Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-01	MPS
BSTR-01	**GMT:**		**SPR** KB2654, 55RF05	**UA** UA-2-A0022	**Manager:**
			IPR	**PR**	
					Engineer:

Title: LH2 4-in ET/Orbiter Disconnect (PD3):a) Slow Closure Following Pad Abortb) Did Not Close at MECO (ORB)

Summary: DISCUSSION: a) The LH2 4-inch External Tank (ET)/Orbiter disconnect valve (PD3) was slow in closing following the launch pad abort on March 22, 1993. The abort was initiated about 3 seconds prior to lift-off when the Space Shuttle main engine (SSME) 3 oxidizer preburner purge pressure exceeded its upper limit due to a leaking check valve. The PD3 open switch indicated off in 0.538 second and its close switch should have indicated on approximately 1.1 seconds after the close command was issued. Lack of disconnect closure was confirmed by engine inlet and SSME pump discharge pressures as compared to tank ullage pressure. The Operational Maintenance Requirements and Specifications Document (OMRSD) file IX close-time requirement for PD3 is no more than 2.8 seconds. The PD3 closed switch indicated on after 642 seconds, coincident with the opening of the topping/replenish valve (PV13). Flow forces associated with the opening of the topping/replenish valve apparently assisted in closing PD3.

The corrective actions taken prior to the STS-55 mission included reviewing the history of the Orbiter and ET disconnects and developing a fault tree to guide in the determination of a most probable cause. Testing at Rockwell/Downey and inspections/testing of the vehicle hardware were employed to address the failure mechanisms developed from the fault tree. The Orbiter disconnect (S/N CRP 1003) was built in 1986 and was intended to be first used on OV-105. Instead, the disconnect was installed on OV-102 prior to STS-35 as a result of the LH2 leak problems that this vehicle experienced prior to that flight. There were no significant problems experienced during acceptance testing and there were no flight anomalies prior to the STS-55 launch attempt. A review of the ET disconnect build/test paper revealed no anomalies. The fault tree logic established the following failure mechanisms: GN2 intrusion into the umbilical plate gap forming N2 ice on the valve mechanism and resulting in binding; hydraulic fluid on the valve mechanism (left-over hydraulic fluid from the hose rupture experienced earlier in the flow) that freezes under cryogenic conditions and binds the valve; contamination in the flowstream that holds the flapper off the flapper seat; binding of the actuator due to moisture instrusion that freezes under cryogenic conditions; and a damaged valve mechanism that results in excessive drive train friction at cryogenic conditions. Actions conducted to exonerate each of these failure mechanisms included testing and/or inspections. These actions included valve cycles, leak checks, and borescope inspections of the hardware on the vehicle,

as well as tests at LN2 temperatures (-300 degree F) in the laboratory at Rockwell/Downey. These actions were unsuccessful in identifying a cause of the failure. The failure was accepted as an unexplained anomaly with the following hardware changes being made to eliminate several concerns and expedite the failure analysis effort. The PD3 actuator and the LV51 (PD3 closing solenoid) vent check valve were removed and replaced. Removal and replacement of the actuator eliminated concerns for actuator-related problems and allowed an inspection of the PD3 gear mechanism and the Orbiter-half negator spring. Removal and replacement of the LV51 vent check valve minimized the concern for cryopumping into the actuator. With these hardware replacements and the successful completion of the required checkouts, the decision was made to fly. It should be noted that cryogenic testing of the removed actuator was completed prior to flight and no anomalies were noted. b) PD3 did not close when commanded following main engine cutoff (MECO). The valve close command was issued 0.040 second after MECO and the open switch indicated off 0.607 second after the close command. The closed indication should have been picked-up within 2.8 seconds of the close command (OMRSD file IX requirement) but it did not. The command to initiate ET/Orbiter umbilical separation was issued 10.360 seconds after the close command and the closed indication was picked-up 0.459 second later (10.819 seconds after the close command). These data indicate that Orbiter-half flapper closed immediately following umbilical retraction. It is not believed that the failure to pick-up the closed indication prior to ET umbilical retraction was due to an intermittent failure of the closed position indicator. The position indicator functioned properly during the STS- 55 launch abort as confirmed by pressure data, and it functioned properly during ambient checkout prior to launch and during postflight troubleshooting. When the vehicle was returned to KSC, troubleshooting was performed. During troubleshooting, an audible leak was detected at the LV50 outlet B-nut (IFA STS-55-V-12). This leak is not believed to be related to the PD3 anomaly. The PD3 actuator was cycled and it operated smoothly even with the leaking B- nut. Wiring, tubing, and foam insulation were visually inspected and no anomalies were noted. There was no hydraulic fluid found on the Orbiter side of PD3. A visual inspection of the valve/actuator mechanism was performed and no anomalies were noted. Vent port leak checks were performed on LV50 and LV51 and the leak rates were within specification. Although troubleshooting on the vehicle failed to determine the cause of the anomaly, the decision was made to replace PD3 since thermally induced binding of the disconnect was a potential cause. LV50, LV51 and the PD3 actuator were also removed and sent with PD3 to Rockwell/Downey for further failure analysis. Testing of the disconnect at LH2 temperatures (the solenoid valves were also chilled) also failed to reproduce the anomaly. From the time of the pad-abort anomaly, it was realized that binding of the ET- half valve mechanism could be the cause of the failure. Binding of the ET mechanism would keep the Orbiter-half mechanism from closing until it was decoupled from the ET-half by the retraction of the umbilical. A review of film from the umbilical-well camera indicated that the ET-half of the valve was closed at ET separation. However, the retraction of the umbilical could have possibly relieved the binding in the ET-half valve mechanism and allowed its negator spring to close the flapper. Therefore, binding of ET-half valve mechanism is still considered to be the most likely cause of the failure. CONCLUSION: No anomalies have been noted with the Orbiter-half of the PD3 hardware or its actuator and controlling solenoid valves. The most likely cause of the anomaly is binding of the ET-half valve mechanism. CORRECTIVE_ACTION: The LH2 4-inch ET/Orbiter disconnect valve (PD3), the valve actuator, and the open and close solenoid valves (LV50 and LV51, respectively) were removed and replaced. They were shipped to Rockwell/Downey for failure analysis, the results of which will be tracked under CAR KB2654. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-02	undefined
None	**GMT:**		**SPR** 55RF01	**UA**	**Manager:**

			IPR	**PR** GNC-2-14-0094	x38278
					Engineer:

Title: IMU 2 Platform Fail Bite Indication (ORB)

Summary: DISCUSSION: During the prelaunch mode change of inertial measurement unit (IMU) 2 from standby to operate at 114:04:32 G.m.t., dc/dc 1 platform, input/output (I/O), and transmission word 1 fail built-in test equipment (BITE) indications were registered. The IMU did reach the operate mode successfully. Several subsequent cycles between standby and operate did not recreate the transient BITE indications. The launch attempt was scrubbed and this IMU, a KT-70 model serial number (s/n) 0017, was removed and sent to the JSC Inertial Systems Laboratory (ISL) for testing. A HAINS model IMU (s/n 205) was installed for launch.

The findings, upon reexamination of the prelaunch data, indicate that the dc/dc 1 card was at fault. The likely scenario is that the voltage was varying above or below tolerance. Of the three voltages supplied by the dc/dc 1 (+5, -15, +15), only the +5 could be ruled out as being at fault since it supplies power to all cards outside the platform; therefore, a failure of this voltage would have resulted in many more BITE indications being registered. A total of 37 upmode sequences were conducted at the ISL on the IMU after removal. In only one of these cases was a discrepancy observed. The -15 V power supply went low by approximately one volt for a period of 8 ms after the IMU was moded to operate. This voltage signature would have caused similar annunciations as the original failure, had it sustained long enough for fault detection software to detect the problem. Due to the signature of the failure, it was concluded that the problem is within the -15 V circuit on the dc/dc 1 card. The card was removed at JSC, and a visual inspection was performed. The card was then sent to the vendor for analysis. At the time of this writing, no cause has been identified. This failure is not believed to be generic in nature because of the failure history of the card. Troubleshooting will continue under CAR 55RF01. CONCLUSION: The problem lies within the dc/dc 1 card. The specific failure within that card is not currently understood. CORRECTIVE_ACTION: The card was removed from the IMU, and is currently undergoing analysis at the vendor. The results of the failure analysis will be documented under CAR 55RF01. EFFECTS_ON_SUBSEQUENT_MISSIONS: None expected.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-03	EPD&C - Hardwar
GNC-01	**GMT:**		**SPR** 55RF06	**UA**	**Manager:**
			IPR 58V-0007	**PR**	x38393
					Engineer:

Title: Forward Motor Control Assembly 2 Operational Status 2 Did Not Invert (ORB)

Summary: DISCUSSION: At approximately 116:16:48:32 G.m.t., the -Y and -Z star-tracker doors were successfully opened. After the -Y and -Z star-tracker doors were opened, the FMCA 2 Operational Status 2 measurement failed to return to the nominal value "1" state. This measurement reflects the status of vent door left/right (L/R) 1-

2 motor 2, L/R air data probe motor 2, star-tracker -Y system 2 motor-close B, and -Z star-tracker system 1 motor-open B relays. The failure remained throughout the mission.

Two hybrid relays are in series for the -Z system-1 motor. Series redundancy enhances system reliability against inadvertent application of power to the motor. For the -Z star-tracker system 1 motor control, the "open B" hybrid relay is K16. The star-tracker doors were fully opened, which allowed normal operations with no mission impact. Postflight troubleshooting at Dryden Flight Research Facility repeated the anomaly. Further troubleshooting at KSC isolated the anomaly to FMCA 2. The FMCA was sent to NASA Shuttle Logistics Depot (NSLD) for teardown, testing, and evaluation. The failure was isolated to the "D" contacts of hybrid relay K16. The "D" contacts serve as instrumentation reflecting the FMCA operational status measurement. The "A", "B", and "C" K16 contacts, which are used to transfer the 3 phase ac power to the motor, remained operational, indicating that the most probable cause of the failure was contact contamination. CONCLUSION: Troubleshooting at NSLD isolated the problem to a faulty contact on the K16 relay. CORRECTIVE_ACTION: The FMCA has been removed and replaced. The failed hybrid relay will be replaced, an acceptance test will be performed on the FMCA, and the FMCA will be returned to KSC stores as the only available spare. The hybrid relay will be sent to Rockwell-Downey for failure analysis and will be tracked by the listed CAR. EFFECTS_ON_SUBSEQUENT_MISSIONS: None. Should this failure recur, a loss of redundancy on the systems that the FMCA 2 supports would occur.

Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-04	Water and Waste
ECCOM-01, EECOM-02	**GMT:**		**SPR** 55RF02	**UA**	**Manager:**
			IPR	**PR** ECL-2-14-0083	
					Engineer:

Title: Waste Water Tank GN2 Leak (ORB)

Summary: DISCUSSION: At approximately 117:16:38 G.m.t., the system 1 nitrogen flowrate was noted to increase abruptly from 0.20 lbm/hr to 0.75 lbm/hr. At the same time the water tank nitrogen pressurization system 1 regulator outlet pressure dropped, inidicating a leak in the supply/waste water pressurization system. The second nitrogen system was activated in parallel with system 1, and it too reached 0.75 lbm/hr, indicating that both water pressurization regulators were flowing their maximum flowrate, and could not keep up with the leak. Leak isolation procedures indicated that the leak was either in the common manifold that pressurized tanks B, C, D, and the waste tank, or in one of the tanks. The crew then performed a visual inspection of the water tanks, and found that the waste tank had several buldges in its side including one portion that was punctured, causing the nitrogen leak. No liquid had leaked. The mission continued with the crew using the contingency waste container as a substitute line to the waste tank. In a later in-flight maintenance procedure, the nitrogen line to the waste tank was capped and the supply water tanks were repressurized.

After the flight, the waste tank was removed and opened. A flat spot/dent in the pressure wall of the tank had caused one side of the tank bellows to jam. The bellows pivoted about the jam and deformed to accomodate increasing amounts of waste water. Two bellows guide feet were forced into the thin (0.04 in.) aluminum wall of the tank, one of them causing the puncture and leak. During the STS-52 (OV-102 flight 13) mission the waste line liquid pressure experienced a drop during waste dumps below 5 percent quantity, indicating some bellows stiction at low quantities. Data review at that time indicated that similar line pressure drops were seen as early as STS-40 (OV-102 flight 11) for dumps below 5 percent. At the time, this phenomenon was attributed to different bellows spring rates and/or urine salt buildup on the bellows, and was not believed to be an issue for tank quantities below 10 percent. During the STS-55 processing flow, a higher-than-normal gas pressure had to be used to dump the tank down to its preflight level. During the first day of the STS- 55 flight, the waste liquid line pressure rose rapidly to 23 psig (normally 15.5 to 19 psig) while water was flowing into the tank prior to the puncture, indicating bellows stiction. CONCLUSION: The waste tank nitrogen leak was caused by a dent on the tank wall, that could have incurred prior to STS-40, eventually jammed the bellows, and forced a corner of one of the bellows guide feet through the tank wall. CORRECTIVE_ACTION: The OV-102 waste tank has been removed and will be replaced with another unit. Detailed failure analysis will be reported under the listed CAR. Supply, waste, and water spray boiler tank data for the other vehicles and for the other OV-102 tanks have been scrutinized for similar signatures that were observed with the OV-102 waste tank before the leak. This scrutinization will contnue for future flights, allowing for early detection of bellows problems. An RCN establishing trend requirements monitoring will be processed. A hysteresis map of each bellows spring force will be performed on all tanks during Orbiter maintenance down periods. This will provide early detection for any changes in the force required to move the bellows. A new piece of ground support equipment is being designed to protect the waste tank when the LiOH box is removed during ground processing. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-05	FC/PRSD
EGIL-01	**GMT:**		**SPR** 55RF03	**UA**	**Manager:**
			IPR 58V-0002	**PR**	
					Engineer:

Title: Fuel Cell 2 O2 Flow Sensor Failed Off-Scale Low (ORB)

Summary: DISCUSSION: At approximately 117:06:51 G.m.t., the fuel cell 2 O2 flowmeter sensor (P/N 810942-01, S/N V102) indicated off-scale low. This did not affect the continuation of the mission since the measurement of cyrogenic consumption and the detection of gross leakage are performed using other parameters.

After the mission, testing at KSC confirmed the problem to be within the flowmeter. An insulating pad had been added to this flowmeter to correct a known generic problem of a power transistor shorting-to-ground, which had previously caused similar symptoms. CONCLUSION: The fuel cell 2 O2 flowmwter failed off-scale low. The cause of the failure is currently unknown, but will be reported under the listed CAR when the flowmeter is available for failure analysis. CORRECTIVE_ACTION: Fly as-is until fuel cell 2 must be removed for other reasons. Replacement of the flowmeter requires removal of the fuel cell. Since this flowmeter is criticality 3, and its function can be performed using other available measurements, flying with this failed flowmeter will not impact mission success or crew safety. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	MET:	Problem	FIAR	IFA STS-55-V-06	OMS/RCS
PROP-01	GMT:		SPR 55RF04	UA	Manager:
			IPR	PR LP05-04-0071	
					Engineer:

Title: RCS L4D Primary Heater Fail-On (ORB)

Summary: DISCUSSION: The aft reaction control system (RCS) primary thruster L4D (s/n 628) heater failed-on at 117:18:38 G.m.t. (1:03:48 mission elapsed time [MET]). The thruster heater had exhibited nine nominal heater cycles prior to failure. Throughout the remainder of the mission, the heater was controlled manually via a cockpit on all aft RCS thrusters on manifold 4 (L4D, L4L, L4U, R4D, R4R and R4U).

The heaters were managed to maintain the injector-tube temperatures on thruster L4D below 160 degrees F and the injector tube temperature of all the affected thrusters above 60 degrees F. The 160 degrees F temperature sensed at the injector tubes corresponds to a 150 degrees F temperature at the valve seat, which is the upper operational limit of the seat. The upper non- operational temperature limit of the seat is 175 degrees F, which corresponds to 185 degrees F sensed at the injector tubes. The 60 degrees F temperature sensed at the injector tubes protects the valve seat lower limit of 50 degrees F. This failed-on heater was the third experienced with OV-105-build thrusters. The other two were heaters on thrusters F4R on STS-49 (s/n 613) and L1U on STS- 50 (s/n 625). The preliminary input from the failure analysis indicated that the failure mode of these two heaters was either a partial short (resistive breakdown) or a short of the capacitor installed in the C3 position in the heater controller circuit. Further investigation revealed that during the build-up of the controllers, the capacitor normally installed in the C2 position was installed in the C3 position on both failed controllers. The C2 capacitor is rated at 6 V and the C3 capacitor is rated at 75 V. With the C2 capacitor installed in the C3 position, it sees full bus voltage (27 V nominal) when the cockpit heater switch is in the "on" position. The failure mode is suspected to be the degredation of the capacitor's dielectric barrier due to excessive current leakage at the higher-than-rated voltage. The time to failure is expected to vary from lot-to-lot (perhaps capacitor-to- capacitor) due to manufacturing variabilities. The F4R (s/n 613) and L1U (s/n 625) thruster heaters failed-on after approximately 20 and 24 hours, respectively. STS-55 was the third flight of thruster s/n 628 (L4D). It is unclear as to how the misinstallation of the capacitors occurred. It is suspected that some C2 capacitors got placed with a batch of C3 capacitors. They would be difficult to distinguish visually. Data recorded during the build process are inadequate to determine if the installed capacitors are of the correct configuration. It should be noted that the correct capacitor was installed in the C2 position of the thruster F4R and L1U heater controllers. The acceptance test procedure (ATP) includes an operational test that supplies 27 V to the controller circuit for 10 to 15 minutes, but this is obviously not long enough to screen for misinstalled capacitors. A field test of flight thrusters is being considered to verify proper heater controller configuration, although this would result in considerable effort. At a minimum, the test could be used to verify the configuration of the spare thrusters. Until then, the OV-105-build thrusters will be operating with some risk of the heater failed-on failure. Note that the OV-105-build forward RCS module and orbital maneuvering system (OMS) pods have each flown three times, and therefore, the majority (perhaps all) of the out-of-configuration heater controllers may have been identified by failures. There is, however, uncertainty in this conclusion due to variability in time-to-failure with this failure mode. Also, OV-105 (next flight is STS-57) has one OV-105-build thruster (R1R) which has never flown a mission. There is no safety-of-flight issue associated with this failure mode. A fail- on thruster heater, while undesireable, is manageable operationally (as was done on STS-49, STS-50, and STS-55). A failed-off thruster heater is more difficult to manage and has the associated risk of thruster valve leakage. However, the vendor had indicated that failed-off is not a concern for the failure mode associated with the misinstalled capacitor. Any failure of the capacitor in the C3 position (open or short) would at worst result in the heater failing on. CONCLUSION: The most probable cause of the heater failure is a short of the capacitor in the C3 position in the heater controller circuit. This is probably due to the incorrect capacitor being installed in the C3 position. CORRECTIVE_ACTION: The thruster was removed and replaced and transferred to the White Sands Test Facility (WSTF). This opportunity will be used to flush the thruster prior to the heater controller being removed and replaced at the WSTF by the vendor. The thruster will then be returned to spares at KSC. The failure analysis of the heater controller and the resulting corrective action are being tracked on CAR 55RF04. EFFECTS_ON_SUBSEQUENT_MISSIONS: None. Should the failure recur, the thruster heater can be manually cycled and the thruster can be used.

Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-07	Hydraulics
MMACS-02	**GMT:**		**SPR** 55RF10	**UA**	**Manager:**
			IPR 58V-0008	**PR**	x39033
					Engineer:

Title: Anomalous hydraulic system 3 main pump case drain temperature. (ORB)

Summary: DISCUSSION: During a review of thermal data, it was noticed that the hydraulic system 3 main pump case drain temperature (V58T0385) had an unusual signature starting at 188:05:12 G.m.t., and the temperature appeared to be responding to the system 3 hydraulic circulation pump operation. Data were reviewed for previous missions of this vehicle and it was verified that the temperature sensor was operating similarly.

This sensor is used primarily for supporting hydraulic water spray boiler and heat exchanger operational periods on ascent and entry. However, the thermal community uses the measurement as an indication of the temperature at the bulkhead during on-orbit operations. During postflight vehicle processing at the Kennedy Space Center, the temperature sensor was inspected and found to be bonded to the incorrect hydraulic line. The sensor was bonded to the system 3 high pressure outlet line of the filter module adjacent to the case drain. The temperature sensor prior to STS-50 was a Developent Flight Instrumentation measurement (DFI). During the Orbiter Maintenance Down Period (OMDP) the measurement was reconfigured to Operational Instrumentation (OI). Current data indicated that the measurement was initially installed on the wrong line as a DFI measurement, and not corrected when it was converted to OI. CONCLUSION: The temperature sensor was bonded to the incorrect hydraulic line. CORRECTIVE_ACTION: The temperature sensor for the hydraulic system 3 main pump case drain will be reconfigured prior to STS-58. EFFECTS_ON_SUBSEQUENT_MISSIONS: none

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Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-08	Active Thermal

EECOM-08	**GMT:**	**SPR** None	**UA**	**Manager:**
		IPR None	**PR**	
				Engineer:

Title: FES Shutdowns (ORB)

Summary: DISCUSSION: The flash evaporator system (FES) performed one 11-hour water dump at approximately 118:04:51 G.m.t. with the supply water tanks depressurized. A second dump with the tanks depressurized was initiated at approximately 118:13:09 G.m.t. Approximately one hour and twenty minutes into the dump, the FES experienced an over-temperature shutdown. The FES was deactivated until the supply water tanks were repressurized. At approximately 118:23:47 G.m.t., a FES water dump was initiated with the supply water tank pressurization restored. After five minutes of stable operation, the FES again experienced an over-temperature shutdown. The crew cycled the FES again controller, and the FES immediately shut down without reaching its control band. All shutdowns occurred on the Primary A controller.

The shutdowns were attributed to ice formation in the topping core while operating at low supply water pressure. Ground tests performed early in the Program had indicated that icing was possible at the low supply-water pressures. The ice was successfully removed from the core using a flush procedure at approximately 120:05:12 G.m.t. The flush procedure involved bypassing the radiators to increase the Freon inlet temperature to the FES and thereby loosen the ice, after which the secondary controller was cycled several times to flush the ice out of the core. A momentary drop in the right topping duct temperature was noted during the procedure, indicating the passage of ice. The FES was then operated for about one hour on the Primary B controller before it was deactivated to conserve power. Two successful FES water dumps were performed later in the flight using the Primary A controller, exonerating the controller as a cause for the shutdowns. After the mission, the FES core was borescoped as a part of normal turnaround processing. No damage or anomalous conditions were visible. CONCLUSION: The FES shutdowns were caused by ice build-up in the topping core as a result of FES operation at reduced supply water pressure. CORRECTIVE_ACTION: The flight rules are being changed so that the FES will not be used with depressurized supply water tanks unless absolutely necessary. If FES operations with reduced supply water pressure must be performed and icing occurs, the existing flush procedure can restore use of the FES. EFFECTS_ON_SUBSEQUENT_MISSIONS: FES operations with the supply water tanks depressurized will not be attempted unless absolutely necessary.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-09	TCS
MMACS-01	**GMT:**		**SPR** 55RF11	**UA**	**Manager:**
			IPR	**PR** TCS-2-15-2052	
					Engineer:

Title: Loose Thermal Cover on Tunnel Adapter Hatch (ORB)

Summary: DISCUSSION: While viewing payload bay camera video downlinked early in the mission, it was noticed that the thermal cover on the tunnel adapter extravehicular activity (EVA) hatch was open approximately 80 degrees. The hatch cover apparently came open during ascent. A thermal analysis and evaluation of possible interference with nearby hardware determined that the open hatch cover would cause no impact to the mission.

The hatch cover is held in place by five Velcro straps. A review of the installation paperwork showed that the Velcro straps had been verified to be properly attached after the cover/ring assembly had been installed on the airlock tunnel. Troubleshooting indicated that the bonding strength of the Velcro straps was inadequate to support even gentle shaking and/or pulling motions. Although not quantified, the binding force of the Velcro straps was somewhet degraded from when they were new. Also, it should be noted that even when new, the Velcro used on these straps has a relatively weak bonding strength. This was understood with the initial selection of this type of Velcro; however, a primary criteria for its selection was that it met the contamination requirements for use in the payload bay. For the near term, two changes will be made. The current Velcro hook and pile will be replaced with a type that is twice as strong, more durable, and certified for use in the payload bay. Also, two additional Velcro straps are being installed on the cover/ring assembly. The changes have been approved, and an engineering order will be released to incorporate the change. For future missions, implementation of a positve latch design is being considered. CONCLUSION: The cause of the loose thermal cover was degradation of the Velcro straps that secure the cover. The loose cover had no impact on the mission. CORRECTIVE_ACTION: The current Velcro hook and pile is being replaced with a stronger and more durable Velcro that is certifed for use in the payload bay. Also, two additional Velcro straps are being installed on the cover/ring assembly. The changes have been approved, and an engineering order will be released to incorporate the change. EFFECTS_ON_SUBSEQUENT_MISSIONS: Should the anomaly recur and there is a thermal impact, the on-orbit attitudes can be managed to maintain proper temperatures. However, this attitude management could have an affect on mission objectives. The STS-57 planned attitude timeline was evaluated and there will be no thermal impact should the anomaly recur.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-10	DPS - MMU
DPS-01	**GMT:**		**SPR** 55FR08	**UA**	**Manager:**
			IPR	**PR** DIG2-15-0172	x38376
					Engineer:

Title: MMU 1 SM Checkpoint Fail (ORB)

Summary: DISCUSSION: At 123:23:35 G.m.t., the crew initiated a systems management (SM) checkpoint to mass memory unit (MMU) 1, serial number (s/n) PD02, with an item 18 execution on SM specialist function page (SPEC) 60. Approximately 13 seconds later, "OFF/BUSY MMU1" and "S60 CHECKPT FAIL" fault messages were annunciated by general purpose computer (GPC) 4. GPC 4 logged a single input/output (I/O) error and an op code indicating an MMU OFF/FAIL during an MMU utility write. The crew power-cycled the MMU per the malfunction procedure and successfully retried the SM checkpoint. The MMU performed well for the remainder of the mission.

Upon further review of data taken during performance of the malfunction procedure from the power cycle done by the crew, it was discovered that the MMU had a second indication of a problem. The ready discrete indicated busy when no MMU transaction was being performed. The MMU was removed and replaced. This MMU experienced a similar failure in October of 1990 while installed on OV-104, and it may be related to this failure (reference CAR KB1549). The failure in 1990 was isolated to a noise problem causing the ready discrete to indicate busy for a longer period of time than it should. A design change was implemented to correct this failure. The power supplies in the MMUs are inherently noisy. Large amounts of noise tend to be generated by the Servo-preregulator power supply when the motor stops. All MMUs see similar noise, but it varies from unit to unit. Although numerous design modifications have been made to filter and reduce the noise within the MMUs, it is still present. The MMU was removed from the vehicle and sent to JSC Avionics Engineering Laboraory (JAEL) for testing. The problem has not yet recurred. IPLs have been conducted daily, and more than ten SM checkpoints have been completed. Also, a complete load and verify was performed. Should the problem recur, the unit will be sent to the vendor for further failure analysis and repair. If the problem does not recur, the unit will be returned to the field as a flightworthy unit [pending agreement with the Problem Review Team (PRT)], with all test results being documented on CAR 55RF08. CONCLUSION: The most likely cause of this failure was an intermittent noise problem with the ready discrete. CORRECTIVE_ACTION: The MMU was removed and replaced. The unit is currently being observed, and data will be collected and analyzed in the event of a recurrence of the failure. EFFECTS_ON_SUBSEQUENT_MISSIONS: Should a failure of this nature occur on a subsequent mission, the MMU would most likely be usable following a power cycle. If it is not, another MMU is available on board for use.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-11	DPS
DPS-02	**GMT:**		**SPR** 55RF07, AD9687	**UA**	**Manager:**
			IPR	**PR** DIG-2-15-0171	x38367
					Engineer:

Title: CRT 4 Failure (ORB)

Summary: DISCUSSION: During on-orbit operations on STS-55, general purpose computer (GPC) 4 annunciated an "I/O ERROR CRT 4" fault message at 124:12:59 G.m.t., when display electronics unit (DEU)/display unit (DU) pair 4 was powered-up for use. The crew reported that the DU was blank and the DEU built-in test equipment (BITE) flag was tripped. The crew performed a power cycle of the DEU/DU, but the cathode ray tube (CRT) was not recovered. The problem appeared to be hardware-related because the power cycle did not recover the CRT or the tripped DEU BITE flag. Due to the belief that the problem was hardware-related, and in the interest of preserving the DEU memory for further analysis, a software initial program load (IPL) of the DEU was not performed. The CRT was left unpowered for the remainder of the mission.

During the postflight debriefing, the crew reported that the "SM ANTENNA" display [operational sequence (OPS) 201 display] looked abnormal at one point when displayed on CRT 4. A photograph of the display, taken at approximately 122:02:12 G.m.t., shows lines on the outline of the projected view of the KU- band antenna extending beyond where the lines should be present. Postlanding, the DEU (serial number 33) was replaced and sent to the vendor for testing. The DEU again failed to produce a display on a CRT. A broken solder joint was discovered between an integrated circuit (IC) and the 6239205- 4 central processing unit (CPU) page within the DEU. The broken solder joint caused errors in the processor memory output data bus, and this resulted in memory alterations. These memory alterations to the DEU rendered it incapable of driving a display on the CRT. Inspection of the other joints between this IC and the CPU page revealed that more of the pins had stress cracks. There are four of these ICs on this page and two of the remaining three also had cracked solder joints. All of these joints have been reworked. Analysis of the failure indicates that the solder joints cracked due to Uralane beneath the IC expanding and contracting from temperature changes, resulting in stress to the solder joints. Three other CPU page assemblies available at the vendor were inspected, and the ICs on these pages showed no indication of joint fractures. However, the ICs on these pages, while identical in function, are different in form to the ICs used on the page that failed. The ICs without fractures have thinner and longer leads, allowing more stress relief. Additionally, the three pages without fractures have slightly more solder than the one with the fractures. There are eight other CPU pages in the field which have the same type ICs used on the page which failed in-flight. Four of these are in spare DEUs at KSC, and four of these are in DEUs currently installed in vehicles - one on OV-103, two on OV-102, and one on OV-105. None of these eight pages have been inspected for indications of stress. A separate broken solder joint was found to be the cause of the "SM ANTENNA" display problem. This broken solder joint on the Circles and Vectors page (part number 6239213-22) caused the vector-length counter to incorrectly calculate display-line lengths, and this condition explains the reported display anomaly. This failure is also due to the expansion and contraction of Uralane that caused stress to the solder joint. Analysis of this failure is ongoing, and will be documented on CAR AD9687. Upon successful completion of acceptance testing, the DEU will be returned to the field as a flightworthy unit. A screen has been put in place to inspect solder joints which are subject to this stress for obvious signs of damage. The manner in which this screen will be imposed is not yet determined, but it will be documented on the CAR. Should a failure of this type recur during flight, the symptoms will most likely be a benign type of DEU failure, similar to that seen on STS-55. If the failure is such that the DEU is unrecoverable, then it would be powered- off for the remainder of the flight. Since the failure occurred in the DEU CPU, it is possible that a broken solder joint could result in erroneous output by the DEU on the display-keyboard (DK) data busses. The possibility of this condition resulting in the redundant set failing due to differential input/output (I/O) error processing cannot be ruled out, nor can the possibility of the DEU generating an erroneous keystroke sequence that the GPC interprets as valid and executes accordingly. A complete risk analysis has not been performed due to prohibitive cost, but given the analysis that has been done, expert opinion of the failure mode is that the likelihood of this problem causing a criticality 1/1 keystroke sequence failure is less than 1.62 X 10 -27, and the likelihood of the criticality 1/R2 PASS set split is less than 2 X 10 -7. An in-flight maintenance (IFM) procedure could be performed to replace a failed forward DEU with aft DEU 4. A minimum-duration flight (MDF) is declared for two DEU/DUs failed. A next primary landing site (PLS) landing is performed for two forward DEU/DUs failed. CONCLUSION: The two anomalous conditions were caused by two separate solder joints fracturing as a result of Uralane expanding and contracting, causing stress to the solder joints. Contributing factors of the blank DU solder joint failure include the amount of solder used, the shape of the leads, and the fact that the devices to which these ICs are attached draw large amounts of current, resulting in high temperatures. CORRECTIVE_ACTION: The DEU was replaced. The solder joints were repaired. For the remaining DEUs, a screen has been put in place to inspect solder joints which are subject to this stress. This manner in which this screen will be implemented is currently undecided, and will be documented on the CARs. EFFECTS_ON_SUBSEQUENT_MISSIONS: Should the failure recur during flight, the actual symptoms exhibited by the DEU could be any number of conditions, including loss of the primary avionics software system (PASS) redundant set or unwanted action generated by the DEU. However, the likelihood of this occurring is so low that it is an acceptable risk. The most likely scenario is the loss of the DEU function. Loss of the redundant set would require freeze-dry GPC activation or backup flight system (BFS) engage, depending on availability. There is no mission impact if a single DEU fails. If a forward DEU fails, an IFM may be performed to replace it with aft DEU 4.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	MET:	Problem	FIAR	IFA STS-55-V-12	MPS
None	GMT:		SPR None	UA	Manager:
			IPR 58V-0006	PR MPS-2-15-0952	
					Engineer:

Title: MPS Pneumatic He Pressure Decay During Ascent (ORB)

Summary: DISCUSSION: During ascent, the main propulsion system (MPS) pneumatic helium system supply pressure, as measured by V41P1600A, decayed 80 psi. The pressure was 4240 psia at liftoff and 4160 psia at MECO. This decay is 20 psi (1-data bit) more than the maximum allowable decay during ascent as specified in Opertional Maintenance Requirements and Specifications Document (OMRSD) File IX requirement DV41AZO.150. This requirement specifies a pneumatic helium system supply pressure change of no more than +20 or -60 psi during the time period from liftoff to main engine cutoff (MECO). There is not a flow demand on the pneumatic helium system during the period from liftoff to MECO. Therefore, the pressure change specification of no more than +20 or -60 psi is based on a single-bit toggle or allowable leakage and thermal effects, respectively. On-orbit, the leak was not evident, and therefore had no impact on the mission.

The intent of the File IX requirement is to verify that no leakage exists in the MPS pneumatic helium supply system through either the pneumatic helium fill check valve (CV4), the interconnect outlet check valves (CV28, 39, and 44), or the interconnect inlet valves (LV59, 61, and 63). However, a pressure drop could also result from leaks in the pneumatic helium system downstream of the 750-psia pneumatic system regulator. Potential leak sources downstream of the regulator include numerous solenoid valves that control pressure to actuators in the MPS propellant-management system. Since the leak was not evident post-MECO, it was speculated that the leak might be associated with the solenoids that were energized for launch and de-energized for MECO and ET separation. These include the solenoid valves that control pressure to the LO2 and LH2 17-inch disconnect valve latch actuators, the LO2 and LH2 prevalve actuators, and the LH2 4-inch disconnect valve (PD3) actuator. When the vehicle was returned to KSC, troubleshooting was performed to isolate the leak. The pneumatic helium system was activated and the solenoids suspected of leaking during ascent were energized. The leak was expected to be audible due to the leak size required to result in the pressure drop seen during ascent. The OMRSD allowable pressure decay of 60 psi corresponds to a leakage of approximately 1000 standard cubic inches per minute (SCIM). The calculated leakage for STS-55 was approximately 3000 SCIM. An audible leak was detected at the LV50 outlet B-nut. LV50 is the opening solenoid for PD3. Checkout of LV50 indicated that the torque on the B-nut was less than specified (40 in-lb versus the specification of 145 to 190 in-lb). The B-nut and the solenoid adapter and seal were removed and inspected and no anomalies were noted. A new adapter and seal were installed and the fitting was retorqued. A leak test was performed and the leak did not recur. It was determined that the improper torque on the B-nut probably occurred as a result of inspections made during clean-up of the hydraulic fluid spill experienced during prelaunch processing for STS-55.

The rupture of a LH2 umbilical retract hose sprayed hydraulic fluid in the aft fuselage. LV50 was one of several solenoid valves inspected during the clean-up operations and the outlet B-nut was removed as part of the inspection. Difficulty in gaining proper access to the valve when reinstalling the B-nut is believed to be the cause of the improper torque. As part of troubleshooting for IFA STS-55-V-01, LV50 was removed and replaced along with LV51 and PD3 and sent to Rockwell/Downey for further troubleshooting of the PD3 anomaly. CONCLUSION: The MPS pneumatic helium pressure decay was caused by a leak at the LV50 outlet B-nut. Insufficient torque on the B-nut was determined to be the cause of the leak. This fitting had been removed and reinstalled on the pad in support of an inspection resulting from spilled hydraulic fluid. CORRECTIVE_ACTION: The LV50 outlet B-nut was removed, inspected, and torqued to the proper value and a leak check was performed. The leak did not recur. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR** BH330154	**IFA** STS-55-V-13	C&T - Audio
INCO-03	**GMT:**		**SPR**	**UA**	**Manager:**
			IPR None	**PR**	x37990
					Engineer:

Title: MS 2 WCCS Crew Remote Unit (Serial Number 1037) Problem (GFE)

Summary: DISCUSSION: During STS-55, Mission Specialist 2 reported a problem with the crew remote unit (CRU) serial number 1037. The push-to-talk function on both intercom and air-to-ground was clipping. A battery replacement did not correct the problem.

The WCCS was not originally designed for a crewperson to operate the CRU on the flight deck/middeck while using the AIU in the Spacelab, or to operate the CRU in the Spacelab while using the flight/middeck AIU. While there has been a good RF link between the middeck AIU with the crewman in the Spacelab, a weak RF link exists between the flight deck and Spacelab. This is due to the RF energy having to propagate through the tunnel and flight/middeck opening. CONCLUSION: Troubleshooting at JSC failed to identify any anomalous condition within the CRU. The most probable cause of the problem was an RF drop-out from the location of the crewman to the AIU that was being used in the Spacelab module. CORRECTIVE_ACTION: Multiple postflight tests were performed and no problems were found. The unit will be returned to flight use. EFFECTS_ON_SUBSEQUENT_MISSIONS: None. Each crewman has a CRU, plus one spare CRU is always flown. Several other means of commmunications are available, such as hand-held microphones and the launch/entry suit communications carrier assembly (a hardline which may be used to backup the wireless communications system), should this failure recur.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR** BFCE 029F069,	**IFA** STS-55-V-14	C&T - Audio
INCO-04	**GMT:**		DR BH330151	**UA**	**Manager:**
			SPR	**PR**	x37990

IPR None **Engineer:**

Title: AIU-B Problem (GFE)

Summary: DISCUSSION: The crew reported that the audio interface unit (AIU)-B failed in the radio frequency (RF) mode, but still worked in the hardline mode. The crew also indicated that the audio terminal unit (ATU) to which AIU-B was connected was operating correctly.

CONCLUSION: The failure could not be reproduced after extensive testing. A probable cause of the failure could not be found. CORRECTIVE_ACTION: The AIU will be returned to flight status with low priority for use. EFFECTS_ON_SUBSEQUENT_MISSIONS: None. Three AIU's are flown on the Orbiter for five crewmember missions and five AIU's are flown on the Orbiter for seven crewmember missions. The crew has the option of manifesting a spare unit on each flight.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-15	OMS/RCS
PROP-02	**GMT:**		**SPR** 55RF06	**UA**	**Manager:**
			IPR 58V-0017	**PR**	
					Engineer:

Title: Right OMS Propellant Tanks Pressure Decrease (ORB)

Summary: DISCUSSION: The postflight data review of the orbital maneuvering system (OMS) deorbit burn indicated that both right OMS propellant tanks (oxidizer and fuel) experienced a pressure decrease during the final 73 seconds of the 2- minute 53-second burn. Coincident with the pressure decrease in the propellant tanks, there was a decrease in the pressure decay rate in the right OMS helium tank. This one helium tank is used to pressurize both right OMS propellant tanks. Following the burn, the propellant tank ullage pressures returned to their nominal lock-up values. A review of engine data indicated that engine performance during the burn was nominal.

The primary regulator outlet pressure under flow conditions is specified to be 257 psig +5/-7 psig with a lock-up pressure of no more than 266 psig. Accounting for pressure drop and transducer bias, the RP05 OMS pod fuel and oxidizer tank ullage pressures should be 247 and 250 psig +5/-7 psig, respectively. During the first two-thirds of the deorbit burn, the fuel and oxidizer tank ullage pressures were steady at 248 and 249.5 psig, respectively. At the end of the burn, following the 73-second period of decaying pressure, the fuel and oxidizer tank ullage pressures were 243 and 245.5 psig, respectively. Although the propellant tank ullage pressures at the end of the burn were within specification, the pressure decay indicates abnormal performance. STS-55 was the third flight of the RP05 OMS pod, its first two being STS-50 and STS-52. A review of STS-50 data indicated similar performance when the pressure decreased 4 psig during the final 60 seconds of the 3-minute 24- second deorbit burn. The fuel and oxidizer tank ullage pressures reached minimum values of 246.5 and 248 psig, respectively. During the STS-52 mission, there was no pressure decrease during the 2-minute 7-second deorbit burn; however, it was unusual in that following engine shutdown, the propellant tanks ullage pressures at regulator lock-up did not increase above the flow pressures. There are no cases of similar performace on other OMS pods and the performance seen on RP05 has only been exhibited during the deorbit burn. Regulator performance problems on RP05 had been observed during earlier processing flows. Prior to STS-52, waiver WK02618 was processed for the ROMS B-leg primary regulator flow pressure when it was out-of-specification low by 1 psi. Also, prior to STS-55, UA-2-A-0018 was processed when the ROMS B-leg primary regulator flow pressure was erratic. Extensive troubleshooting was performed at the Obiter Processing Facility (OPF) during the STS-58 flow. This troubleshooting included regular turnaround testing as specified in the Operational Maintenance Requirements and Specification Document (OMRSD); special tests to simulate the OMS-2, deorbit, and return-to-launch site (RTLS) abort dump burns; and a special test to simulate a burn with low-helium pressure at the start of the burn. Finally, the deorbit burn and the low-helium pressure burn simulations were performed on RP01 in the Hypergolic Maintenance Facility (HMF) to see if this pod exhibited similar performance. The simulated OMS-2 burn was performed with both the A- and B-legs on-line. The deorbit burn simulations were performed three ways; with both the A- and B-legs on-line, with the A-leg only on-line, and with the B-leg only on-line. The RTLS abort dump and the low- helium pressure burn simulations were performed with the A-leg only and the B- leg only on-line. Testing did repeat the propellant tank ullage pressure drops experienced during flight. Consistent with flight data, the pressure drops were seen only on RP05 and only when the helium inlet pressures and temperatures were low (less than 2000 psia and 20 degrees F). Testing also isolated the problem to the A- and B-leg primary regulators. Deorbit burn simulations isolating the secondary regulators did not indicate a pressure drop. Note that the magnitude of the pressure drops for the deorbit and RTLS dump burn simulations were within specification and would not affect orbital maneuvering engine (OME) performance. Also, results from normal OMRSD regulator testing indicated that all parameters were within specification and there was no performance degradation from previous tests. The most probable cause of the anomaly is believed to be leakage past the primary stage piston slip ring and/or abnormally high frictional load at the piston shaft seal in the A- and B-leg regulators. The primary stage piston provides the opening force for the regualator main poppet. Leakage or high friction would prevent the main poppet from opening as far as it should, therefore reducing helium flow through the main poppet. This piston leakage and/or high friction, the magnitude of which is temperature dependent, could be due to contamination or improper sizing of the rings and/or seals. Since the regulator poppet opening force is normally reduced at lower inlet pressures, the effect of piston leakage and/or high friction would be more significant at lower pressures. The decision has been made to fly STS-58 as-is. Regulator usage during STS-58 will be benign compared to past missions due to the large ullage in the propellant tanks and the relatively short duration OMS-2 and deorbit burns. Also, regulator performance during the three flights of RP05 and during STS-55 postflight testing has been consistent and indicates that the propellant pressurization system will support any required OMS burns during the STS-58 mission. A reflight decision will be made after STS-58 based on an evaluation of the flight performance. CONCLUSION: The propellant tank ullage pressure drops seen during flight were repeated during ground testing. Test results indicate that the problem is in the A- and B-leg primary regulators. The performance degradation seen during flight and in testing only occurs at low inlet temperatures and pressures. The most probable cause of the anomaly is leakage past the primary stage piston slip ring and/or abnormally high frictional load at the piston shaft seal due possibly to contamination or improper sizing of the ring and/or seal. CORRECTIVE_ACTION: The decision has been made to fly STS-58 as-is. Regulator usage during STS-58 will be benign compared to past missions due to the large ullage in the propellant tanks and the relatively short duration OMS-2 and deorbit burns. Also, regulator performance during the three flights of RP05 and during STS-55 postflight testing has been consistent and indicates that the propellant pressurization system will support any required OMS burns during the STS-58 mission. A reflight decision will be made after STS-58 based on an evaluation of the flight performance. EFFECTS_ON_SUBSEQUENT_MISSIONS: None expected.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-55-V-16	undefined
FDO-01	**GMT:**		**SPR** N/A	**UA**	**Manager:**
			IPR N/A	**PR**	x38343
					Engineer:

Title: SPoC PGSC data input problem (GFE)

Summary: DISCUSSION: At the STS-55 MOD/MER Crew Debriefing, the crew reported having data input problems with the Shuttle Portable Computer (SPoC) software on flight day 8. On the day before landing, the software would not accept the Commander's attempts to input an updated state vector. During this same period, the SPoC deorbit software would not accept the Pilot's request to enter flight specific information. The SPoC Team has been unable to duplicate the reported problems using either SPoC pre-mission configured software or the actual software retrieved from the STS-55 Payload and General Support Computer (PGSC) flight hardware.

The CDR stated that he first noticed that the SPoC computed acquisition of signal (AOS) and loss of signal (LOS) times for communication sites had drifted, indicating that the SPoC state vector was getting old. He exited the World Map application and immediately selected the SPoC state vector input option, intending to update the state vector from General Purpose Computer (GPC) Specialist Function Page (SPEC) 34. After backspacing over the displayed entries, the software would not accept keyboard entries for new data. He then exited the state vector option and returned to the World Map application. The PLT stated that he first attempted to change the deorbit opportunity window from 6 to 2 hours. After backspacing over the default entry of 6, he could not enter a new number. He also tried to change the vehicle weight, but the application would not accept new entries. He then exited and re-entered the application, accepting all of the default settings. He was then able to generate a Powered Explicit Guidance (PEG) 4 deorbit solution for KSC similar to the solution voiced up from the Mission Control Center (MCC). Both problems occurred on the final day on-orbit. The CDR stated that the Portable Audio Data Modem (PADM) had been used consistently throughout the mission for updating the SPoC software; however, he suspected that on some days the state vector was not getting updated after PADM operations were performed. After being notified of the anomaly, DM41/Frank Wood immediately attempted to independently duplicate the reported problems. Since the actual flight software could not be retrieved for several weeks, the STS-55 SPoC flight software which had been delivered pre-mission and placed under configuration control on the SPoC file server was configured on a PGSC Ground Development Unit (GDU) for testing. The software was updated with data files which had been prepared by the SPoC Team during the mission and uplinked to the crew via PADM. The estimated mission elapsed time when the problems occurred was also entered. The exact user actions that created the problems were not known at this time so multiple combinations of selections were performed. No problems were observed. The STS-55 SPoC flight software was retrieved when the three flight PGSC's became available in the PGSC hardware laboratory. By examining the contents of the hard disks on the flight units, it was determined that unit Serial Number 1074 was used on the forward flight deck and unit Serial Number 1072 was used for PADM operations on the middeck. Since SPoC softare is used primarily on the forward flight deck PGSC, the software retrieved from unit Serial Number 1074 was used for the majority of further testing. PGSC Serial Number 1074 was also retained for hardware diagnostic testing. The SPoC flight software was tested first by loading a "flight-like" CLASS III PGSC with the retrieved software. The crew was unavailable at this time so these tests were performed independently by DM41/Frank Wood. As before, multiple combinations of selections were performed in an attempt to recreate the problems and as before, no problems were observed. Once the crew was available, a meeting was held; it had been over five weeks since touchdown. After refamiliarizing themselves with the software, they were reasonably sure that they had duplicated the same user actions which had produced the on-orbit problems. The software performed normally throughout these tests. The STS-55 SPoC flight software had been tested by SPoC Team personnel both independently and jointly with the STS-55 crew with no replication of the reported in-flight anomaly. Nominal operation of the software was observed while testing pre-mission delivered SPoC software and post-mission SPoC flight software. Hardware diagnostic testing on PGSC Serial Number 1074 has also been performed by EK/Kip Chow with no problems found. This type of anomaly has never been reported previously either in training or flight. After thorough testing of the STS-55 SPoC flight software, the SPoC Team has been unable to reproduce the reported in-flight anomaly. By the nature of the anomaly, characterized by an adequate work-around with no mission impact, and the criticality III status of the PGSC hardware and software, the SPoC Team recommends closure of this in-flight anomaly as a non- reproducible occurrence. The anomaly has been documented with an internal discrepancy report which will remain open to maintain a history of future reference. CONCLUSION: The SPoC software and hardware were tested thoroughly without replication of the anomaly. CORRECTIVE_ACTION: Internal Discrepancy Reports SP206 and SP207 will remain open. EFFECTS_ON_SUBSEQUENT_MISSIONS: None