the "latch" position, proper talkback was received, and the switch was returned to "off". In moving the panel control switch to the release position momentarily, all four latch actuators were driven toward an unlatched position; however, because of the short drive time, only the switches in the starboard forward panel actuator transferred. Therefore, when the panel control switch was returned to the "latch" position, the starboard forward actuator was the only one of the four actuators capable of being powered toward the latched direction. This left the three remaining actuators in a position such that the actuator control/indicating switches were "near" their transfer point.

Conclusions: The switches transferred during entry because of either vibration or a temperature increase causing the system to backdrive and the switches to transfer. Because of the overcenter position of the latch linkage and the latch mechanical stop, the system remained in a safe, completely latched position. This design/rigging feature assures that once the crew observes the proper talkback indication, with the panel switch in the "off" position, the system will remain safely latched for entry.

Corrective action: None required - System design and crew procedures are adequate to assure a safe latch posture for entry.

APPROVED

Sidney Cohen
A. Cohen

5/10/82
Date

Effective on subsequent missions: None

Personnel assigned: *N. Jervas 5/7/82* *A. J. Ward 5/7/82*
N. Jervas/EW3; R. J. Ward/WA3

Resolution: CLOSED 5/10/82

Statement of problem: MDM FF4 Input/Output Error

Discussion: During entry preparations, all redundant set general purpose computers reported a one-time input/output error (word-gap time out) against MDM FF4. This was coincident with MEC power on and a not unexpected "MIA Burp" on the FA MDM's. Post-flight interrogation of FF4's BITE status register indicated an attempted simultaneous access on both data bus ports. This would invoke internal interlock and cause the MDM to terminate whatever legitimate response was in progress.

Testing in FSL (Flight Software Laboratory) and at the MDM vendor confirmed that the MEC power-on event as seen by the MDM (MIA burp) can trigger MDM address detection logic to look for its address, in this case a 1010 hexadecimal pattern unique to this MDM. This occurrence on the MDM secondary port simultaneous with its legitimate output on the primary port causes the MDM bite bit and a termination of its legitimate output. The output termination is sensed by the GPC's and causes an input/output error for 1 cycle only. The next GPC cycle will be clear of any fault and therefore no impact is evident in the GPC functions.

Conclusions: Testing in FSL (flight software laboratory) and at the MDM vendor indicates the "MIA Burp" signature on FA4 bus could confuse the MDM to the extent that it would "see" a legitimate access attempt. In response, it then terminates an ongoing response on the FF4 bus thereby causing a GPC input/output error.

Corrective action: None. One time input/output errors are not detrimental to overall performance. The ground would see the indicated input/output error but the crew would not.

APPROVED

A. Cohen
A. Cohen

5/24/82
Date

RBB

CT

Effective on subsequent missions: None

P. Sollock
Personnel assigned: P. Sollock

5-21-82

Resolution: CLOSED 5/20/82

Statement of problem: Electrical Leaks in Main Propulsion System Engine 3 LOX prevalve Remote Power Controller C

Discussion: While on-orbit, a LOX prevalve measurement (PCA-MPS 3 LOX prevalve close RPC C) came on, and remained on, indicating either a failed on or leaking RPC. This RPC provides power thru a series driver to a solenoid operated valve in the He system which pneumatically closes the MPS #3 prevalve.

Initial postflight testing could not duplicate the problem. However, after about 1 week of power-on testing, the measurement came on again. Testing verified that the valve was in the open position and that the remote power controller (RPC) had not failed on. The RPC was leaking (shorting) internally, causing the failed-on indication. Internal leaks of the RPC's have been experienced on Orbiter 102 because of the sensitivity of the Mark I transistors to humidity. A shimmed mounting technique that will retain RPC hermeticity and Mark II transistors that have a thick dielectric layer over the collector base junction will be used on Orbiter 099 and subsequent and on all replacement units for OV-102.

Should this leaking RPC go to a failed on condition the series driver prevents the LOX prevalve from closing. If the series driver also fails on the pre valve is prevented from closing during engine main stage by the pneumatic pressure applied with the same type control system on the LOX prevalve open side. Therefore, adequate redundancy exists even if the leaky RPC degrades further.

Conclusions: The RPC is electrically leaking but not enough to supply power to close the solenoid.

Corrective action: If the RPC goes to a continuous "on" state it will be retested after STS-4 to verify that it has not degraded further to a "failed on" condition.

APPROVED

A. Cohen
A. Cohen

5/10/82
Date

Effective on subsequent missions: None

Personnel assigned: *L. J. Moon* *5/10/82* *R. J. Ward* *5/10/82*
L. J. Moon/EH5; P. Cota/EP2; R. J. Ward/WA3

Resolution: CLOSED for STS-4 5/10/82

Statement of problem: High MPS helium usage during entry.

Discussion: The post landing LOX side MPS purge ended in 1 hour rather than the predicted 5 hours. The flow rates were high and the MPS 3 inlet pressure matched the LOX manifold pressure during manifold repressurization although it should have been much less. A similar flow rate to that observed during the post-landing purge was seen during a prelaunch external tank blowdown test. The high prelaunch flow rate was stopped by closing valves upstream of the POGO recirculation check valves.

Troubleshooting isolated the problem to the MPS 3 POGO recirculation check valve. X-ray confirmed the valve had an open flapper. The valve was removed and found to have a broken spring. The spring had been installed backwards causing the spring to fail due to adverse spring loading.

A similar failure on STS-4 would result in high He usage during entry or the possibility of undesirable oxygen leakage with one engine shut down during ascent.

Conclusions: The MPS 3 POGO recirculation check valve was failed for the entire flight due to a broken internal spring. Adverse spring loading due to improper installation caused the spring to fail.

Corrective action: The failed check valve has been replaced and leak checked for STS-4. Valve springs in spare units have been inspected to verify proper spring installation. Inspection procedures now require definite verification of proper spring installation during assembly.

APPROVED

WTM

Aaron Cohen
A. Cohen

6/11/82

Date

Effect on subsequent missions: None

Ken Hall 6/11/82 *Sp/Dir 6/11/82*
Personnel assigned: K. Kroll/EP2; C. J. Walsh/WC6

Resolution: CLOSED 6/11/82

Statement of problem: The crew observed the "NO VIDEO" light on the VTR at various times within the mission.

Discussion: The illumination of the "NO VIDEO" light is intended to indicate the absence of video at the VTR (Video Tape Recorder). However, the crew played back several tapes which were recorded when the "NO VIDEO" was on and verified good recordings. Post-mission tests have determined that a potentiometer within the video level detection circuitry was not properly adjusted.

The setup procedures specify a fully clockwise setting of the video level potentiometer. It was set fully counterclockwise on this VTR, apparently a technician error during initial setup.

Conclusions: The problem was caused by an improperly adjusted potentiometer in the video level detection circuitry.

Corrective action: The VTR video level potentiometer has been readjusted for STS-4.

APPROVED

A. Cohen

5/18/82
Date

Effective on subsequent missions: ~~5/17/82~~ None

Personnel assigned: C. Hyman/EE2 x6154; C. R. Jamiston/EE2 x4191

Resolution: CLOSED 5/13/82

Statement of problem: Evaluation of high winds aloft on "wave-off" day and landing with a high touchdown velocity, a lateral offset error, and a pitchup during derotation.

Discussion:

Winds aloft: Although the "wave-off" on the seventh day was based on excessive surface winds, the winds aloft were also excessive. It was doubtful that the planned approach geometry with the extreme winds above 25,000 feet would have provided a manageable energy situation. Simulations have been conducted to evaluate procedures and system capability for accommodating various extreme wind conditions. The result is a set of placards that relate the wind magnitude and direction to the direction of the heading alignment turn. This will insure that the resulting trajectory is within the vehicle energy management capability.

Touchdown velocity: The velocity at touchdown was about 225 knots estimated air speed rather than the desired 195 knots. At crew takeover from autoland, the airspeed was 6 knots greater than nominal. Simulations indicate that autoland would have flown the vehicle to touchdown at 201 knots. The planned late takeover from autoland did not provide sufficient time for the pilot to feel the vehicle response. Attempts to make some minor trajectory adjustments resulted in a touchdown sooner than intended at a higher than planned airspeed. Future flight procedures will plan the takeover prior to preflare or let the autoland go to touchdown.

Offset error: The landing offset, some 28 feet to the right of the runway centerline, was a known problem with the MSBLS antenna system and is acceptable for lakebed landings. A hardware modification to the ground antennas will be made prior to STS-5 to eliminate the offset.

Derotation pitchup: This problem was aggravated by the high landing velocity. The derotation after main gear touchdown was allowed to start at too high an airspeed and required the pilot to try to stop it at too low a pitch angle. The rapidly changing elevator trim requirements made it difficult to avoid over-controlling in this situation. The way to avoid it is to concentrate immediately after touchdown on keeping the nose up until the vehicle slows to 180 knots. Flight procedures will be more explicit for STS-4 and the SMS simulation of this situation is being improved to provide better crew training.

Conclusions: The actual "wave-off" was based on excessive surface winds, but a "wave-off" due to excessive winds aloft was also required. The planned late takeover from autoland did not allow the pilot time to get the feel of the vehicle, resulting in a premature touchdown at high speed. The landing offset was a known problem with the MSBLS antenna system and is acceptable for lakebed landings. The pitchup during derotation resulted from starting the pitchover above 180 knots where over-controlling is difficult to avoid once the nose drops below about 3 degrees.

Corrective action: The new wind placards will be adopted for STS-4 and subsequent flights. On future missions, manual takeover from autoland will not be planned between start of preflare and touchdown. MSBLS antennas are being modified prior to STS-5 to eliminate the lateral offset error. Procedures will be more explicit, the SMS updated, and more specific crew training provided in the derotation maneuver, especially for high-speed landings.

APPROVED Aaron Cohen 5/24/82
A. Cohen Date

Effective on subsequent missions: ^{RCB} None DL Gilbert 5/21/82 RJ Ward 5/23/82

Personnel assigned: D. W. Gilbert/EH4; R. J. Ward/WA3

Resolution: CLOSED 5/21/82

Statement of problem: Early Shutdown of Debris Camera

Discussion: The debris camera shutdown approximately 1.5 seconds after SRB ignition. The camera controller electronics were designed to remove battery power from the camera after approximately 3.5 minutes.

Postflight tests of system elements identified a properly operating camera, adequate remaining battery capacity, and a "timed out" condition of the controller. The 10 seconds of exposed film contained satisfactory imagery with no indication of a camera problem.

Vibration testing was performed. During the first test the camera stopped, and the power connector to the camera was found demated. A second vibration test was successfully performed without identifying any cause for the early shutdown. The connector involved was a Deutsch DM 9702-195 push/pull type of the same generic family used on Apollo utility outlets.

Further bench testing of the system identified the condition of camera operation for a time period of approximately 10 seconds, even with a timed out controller. A charge counting device (E-cell) in the controller acts as a capacitive element under timed out (i.e., discharged) conditions, and stores enough energy for a short period of camera operation. Also, tests showed that the controller would time out if the system were switched on with or without the connector mated to the camera. An additional time out (discharge) mode was identified as a possibility when using a VOM for circuit/battery condition checks.

No single cause of early shutdown was identified. No report was made by the flight crew or destow personnel of connector demate.

Conclusions: The camera operate time correlates with a timed out controller. The controller may have been timed out by either circuit testing of controller/battery condition during flight system integration on system "turn on" with electrical connector demated before or after installation, or electrical connector demate during launch phase. No other possible causes for a timed out controller conditions were identified.

Corrective action: The controller electronics have been removed and operation shall be via crew-actuated hard-wired toggle switch for "on" and "off". The electrical connector interface to the camera will be taped securely to preclude a possible disconnection for any reason.

APPROVED


R. Machell

5/18/82
Date

Effective on subsequent missions: Requires crew participation for camera turn off.

Personnel assigned:  J. Bohannon/ED5; R. J. Ward/WA3

Resolution: CLOSED 5/17/82

Statement of problem: Cracked Rotor on Right Hand Outboard Brake

Discussion: One of the orbiter's 16 brake rotors had a 3/8-inch crack at one of the drive notches on its periphery. The small crack was on the outermost rotor of the right hand outboard (RHOB) brake. The RHIB and LHIB brakes also had galling and bending on some rotor drive lugs. The outboard rotors on all brakes had scraped their inside diameters on the torque tube. One stator on the LHIB brake was scraped by the wheel.

Postflight evaluation at the vendor has determined that the damage was probably caused by excessive axle, wheel and brake deflections resulting from high loads during landing and roll out.

Most of the damaged rotors can be reused. Some may need minor refurbishment or the direction of rotation can be reversed by exchanging sides. This reversal would apply the loads to the unused rotor drive lug surfaces.

The STS-3 brake damage is not a safety of flight issue but the galling and scraping would substantially reduce brake life increasing refurbishment and turn around costs.

Conclusions: High loads during landing and roll out probably resulted in excessive axle, wheel and brake deflections causing the damage to the brake rotors. This is not a crew safety problem but it is a brake life, cost and turn around issue.

Corrective action: Four new brake assemblies were installed for STS-4 and full design capability exists. Moderate braking not to exceed 8 to 10 fps² is planned for STS-4. Proposed redesign include adding C-5A type caps to the rotor drive lugs, stiffening the axle, and adding a foot support between the axle and torque tube.

APPROVED

A. Cohen
A. Cohen

5/24/82
Date

RCS

Effective on subsequent missions: Similar brake damage may occur on STS-4. All brake rotors will be inspected for evidence of galling, bending or scraping after STS-4.

Charles C. Campbell 5-21-82 *Robert J. Ward* 5/22/82

Personnel assigned: C. Campbell/EW3; R. J. Ward/WA3

Resolution: CLOSED for STS-4 5/19/82

Statement of problem: Water in Airlock Drag in Duct

Discussion: The STS-3 crew observed water collecting/around the outside of the airlock flex duct where the flex section enters the fairing on the airlock. Later in the mission the crew disconnected the duct and found water collecting inside the duct.

Post flight inspection of the cabin heat exchanger face showed chipped hydrophilic coating. The coating also failed the H₂O droplet test. However neither of these irregularities would cause the inflight problem based on vendor certification testing. The air flow measurement with each separator running indicated 13-14 lb/hr discharge air flow as compared to the specified value of 37 to 39 lb/hr minimum. With the separator disconnected from the heat exchanger slurper section the separator air flow increased to 68 lb/hr. Fiber optics inspection of the slurper interior and separator inlet check valves indicated no abnormalities.

The slurper was subjected to reverse GN₂ pressurization of > 4 psig. A small amount of dust was observed to be blown free from the heat exchanger outlet face. Retest with the fan separator attached to the heat exchanger slurper resulted in discharge air flow of 51.3 to 51.8 lb/hr.

Conclusions: The problem was caused by clogging of the slurper portion of the heat exchanger assembly. Should the clogging recur on STS-4, the air-cooled electronics would not be affected by the water in the cooling ducts. Contingency water removal equipment is onboard.

Corrective action: The slurper obstruction has been removed with the GN₂ purge. Analysis of the material blown from the slurper is in progress at the vendor. Cleanup and maintenance procedures for the slurper/heat exchanger are being developed.

APPROVED

A. Cohen
A. Cohen

5/20/82
Date

REB

Effective on subsequent missions: Periodic cleaning of the hydrophilic coating and heat exchanger slurper will be required during turnaround.

R. N. Prince
Personnel assigned: R. N. Prince/EC3

Resolution: CLOSED for STS-4 5/25/82

Statement of problem: Broken tabs on vertical tail graphite seals.

Discussion: A 1/4-inch cube of graphite was found in the Orbiter main engine 1 expansion chamber and was identified as a part of the vertical tail graphite seal.

This thermal conical seal consists of graphite rubbing seal blocks which press against the conical panels on the vertical tail. These seal blocks are contained within Inconel channels and are spring loaded against the Inconel panels. The seal blocks overlap and have tabs which interlock to provide redundancy in their support. Ten (10) tabs were broken of a total of approximately 150 seal blocks. Five were broken on each of the upper and lower panels, (6 on the left-hand side and 4 on the right-hand side).

The vertical stabilizer test article (VSA-16) was inspected and no broken tabs were found. It has been subjected to the 100-mission vibro/acoustic tests. Tab breakage could have occurred during installation or during the flight.

The broken tabs do not affect the retention of the seal blocks within the Inconel channels. A missing tab will result in a minor flow path not affecting the primary sealing surfaces. Any resulting increased heating would be localized and analysis shows it to be well below the thermal limits of the seal materials.

Conclusions: The cause of the broken tabs is unknown; however, the seal is fully effective with broken tabs.

Corrective action: None.

APPROVED

A. Cohen
A. Cohen

6/7/82
Date

Effective on subsequent missions: None.

Personnel assigned: *M. Piercey* 4/3/82
(N. Piercey/EWS; C. J. Walsh/WC3)

Resolution: CLOSED 6/4/82

Statement of problem: Urine leak between the urine cup and filter.

Discussion: The STS-3 crew reported that urine would build-up outside the urinal at the cup and filter interface after the spare filter was installed. A dimensional check of the flight urine cup and filter determined that the gasket on the spare filter was too thin. The spare gasket had been shaved during a fit check to allow mating into the locked position because the material was not soft enough to compensate for the dimensional tolerances of the cup and filter. Fit checks postflight found some flight units could not be locked into place because the gaskets were too thick and stiff.

Conclusions: The urinal filter gasket was too thin and stiff to compensate for the mating dimensional tolerances and ensure a seal between the urine cup and the filter.

Corrective action: A new softer and thicker material shall be used in the urinal filter gaskets to seal the interface between all urinal cups and filters, but still allow the units to be easily locked together. The urine quick disconnect adapter gasket will be changed to the same material. The new gaskets have been installed for STS-4.

APPROVED *Aaron Cohen* 6/7/82
REP A. Cohen Date
6-4-82

Effective on subsequent missions: None

Henry A. Rotter *Robert J. Ward* 6/2/82
 Personnel assigned: H. A. Rotter/EC3; R. J. Ward/WA3

Resolution: CLOSED 6/1/82