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A Summary of the Cassini System-Level Thermal Balance Test: Engineering Subsystems

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ABSTRACT

The Cassini spacecraft, NASA's mission to investigate the Saturn system, has undergone a system-level thermal balance test program to permit verification of the engineering subsystem thermal designs in the simulated worst-case environments. Additionally, other objectives such as functional checkouts, collection of thermal data for analytical model adjustment, vacuum drying of propellant tanks, and flight temperature transducer verification were also completed. In the interest of cost and schedule, transient off-Sunpoint conditions were not tested,

The testing demonstrated that the required system resources such as heater power and radiator area were adequate for all engineering subsystems. The only changes required from the results were related to the operation of some of the subsystems. In the instance of the thruster cluster assemblies, allowable flight temperature limits were exceeded for the assumed operational environment. The cause was attributed to a non-flight use of all the catalyst bed heaters. In order to assure that the propellant control assembly would be controlled by a computer-controlled heater within a specified temperature range, a change in the controlling flight temperature sensor set-point was instituted. Additionally, suitable heater capability was demonstrated for the propulsion tank and main engine thermal conditioning.

INTRODUCTION

TEST OVERVIEW - Cassini, NASA's mission to investigate the Saturn system, has recently completed its system-level thermal balance in February 1997. This test permitted the verification of the flight thermal design for most of engineering and science subsystem thermal designs. Other important test objectives including the operation of the engineering subsystems in a simulated space environment, collection of temperature data for analytical model calibration, vacuum drying of the propellant tanks, and verification of flight temperature sensor measurements were also achieved.

SCOPE - The purpose of this paper is to summarize the Cassini system-level thermal balance test from an engineering perspective. First, the Cassini mission and the spacecraft (S/C) configuration including the engineering subsystem thermal design approach are described. The system-level thermal balance test philosophy and configuration will be presented. A discussion of the test results including design modifications (during and after the test) will follow. Lastly, the validation of the thermal design through direct test results will be discussed.

MISSION DESCRIPTION - The Cassini spacecraft is planned for launch on a Titan IV/Centaur in October 1997. Since the launch energy is not sufficient for a direct trajectory, planetary gravity-assists from Venus (twice), Earth, and Jupiter enable the S/C to reach Saturn by July 2004 (see figure 1). After its Saturn arrival, the Huygens Probe will be released and will descend into Titan's atmosphere. The S/C Orbiter will tour the Saturn system for a period of four years, and it will investigate Saturn, its rings, its magnetosphere, and its satellites. The S/C heliocentric distance is expected to vary between 0.67 astronomical units (au) and 10.07 au for the primary launch opportunity. Other back-up and secondary opportunities can bring the S/C to a perihelion as close as 0.62 au. During the cruise to Saturn, the three-axis stabilized S/C normally points its high-gain antenna (HGA) toward the Sun. However, during trajectory correction maneuvers (TCM's), the S/C is turned away from Sun-point since the burn direction is not usually aligned with the solar vector.

Although there are three radioisotope thermoelectric generators (RTG's) on-board, their electrical output is insufficient to operate all engineering and science subsystems at once. Consequently, power sharing is

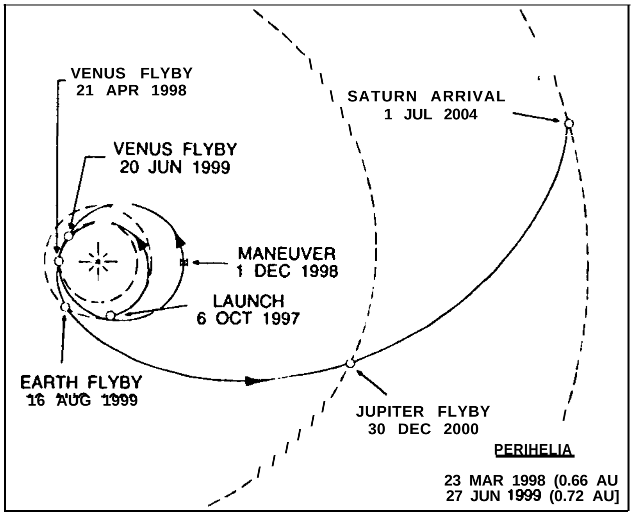


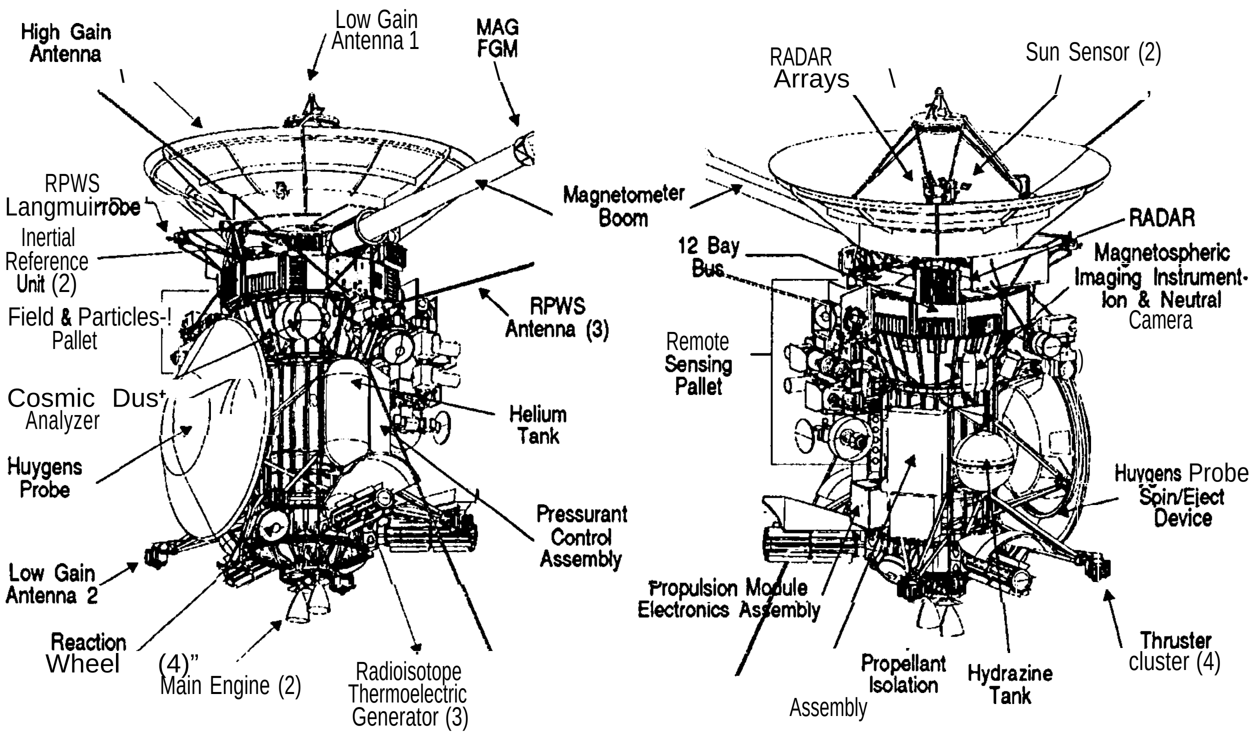
Figure 1: Cassini mission trajectory

instituted in most of the operational modes. These modes are discrete power states for the S/C. Use of these modes during the Saturn tour will permit science-gathering on a timeshare basis.

SPACECRAFT CONFIGURATION - The S/C configuration is shown in figure 2, The S/C is composedof the Orbiter and the Huygens Probe. The Orbiter is provided by JPL except for the HGA which is provided by the Italian Space Agency and the propulsion module subsystem (PMS) which is provided by Lockheed-Martin. The European Space Agency *(ESA)* provides **the** Probe. The engineering systems are mounted throughout the S/C, most notably on the Bus and the central body. The most dominant S/C feature is the propulsion module central body (PMCB) which is composed of the PMS, upper support structure assembly, and the lower equipment module (LEM). There are two main engines for redundancy, and during cruise, they are protected from micro-meteoroid impact by a deployable, large, hemispherical cover.

THERMAL DESIGN REQUIREMENTS -The large variation in heliocentric distance coupled with off-Sun maneuvers present a formidable challenge. Although the Galileo S/C trajectory was comparable, its thermal design did not have to contend with solar exposure due to off-Sun maneuvers at small heliocentric distances [1]. In order to provide mission trajectory design flexibility, the thermal design is required to tolerate heliocentric distances as small as 0.61 au. Electrical heater power demand, especially during the Saturn tour, had to be reduced to the minimum practical extent in order to maximize science [2].

THERMAL DESIGN APPROACH - Figure 3 illustrates the features of the S/C thermal design. The design must maintain acceptable temperatures for a wide range of mission heliocentric distances. In addition, the S/C may be oriented away from Sun-point during transient TCM's, thus exposing much of the S/C to direct insolation. The thermal control design strives to minimize



NOTE: Main engine cover not shown for clarity

Figure 2: Cassini spacecraft configuration

the sensitivity to the widely varying environment by employing mainly flight-proven passive techniques. Since the spacecraft is normally Sun-pointed, the HGA serves as a global shade. Low α_s / ε thermal paint has been applied to the HGA to mitigate temperatures. The HGA, itself, is thermally isolated from the bus to the extent practicable. The magnetometer boom, whose thermal design is inherited from Galileo, is protected by stowing it within its canister until the spacecraft is permanently beyond 0.97 au. Local shading is employed on the thruster clusters since they lie outside the shadow of the HGA. An improved fabrication technique for the bus louvers permits direct insolation, even at small heliocentric distances. During TCM's, the Huygens Probe with its considerable thermal capacitance, will be used as a shade to protect most of the Orbiter. In order to reduce electrical heater power demand, the S/C configuration has been driven toward an integrated thermal control subsystem to. the maximum extent possible (the Bus, PMS, and pallet-mounted science are thermally coupled).

Developmental techniques such as RTG waste heat utilization, variable radioisotope heater units (VRHU's) on the thruster clusters, a reverse louver set on the remote sensing pallet (RSP) have been implemented to reduce electrical heater power demand, This demand was further reduced by utilizing radioisotope heater units (RHU's) wherever practical, especially in the PMS and Probe. These techniques saved the electrical power output equivalent to one RTG. Implementation of other standard techniques completes a robust design. The SIC is covered extensively with thermal blanketing, Louvers on the bus and science pallets help ***accommodate large*** variations in heat loads. Electrical heaters thermostatically controlled by the Command and Data Subsystem (CDS) are used where the allowable flight temperature (AFT) range is relatively small. Other fixed electrical heaters assist in maintaining acceptable temperatures. The cryogenic instruments such as the Composite Infrared Spectrometer Optics Assembly and the Visible and Infrared Mapping Spectrometer-Infrared are protected from RTG heating by shades placed nearby the RTG's.

Extensive analysis has been performed to develop the thermal design and has undergone peer and formal reviews. Developmental testing was performed where: system resources required quantification (e.g., heater power and radiator area), special thermal issues required resolution (e.g., solar focusing on main engine nozzle interior), and new thermal control hardware feasibility required demonstration (e.g., RTG waste heat utilization, VRHU's, and the reverse louver) [3].

SYSTEM-LEVEL THERMAL BALANCE TEST

OBJECTIVES - The primary test objective was to demonstrate that the various engineering subsystem and instrument thermal designs maintained temperatures within AFT limits in simulated worst-case hot and cold thermal environments. Secondary test objectives included: 1) functionality verification of engineering subsystems or instruments where a flight-like environment is essential; 2) demonstration of in-specification functional performance of subsystems and instruments when in a

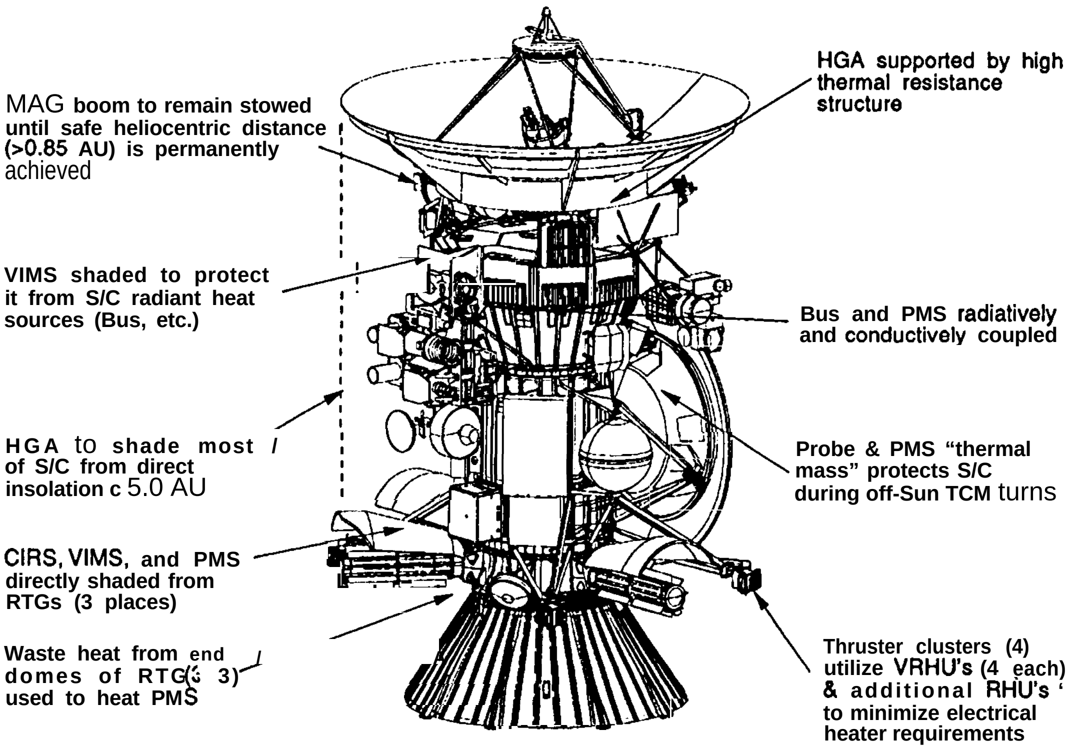


Figure 3: Spacecraft thermal design

flight-like environment including temperature range; 3) demonstration of functional margin when beyond allowable flight temperature limits, but within flight acceptance test limits; 4) obtain data to calibrate and adjust the system-level S/C analytical thermal model for use in mission operations; 5) verification of flight temperature transducer measurements where feasible; and 6) perform vacuum drying of PMS tanks of isopropyl alcohol that had been used as a referee fluid during acoustic testing. Secondary objectives were accomplished in a manner that would not interfere or take precedence over the primary objective.

FACILITIES - Testing was performed at JPL's largest space simulator chamber from January 17, 1997 to February 6, 1997. It is a side-opening cylinder 8.2 meters in diameter and 26 meters high. For space simulation, the liquid nitrogen cooled chamber shrouds are finned aluminum panels which are painted black on all surfaces that face the test volume.

The off-axis solar simulation system consists of an array of thirty-seven xenon arc-lamps, each with a maximum power rating of thirty kilowatts, an integrating lens unit which condenses the light for the lamps, a fused-quartz chamber penetration window, and a 7.0 meter diameter collimating mirror mounted at the top of the chamber (shown in figure 4).

Prior to this test, the HGA was baked-out in the same facility with the solar simulator system providing 1.5 Suns and the shroud maintained at 95°C. A single lamp failed and sprayed debris which caused another eleven lamps to fail. There were replacement lamps available which brought the number of functional lamps to thirty-five. Operating the lamps near thirty kilowatts was believed to be the cause of the lamp failure. After much consideration, the project management recommended that twenty-five lamps be used for the thermal balance test and

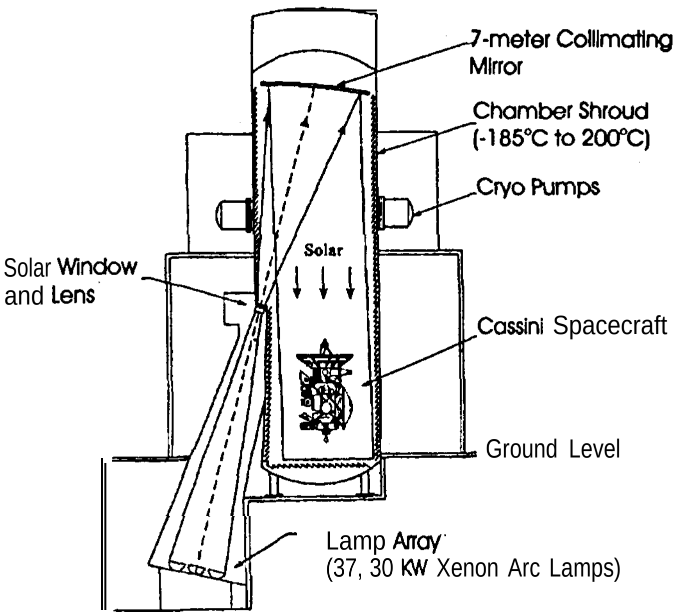


Figure 4: 8.2-meter chamber

that the maximum lamp power level would not exceed twenty-five kilowatts. This resulted in a maximum solar simulation of 1.6 Suns (versus 2.7 Suns at 0.61 au). The risk trade-off was a lower solar simulation with a spare lamp reserve versus a high solar simulation with no lamp reserve and a potential for losing solar simulation capability altogether.

During the decision-making process, the JPL thermal engineering team indicated that the HGA is primarily a radiative boundary condition for the rest of the S/C since the HGA is conductively isolated via titanium support struts. Since the S/C is protected with thermal blanketing to a large extent, the rest of the S/C is rather insensitive to HGA temperature, In the terms of the HGA thermal design, the HGA experienced temperatures beyond the maximum allowable temperature limits during the bake-out without any ill effects. The thermal balance test would use several solar irradiance levels, and the system-level analytical thermal model would be adjusted with this data to determine S/C temperatures for a 0.61 au condition.

In preparation for the test, an extensive solar mapping was done prior to and after phase one to verify the uniformity of the chamber solar simulator irradiance. Mapping results showed that the solar beam was spatially uniform within 7% of the nominal value measured at the Kendall radiometer location. Although using a xenon lamp source simulates a slightly different energy spectrum than the Sun, these differences only affect "yellow" surfaces. The surfaces which were illuminated during the test include the HGA, which is painted primarily with PCBZ white paint, and the deployed mag boom and thruster clusters which have second-surface Kapton outer layers. Effects from the use of xenon lamps cause about 20% less energy to be absorbed by these "yellow" surfaces. This effect will be accounted for in the post-test model correlation, but does not affect the overall conclusions on thermal design acceptability.

TEST CONFIGURATION - The S/C test configuration in the space simulator chamber for phase one, two and in-air are shown in figures 5 and 6. The S/C was suspended from the chamber top structure by six stainless steel cables with three attach points on the chamber side. Furthermore, the S/C was anchored to the chamber floor by three stainless steel cables at the bottom. The six suspension cables carry the weight of the S/C while the three anchoring cables prevent the S/C from swinging in the event of an earthquake.

The magnetometer boom on the S/C remained stowed throughout the entire test. A deployed stub boom was installed 0.9 m from the chamber floor on the Probe-side of the S/C. This installation permitted the solar illumination of the boom. The boom structure and magnetometers consisted of developmental hardware.

A cold target assembly was positioned fifteen centimeters from the radiator plane of the RSP instruments. The purpose of this target was to provide a -243"C background temperature for the CIRS OA and VIMS IR radiators so that these instrument focal planes

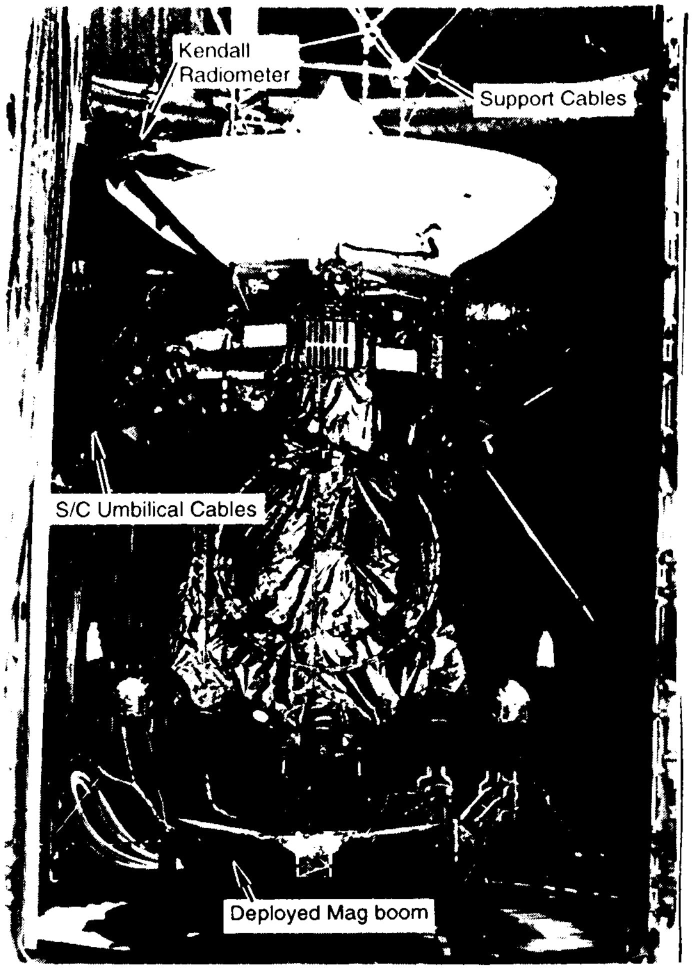


Figure 5: Phase one configuration

Figure 6: Phase two and in-air configuration

could be cooled below -188°C during functional testing. The cold target was cooled to -243°C with liquid helium during functional testing. However, during the majority of the thermal balance test, liquid nitrogen was used on the cold target since the focal planes were not operational.

The S, Ka, Ku, and X band waveguides had test interfaces that disturbed the continuity of the flight path. These interfaces were necessary to absorb any radio frequency energy generated by the S/C when the radio frequency subsystem or the radio science instruments were activated.

The hardware fidelity requirement dictated that all S/C equipment be the flight hardware. However, there were notable exceptions. The RTG's and RHU's, being radioactive in nature, were simulated with electrical counterparts. The Huygens Probe was represented with a geometric mock-up with a flight-like interface since the Probe is fairly isolated from the S/C. A previously-fired linear separation assembly was attached to the LEM to represent the post-separation configuration (free-flight condition) from the launch vehicle.

INSTRUMENTATION - A total of 524 chromel-constantan thermocouples (TC's) and two low-temperature cryogenic diodes were used for temperature measurement of S/C and test support hardware. Typically, 26-gage TC wire with Teflon insulation was used, however, where the heat leak through the TC was critical, 30-gage TC wire was used. In areas used where the TC was illuminated by the solar simulator, care was taken to cover the TC with a surface finish whose optical properties were similar to the surface underneath the TC. A majority of the TC'S were located where compliance with AFT limits could be directly observed. Cryogenic diodes were attached to the cold target since the operational temperature range was beyond the calibration accuracy of data acquisition system. A number of TC'S served as the control measurement for temperature-controlled test heaters. Lastly, other TC'S were used to verify the readings of flight temperature transducers, There were a total of 380 platinum resistance thermometers (PRT's) onboard the WC; these represent the total number of flight PRT's except those located on the Huygens Probe and RTG's. Of these, only the ones read by the spacecraft remote engineering units (191 total) were verified; the rest are located internal to each subsystem and their partial validation will be done in the future.

There was a total of 76 test heater circuits. Generally, the test heaters could be divided into two types: RHU/RTG simulators and flight and test hardware safing/acceleration. The RTG and RHU simulators have the mechanical configuration of the flight units and utilized custom electrical heaters. In order to protect the flight hardware from a S/C power failure during testing, safing

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heaters were installed on vulnerable items. Additionally, these heaters were used to accelerate achievement of thermal equilibrium when possible, and they were also exercised to accelerate the warm-up to ambient conditions prior to a chamber break. Guard heaters were installed on ground support cabling to reduce heat leaks to or from the S/C via this heat path.

During those test cases where the solar simulator is activated, the solar irradiance level was measured by a Kendall cavity radiometer,

TEST CASES - For engineering subsystems, there are three driving cases: i) hot operation (with the main engine (MEA) cover closed and Probe attached during cruise and at perihelion); ii) hot operation (with MEA cover open and Probe jettisoned at Saturn); iii) cold non-operation (usually at Saturn with MEA cover open and Probe jetissoned). The science instruments had similar driving cases, except hot operation usually occurs at Saturn.

The planning of the test matrix was a huge undertaking since there were competing factors: identifying opportunities to verify all engineering subsystems and instrument thermal designs including special tests (i.e., exercising CDS thermostatic heater control or hardware temperature sensitivity to power), cost, and schedule. Non-flight S/C power modes were implemented to create bounding thermal design conditions which sometimes created conflicts between engineering subsystems and science instruments (e.g., hot bus versus instruments coupled to bus in cold state). In order to obtain verification of the thermal designs in the most expeditious manner, the extreme hot and cold thermal environment was simulated in the first phase. If any serious design inadequacies were discovered such as deficient heater power, thermal designs could be modified before starting phase two. Thermal design modifications would be re-tested in the same extreme environments as phase one. Pre-test predictions indicated that the extreme hot case temperatures were near room temperature so this case was conducted first. Special tests were appended to the end of certain cases where the steady state represented the appropriate initial condition (e.g. main engine warm-up after equilibrium reached for cold case). Contingency plans were formulated for thermal designs with adjustability (e.g. radiator area or obtaining RHU power sensitivity).

The test was divided into three phases. The intent of phase one was to simulate bounding thermal environments encountered when the S/C configuration represents Saturn tour (i.e., Probe simulator removed and the MEA cover open, see figure 5). Hot operation and cold non-operation testing was performed. In phase two, the S/C was in the cruise configuration (i.e., Probe simulator attached and the MEA cover closed, see figure 6), and it was exposed to the bounding Cruise thermal environment, The use of a black Kapton “sock” on the deployed magnetometer boom thermal design was also tested. For the last phase, in-air testing was performed to obtain pre-launch temperatures for the PMCB (performed at ambient conditions while the S/C was a representative launch configuration: Probe simulator attached and the MEA cover closed) to understand *in-flight conditioning of* the PMS tanks. The spacecraft power states for each test case are summarized in Table 1.

Figures 7 and 8 shows the actual test timelines for phases one, two and in-air.

TEST RESULTS -The discussion will focus on engineering subsystems, the science instruments are addressed elsewhere [4]. The phase summaries present the major results with design modifications caused by these results being discussed in the subsequent section, The HGA, Bus, PMCB, MEA and Thruster Cluster Assembly (TCA) test results are summarized in figures 9 through 16.

<u>Phase One Summary</u> - In phase one, most engineering subsystems demonstrated an acceptable thermal design with margin. Although the HGA was not tested with the maximum mission solar irradiance, results showed good agreement with test predictions and acceptable temperatures are expected for flight.

Excellent results from the RTG waste heat concept [5] were demonstrated in this phase. The thermal blanketing approach coupled with heat from the RTG simulators showed a very effective distribution of radiant heat throughout the PMCB. This implementation was the largest risk to the overall spacecraft thermal design since this was the first system-level verification of the concept. Spatial temperature gradients (which give an indication of heat distribution) showed fairly uniform temperatures throughout the PMCB.

Results for the coldest configuration in phase one confirmed that an existing Bus supplemental heater needs to be activated to assure the Bus does not violate its minimum AFT limits. This configuration was designed to be a conservative bound from a power state and RTG/RHU degradation perspective and, because of this, temperatures of the PMCB showed little or no margin from the AFT limits. The diode assembly located on the MEA marginally violated its limit by 2 “C. This violation is not predicted for flight nominal conditions.

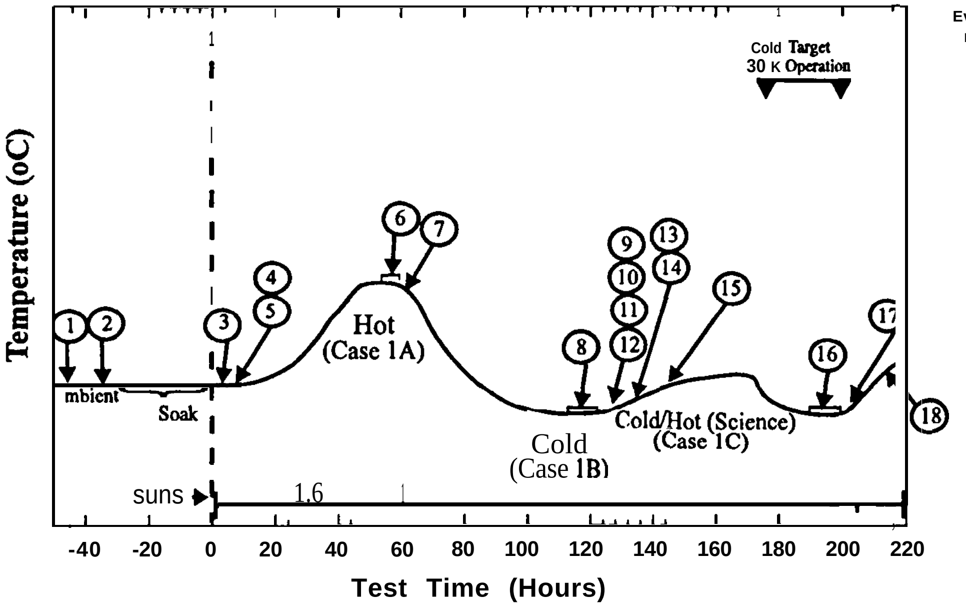
The catalyst-bed 90-minute and MEA 8-hour thermal warm-up transients were done during this phase. They both showed acceptable warm-up capability with their current heater sizes.

AFT limit violations were discovered in two areas. The stellar reference units (SRU's) CCD's violated their operating AFT limits of -30 “C by 10“C. This discrepancy was expected since subsystem test thermal results also showed this CCD temperature violation. The instrument engineer opted not to modify the thermal design and waive the upper operating AFT limits for the Saturn tour. Until the Earth flyby, the SRU supplemental heater will also be turned OFF since development test hardware has shown a significant thermal coupling between the optics and CCD. In addition, a minor violation of the TCA upper AFT limit was observed during the 1.6 Sun case. This violation was partially aggravated by an over-conservative catalyst bed heater configuration on each cluster. All four catalyst bed heaters were operated while results showed that from an operational standpoint only two catalyst bed

Table 1: STV power states

ubsystem	Assembly	Teat Case				
		1A	1B	1C	2A	2B
IAG	Stowed					
	FGM	ON/OFF	OFF	OFFION	OFF	OFF
	VSHM	ON/OFF	OFF	OFFION	OFF	OFF
	Deployed					
	FGM	OFF	OFF	ON	ON	OFF
	VSHM	OFF	OFF	ON	OFF/ON	OFF
CAS	MAG Alignment Coil	ON	OFF	OFF	OFF	OFF
ACS	IRUs	ON	OFF	OFF	OFF	ON
	Flight Comp. (Bay 1)	Both ON	1 ON	1 ON	Both ON	Both ON
	BAIL (Bay 9)	ON	OFF	OFF	OFF	ON
	RW Elex (Bay 10)	ON	OFF	ON	OFFION	ON
	Accel Efex (Bay 12)	OFF	OFF	OFF	OFF	OFF
	SRU-A	ON	ON	ON	OFF	OFF
	SRU-B	OFF	OFF	OFF	ON	ON
	PMSEA (Bay B)	Both ON	EGE OFF VDE ON	EGE OFF VDE ON	EGE OFF VDE ON	Both ON
	EGAs	ON	OFF	OFF	OFF	ON
	RWAS 1-4	3 ON	OFF	3 ON	OFF/3 ON	3 ON
FIS		ON	OFF	OFF	OFF	ON
DS	Flight Computers	Both ON	Primary ON	Both ON	Both ON	Both ON
SR	SS Recorder (Bay 9)	Both ON	Primary ON	Both ON	Both ON	Both ON
Pws		ON	OFF	ON	OFF	ON
ADAR		OFF	OFF	ON	OFFION	OFF
DA		Sleep	OFF	ON	OFF	Sleep
S	NAC and WAC	Sleep	OFF	ON	OFF	Sleep/ON
MS		Sleep	OFF	ON	OFF	OFF/Sleep
VIS		ON	OFF	ON	OFF	OFF
IRS		Sleep	OFF	ON	OFF	OFF
'MS		Sleep	OFF	ON	OFF	ON
IMI		Sleep	OFF	ON	OFF	ON
4PS		Sleep	OFF	ON	Sleep	ON
MS	TCA Primary CAT-Bed Htr	ON	OFF	ON	ON/OFF	ON
	TCA Secondary CAT-Bed Htr	OFF	OFF	OFF	ON	OFF
?OBE	RFE	OFFION	OFFION	OFF	OFF	OFF
	PSAe and RUSO	OFFION	OFF/ON	OFF	OFF	OFF

The following **assemblies** were **ON/ACTIVE** throughout the STV **test:** MAG Boom Rate **Limiter,** Sun **Sensors,** PPS end **RFS.**



Event No.	Description
1	S/C Baseline Test
2	Clew Chamber, Pumpdown
3	Nitrogen Flush Start Cooling shrouds
4	Turn OFF Purge
5	Configure for Case 1A
6	4 hour Probe Checkout
7	Configure for Case 1B
e	MEA 8 hour transient
9	CAPS HV, CDA I/F Tact
10	ISS Interference Test
11	Radar Turn-ON
12	RWA Turn-ON
13	Accelerate with tsar heaters
14	Configure for Case 1C Instrument
15	Functional lasts
16	CIRS, VIMS Functional Tests& Microphonics/IF Test
17	Configure for Backfill
18	Turn ON Purge

Figure 7: Phase one timeline

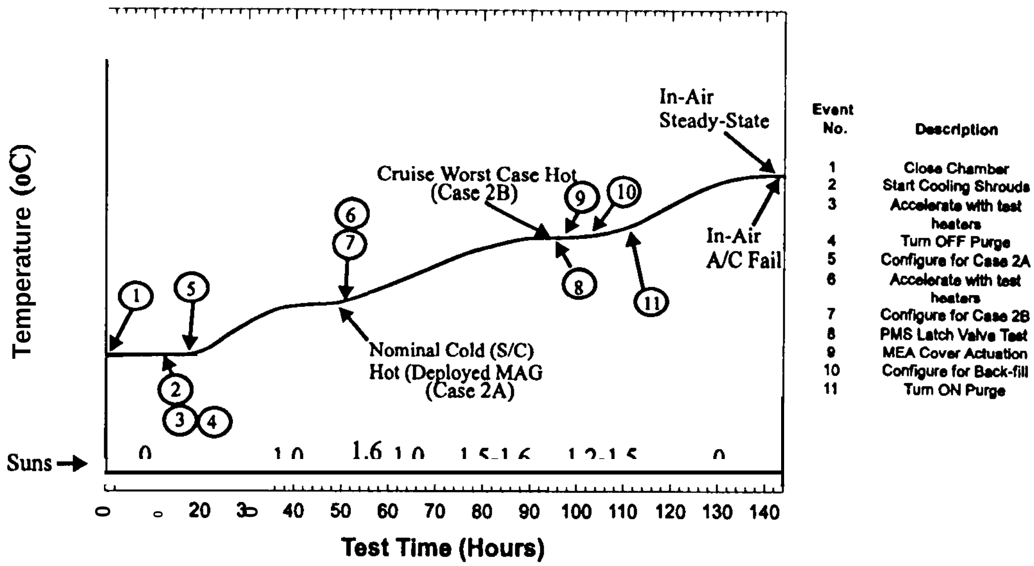


Figure 8: Phase two and in-air timeline

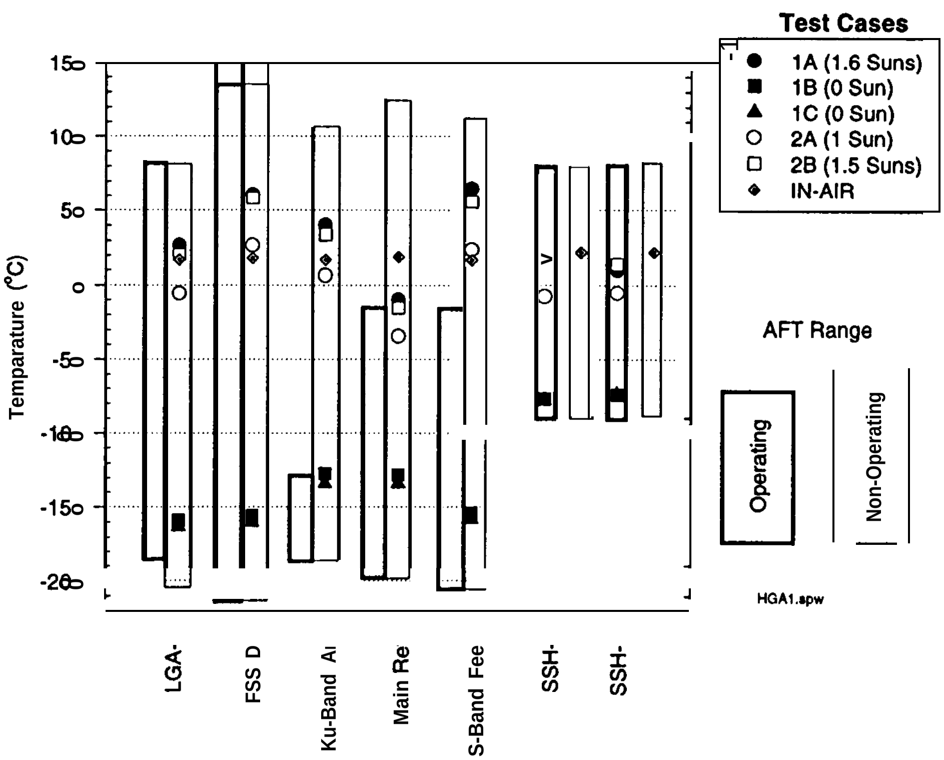


Figure 9: GA/S sensors ST v results

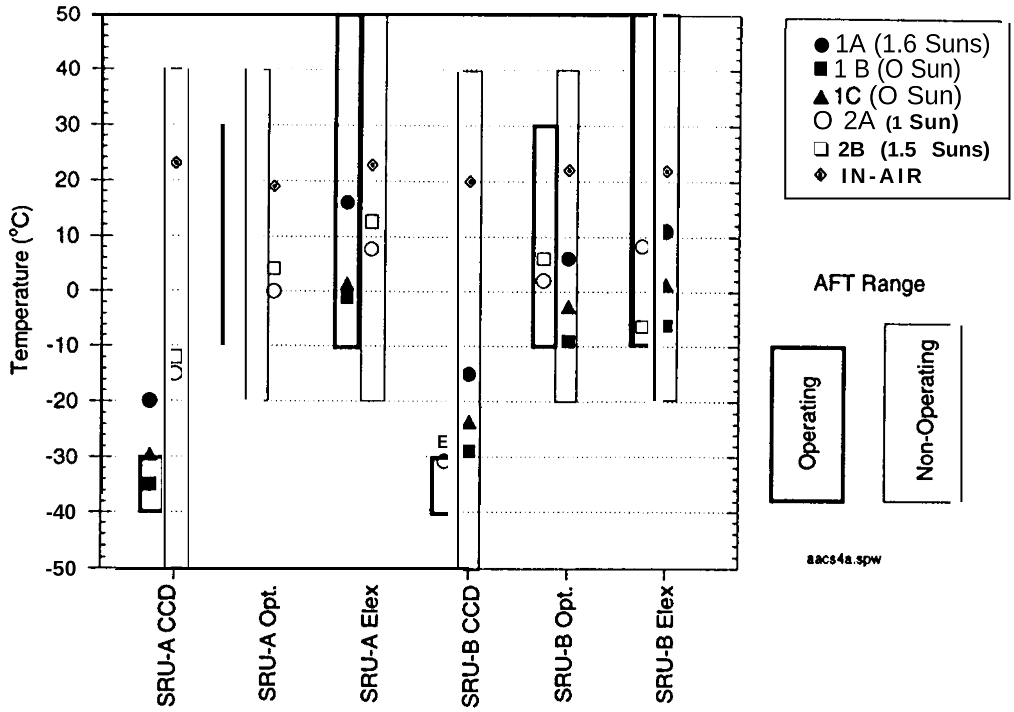


Figure 10: SRU STV results

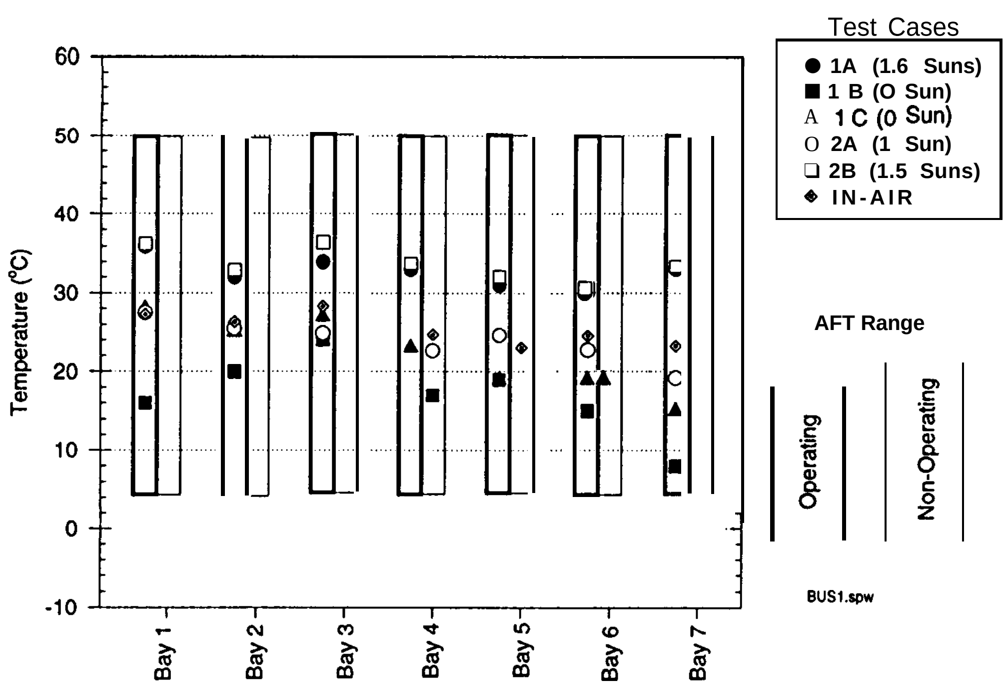


Figure 11: Bus bays 1-7 STV results

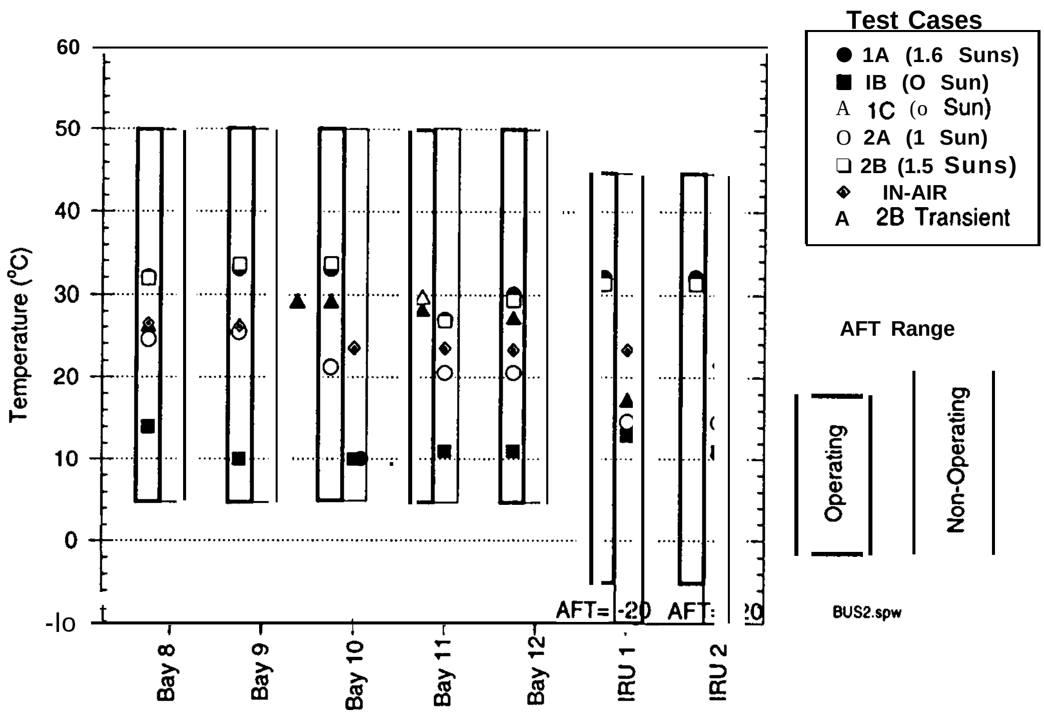


Figure 12: Bus bays 8-12 STV results

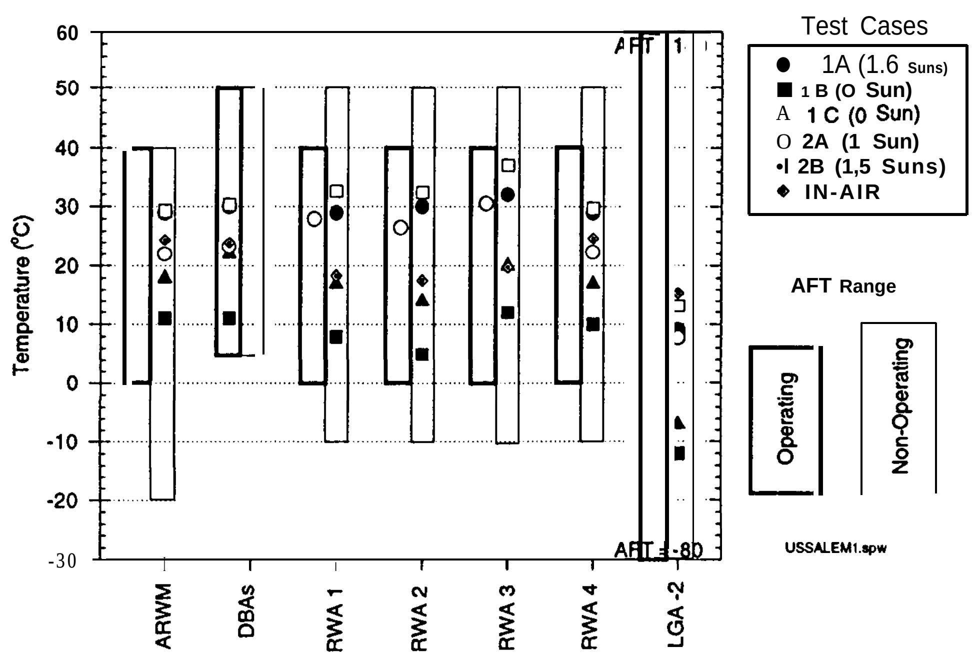


Figure 13: USSA/LEM STV results

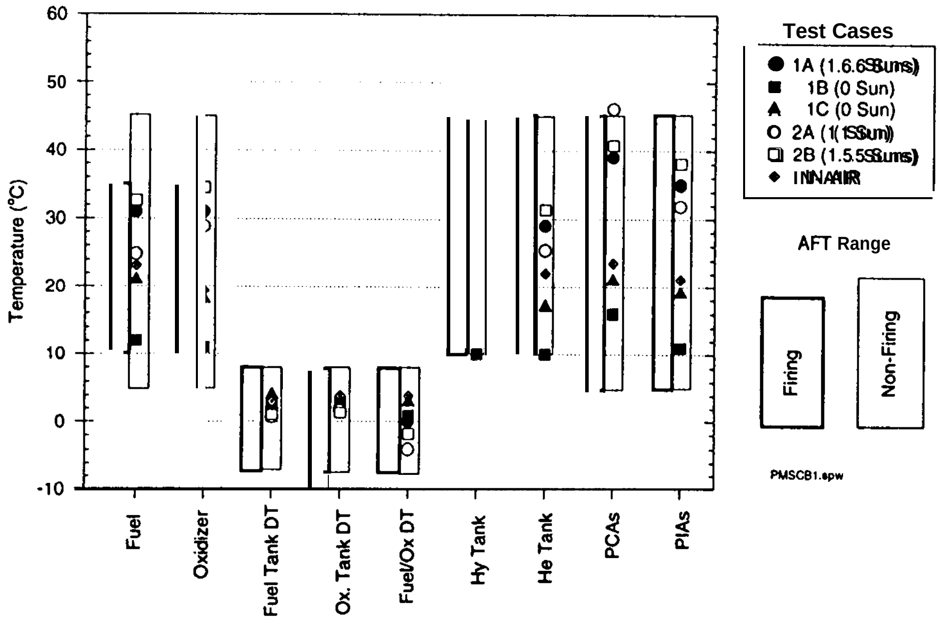


Figure 14: PMS STV results

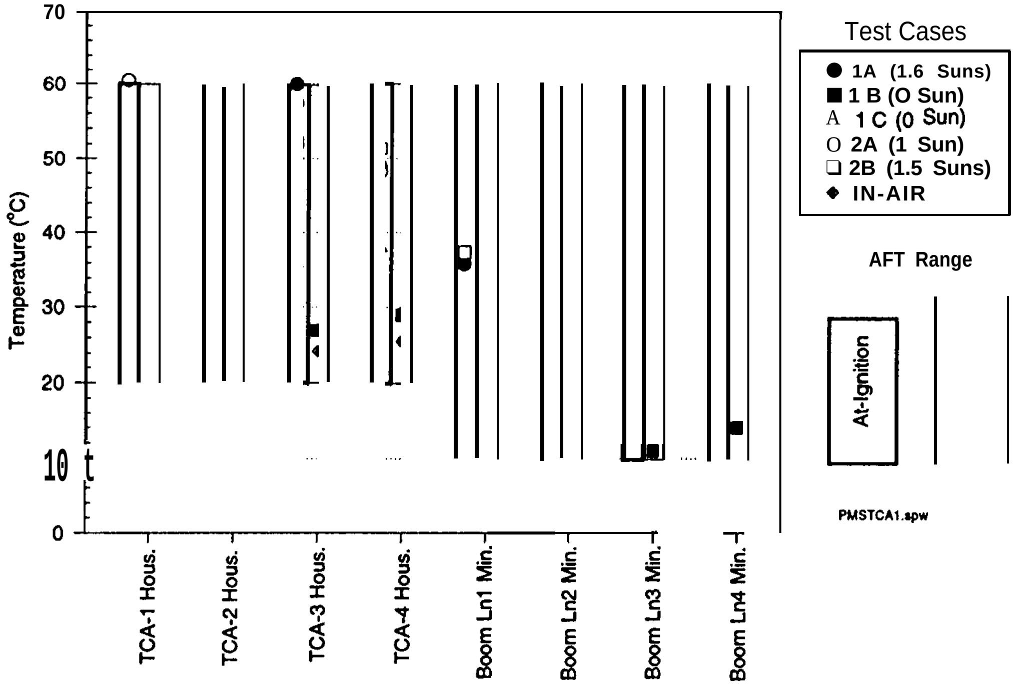


Figure 15: TCA STV results

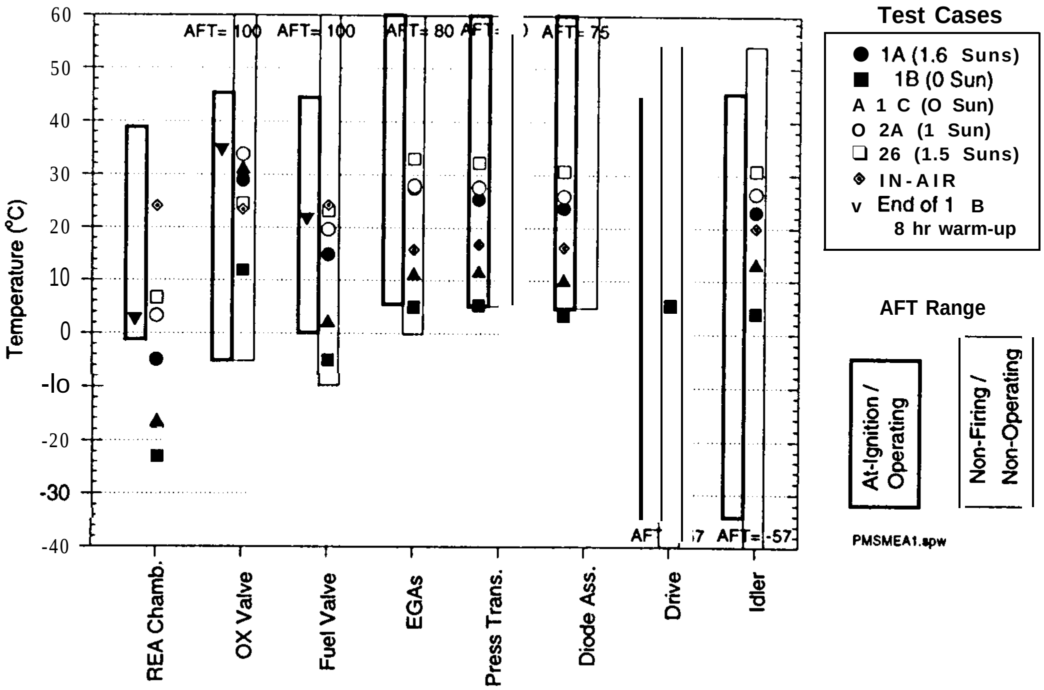


Figure 16: MEA STV results

heaters per cluster are required. Although this was a surprise, subsequent test points showed that turning OFF two heaters on each cluster will yield acceptable flight temperatures.

The correlation of flight temperature transducers showed very few discrepancies which need correction, The HGA and rocket engine assembly (REA) showed unacceptable discrepancies as shown in Table 2. Although these errors are being investigated, incorrect calibration curves for the transducers could be partial contributors.

The evacuation of referee fluid of the bipropellant tanks was also successfully completed in phase one. Residual isopropyl alcohol, used in the previous dynamic testing, was purged to an acceptably low level by the middle of phase one (this precluded need for further evacuation in phase two)

Phase Two Summary - With the success of phase one, the project management requested reducing the amount of phase two testing. Project management also decided to install aperture covers over several of the RSP instruments to protect them for any further possible molecular contamination from the HGA. In addition, further contamination was observed on the outer layer of the MLI blankets during the phase one and two chamber break. The source is partially blamed on the blanket lacing cord but other additional sources are still being investigated.

Phase two was designed to show temperatures for a nominal cruise configuration and to show the impact of closing the MEA cover and having an attached Probe on S/C temperatures. A nominal cruise configuration would show actual design temperature margin, as opposed to phase one where design margin was shown with respect to spacecraft power configuration. This nominal configuration also yielded realistic bounds for conditioning the bipropellant tanks after launch for optimal MEA operation. The first test case showed that cruise temperatures for the Bus and PMCB were well within AFT limits and that the thermal design was robust. The only surprise in the nominal case was the operation of the pressurant control assembly (PCA) CDS-controlled heaters. These heaters, which are used to mitigate propellant condensation and migration on the panel, demonstrated that the control set-point has to be modified for flight, otherwise the upper AFT will be violated for one of these panels. This test point also provided verification of the magnetometer boom structure temperature acceptability at 1 Sun and a correlation point for all HGA hardware including the Sun sensors. The second test point confirmed that the same power state used in phase one but with the Probe attached and cover closed produced temperature increases on the PMCB as expected.

The end of phase two also completed other functional tests on hardware, including the MEA cover and PMS latch valves. The functional objectives were all successfully met at the completion of this phase.

In-Air Summary - The in-air testing objective was to characterize the PMCB temperature sensitivity to air inlet temperature in the payload fairing. This would yield a

Table 2: Engineering flight transducer discrepancies

Channel	Description	Test Temperature Range (C)	Largest Deviation between Flight and Test Transducers (C)
E-1 743	REA-A Head End Temperature (TE1)	-23 to 16	7
E-1748	REA-B Head End Temperature (TE3)	-25 to 16	8
E-1 752	REA-A Fuel Boom Temperature (TEB2)	16 to 44	5
E-1943	REA-A Head End Temperature (TE2)	-23 to 17	10
E-1 945	REA-A Fuel Valve Temp. (TEV2)	-11 to 26	7
E-1948	REA-B Head End Temperature (TE4)	-25 to 17	8
E-1950	REA-B Ox Valve Temperature (TEV3)	10 to 44	18
E-2500	HGA Ku-Band Feed Beam 1 Temperature 6	-134 to 39	11
E-2501	HGA Ku-Band Feed Beam 2 Temperature 7	-133 to 43	13
E-2502	HGA Ku-Band Feed Beam 5 Temperature 8	-131 to 38	7
E-2503	HGA Ku-Band WG Beam 3 Temperature 9	-96 to 23	21
E-2505	HGA Reflector Rear Surface Temperature 1	-162 to 18	5
E-2506	HGA Reflector Rear Surface Temperature 2	-132 to 20	24
E-2507	HGA X/Kc/Ku-Band Feed Temperature 5	-115 to 40	42
E-2508	X-Band FSS Temperature 3	-160 to 61	20
E-2509	Ku/Ka-Band FSS Temperature 4	-160 to 61	19
E-2510	LGA1WG Temperature 11	-163 to 34	11

bound for the pre-launch temperature of the bipropellants and thus bound the time required for warm-up to the cruise target temperature by using the bipropellant tank heaters. Although the configuration in the test chamber was not thermally identical to the payload fairing, valuable correlation data was obtained for bounding convective effects by running a pre-launch simulation with the chamber closed and the facility air-conditioning at the coldest measured temperature. Results showed that although the PMCB was directly driven by air-conditioning temperatures, the bipropellant tanks are still substantially warmer (~4°C) than the surroundings because of the local effects of the RTGs and their convective heating. This implies that the current flight warm-up period is consistent with the target and pre-launch temperatures.

DESIGN MODIFICATIONS - The three areas which had surprises, SRU CCD's, TCA's and PCA's, require no hardware design change. These deficiencies will be corrected by waiving the AFT limits and modifying the use of operational heaters. The rest of the engineering subsystems require no design modifications for flight.

REQUIRED RESOURCES - The pre-test estimates for instrument radiator area, electrical heater power, number of louvers, and number of RHU's were demonstrated to be adequate. The cornerstone of the S/C thermal design was the RTG waste heat utilization for the PMCB. The radiant heating of the PMCB without the need for electrical heat enables the engineering subsystems and science instruments to be thermally coupled to the PMCB. This approach facilitated the S/C thermal design and was verified during this test.

CONCLUSION

The system-level thermal balance test met all of its objectives successfully. Primarily, the Sun-pointed thermal design was verified in the worst extreme thermal environments. Where verification was not direct, a sufficient amount of test data was taken to enable analytical verification. Although facility limitations only provided a maximum solar irradiance of 1.6 Suns, sufficient correlation data was obtained for the HGA and all other au dependent thermal designs. A correlated spacecraft thermal model will be used to verify hot (2.7 Suns) cases for solar dependent hardware (e.g. HGA, TCAS, Sun sensors.. etc).

LESSONS LEARNED

A successful thermal balance test requires detailed planning and coordinated efforts by not only the thermal design people but the whole spacecraft team. A detailed test plan and procedure should be reviewed and finalized weeks before the beginning of the test. To minimize risk to flight hardware, significant planning should be implemented into the contamination control plan used during testing. Lastly, the importance of instrumentation reliability and redundancy should never be sacrificed for budget constraints.

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