97 ES-33

A Summary of the Cassini System-Level Thermal Balance Test: Science Instruments

Glenn T. Tsuyuki, Virgil Mireles, Edward 1. Lin and Arturo Avila
Jet Propulsion Laboratory, California Institute of Technology

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 science instrument thermal designs in the simulated worst-case environments. Additionally, other objectives such as functional checkout, collection of thermal data for analytical model adjustment, and flight temperature transducer verification were also attained. In the interest of cost and schedule, transient of f-sunpoint conditions were not tested,

The test demonstrated that the required system resources such as heater power and radiator area were adequate. In the instance of the Cosmic Dust Analyzer, allowable flight temperature limits were violated, but this problem is being addressed without a significant impact to system resources or thermal design robustness, Finally, the thermal acceptability of a black Kapton "sock" was demonstrated for the magnetometer boom.

INTRODUCTION

SCOPE - The purpose of this paper is to summarize the Cassini system-level thermal balance test from a science instrument perspective. First, the Cassini mission and the spacecraft (S/C) configuration including the instrument thermal designs are described. The underlying philosophy behind the thermal balance test is discussed with particular attention to methodology and objectives. A summary of the test results in terms of thermal design adequacy is presented.

TEST OVERVIEW - Cassini, NASA's mission to investigate the Saturn system, has recently completed its system-level thermal balance test 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 included the operation of the instruments in a simulated space environment, collection of temperature data for analytical model calibration, and verification of flight temperature sensor measurements were also achieved.

MISSION TRAJECTORY - 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 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

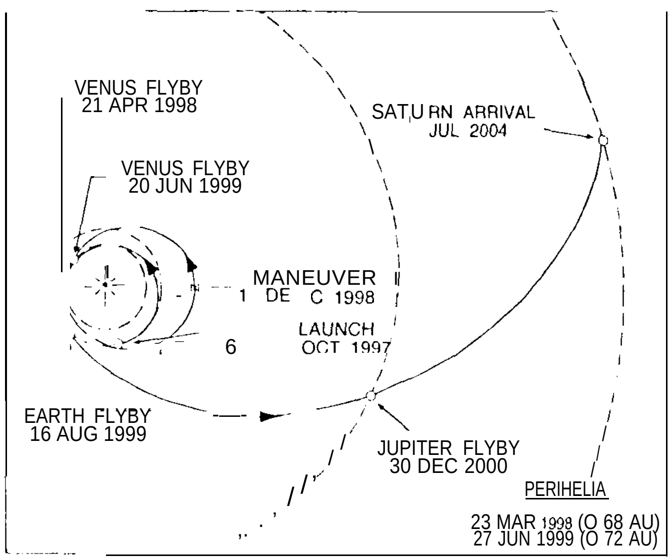


Figure 1: Cassini mission trajectory

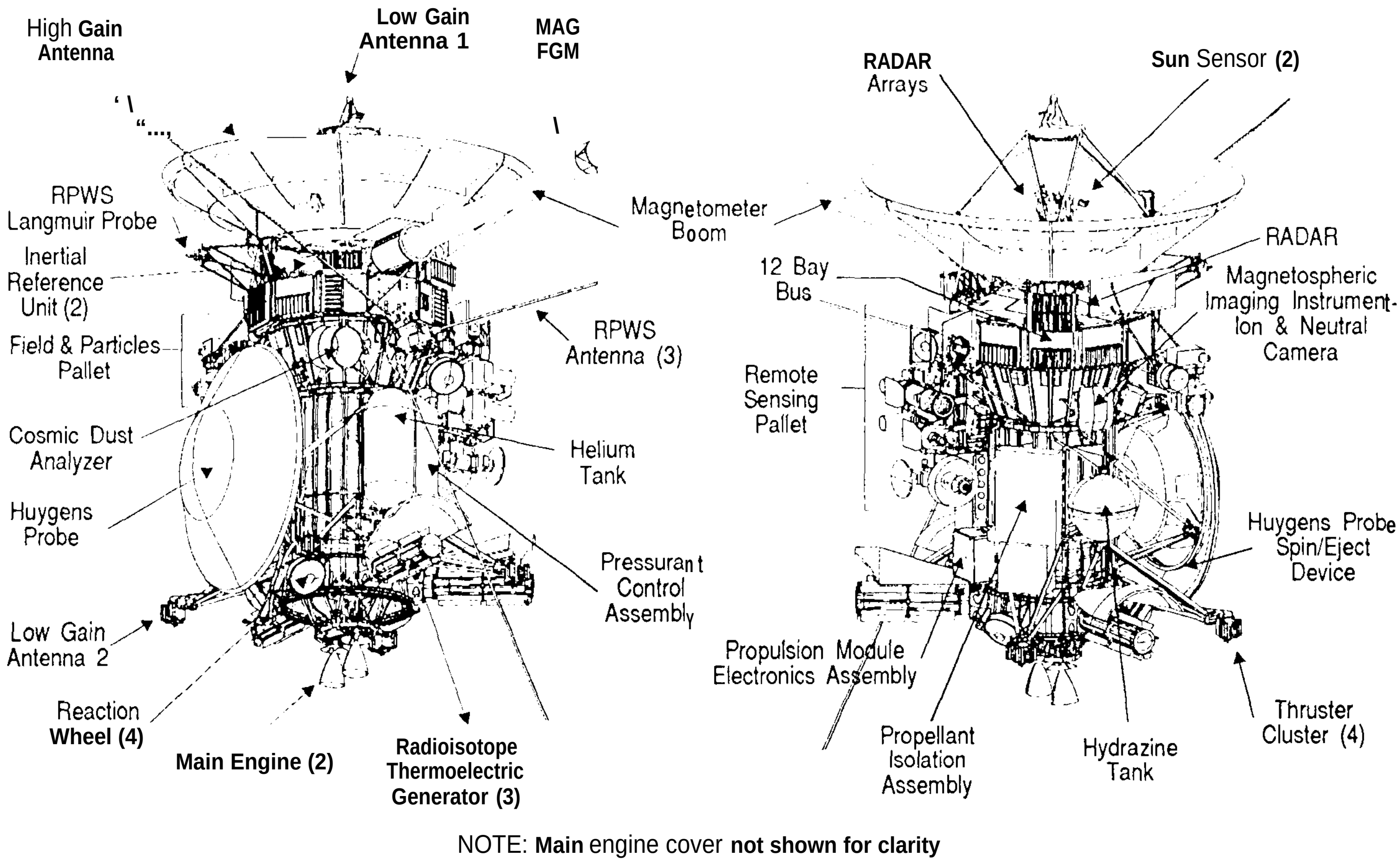


Figure 2: Cassini S/C configuration

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, there is not adequate electrical power to operate all engineering and science subsystems at once. Consequently, power sharing is instituted in some 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 composed of the Orbiter and the Huygens Probe. The European Space Agency (ESA) provides the probe. The most dominant S/C feature is the propulsion module central body (PMCB) which is composed of the bus, upper support structure assembly (U SSA), propulsion module subsystem (PM S), 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, spherical cover. The bus houses much of the SIC electronics. The Italian Space Agency (ASI) provides the fixed HGA.

The science instruments are mounted throughout the SIC, most notably on the Huygens Probe and two science pallets (the remote sensing pallet (RSP) and the fields and particles pallet (FPP)). The RSP houses the wide- and narrow-angle cameras, and several spectrometers. Instruments that measure Saturn cloud composition, Saturn ring environment, and Saturn satellite interaction with Saturn's magnetosphere are located on the FPP. Two of the four FPP instruments (CAPS and MIMI LEMMS) require articulation for science gathering. The dual magnetometers are located on a deployable boom which mounts to the bus. A pivoting dust analyzer and a plasma and radio wave instrument are attached to the USSA. Table 1 summarizes the science payload acronyms, principal investigator, and lead institution. An international team consisting of approximately 1300 people in 16 European countries and 3000 people in 32 states in the United States is involved in various aspects of the mission including design, fabrication, planning, and flight operations.

THERMAL DESIGN REQUIREMENTS -The large variation in heliocentric distance coupled with off-sun maneuvers presents 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 was 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. The allowable flight temperature (AFT) limits are specified in project documentation [21.

_ **Table 1: Cassini science payload**

Instrument	Principal investigator/Tean~ Leader	*Instrument Provider*
REMOTE SENSING		
Composite Infrared Spectrometer (CIRS) Optics Assembly (OA) Central Electronics Assembly (CEA)	Virgil Kunde	Goddard Space Flight Center (GSFC)
Imaging Science Subsystem (1SS) Narrow Angle Camera (NAC) Wide Angle Camera (WAC)	Carolyn Porco	JPL
Ultra-violet Imaging Spectrometer (UVIS)	Larry Esposito	University of Colorado
Visible and Infrared Mapping Spectrometer (VIMS) Infrared (IR) Visible (V)	Robert Brown	JPL/Officine Galileo, Florence, Italy
FIELDS, PARTICLES, AND WAVES		
Cassini Plasma Spectrometer (CAPS)	David Young	Southwest Research Institute
Cosmic Dust Analyzer (CDA)	Eberhard Grüen	Max-Planck-institute fur Kernphysik, Heidelberg, Germany
Ion and Neutral Mass Spectrometer (INMS)	J. Hunter Waite, Jr.	GSFC
Magnetometer Subsystem (MAG) Fluxgate Magnetometer (FGM) Vector/Scalar Helium Magnetometer (V/SHM)	David Southwood	Imperial College, London, England (FGM); JPL (V/SHM)
Magnetospheric Imaging Instrument (MIMI) Charge-Energy-Mass Spectrometer (CHEMS) Ion and Neutral Camera (INCA) Low Energy Magnetospheric Measurement System (LEMMS)	Tom Krimigis	Applied Physics Laboratory, John Hopkins University
Radio and Plasma Wave Science (RPWS) Dipole Antenna Assembly Langmuir Probe (LP) Magnetic Search Coil (MSC)	Donald Gurnett	University of Iowa
MICROWAVE REMOTE SENSING Radio Science Subsystem (RS)	Arvyds Kliore	JPL
Radar Subsystem (RADAR)	Charles Elachi	JPL/ASI
PROBE MISSION		
Huygens Probe	--	ESA

DESIGN RESPONSIBILITY - In the early thermal design conceptualization, the Jet Propulsion Laboratory (JPL) learned from its Galileo experience that it should retain as many of the instrument bulk thermal designs as practicable. This facilitated the development of the design in a timely fashion since the system-level thermal design was highly integrated. Furthermore, the general design knowledge of each instrument would be retained at a system-level to support flight operations. However, there were instances where the instrument was thermally isolated from the S/C or the thermal design would affect the quality of science. In these situations, the bulk thermal design responsibilities were given to the instrument team. Table 2 specifies those instruments that were responsible for their entire bulk thermal design. The definition of the thermal interface,

Table 2: instrument Teams Responsible for Bulk Thermal Control

Instrument	Rationale
CIRS OA inducting focal plane	Thermally isolated from RSP
CDA	Complex mechanical configuration for multiplier
INMS Sampling Area	Thermally isolated from INMS electronics
ISS-NAC & WAC focal planes	Thermally isolated from main camera body
RPWS MSC	Science data quality dependent on thermal blanket construction
VIMS-IR & V focal planes	Thermally isolated from spectrometer body

the roles and responsibilities of the instrument team and JPL, the specification of the instrument thermal environment were documented in thermal interface design documents or in memoranda of understanding.

DESIGN APPROACH - Figure 3 illustrates the salient features of the instrument 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 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. During TCM's, the Huygens Probe with its considerable thermal capacitance, will be used as a shade to protect most of the Orbiter. In addition, the outer layer of thermal blankets consists of second surface aluminized Kapton (low α_s/ε) to reduce the effect of the sun. 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 RSP, CDA, MIMI-INCA, RPWS antenna assembly are coupled with the PMCB; the FPP, RADAR, and Huygens Probe RFE are coupled with the bus. Developmental techniques such as RTG waste heat utilization to maintain PMCB temperatures and a reverse louver set on the RSP to enhance coupling to the PMCB have been implemented to reduce electrical heater power demand. Additionally, standard louvers have been employed on the science pallets to accommodate instrument power variations without the need for electrical heat. An improved fabrication technique for the bus louvers permits direct insolation, even at small heliocentric distances.

Implementation of other standard techniques completes a robust approach. instruments are covered extensively with thermal blanketing with the exception of apertures and thermal radiators. 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. 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-power electrical heaters assist in maintaining acceptable temperatures when instruments are on or off. The cryogenic instruments such as CIRS OA and the VIMS-IR are protected from radiant heating by shades placed nearby the bus and 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, variable radioisotope heater units (VRHU's), and the reverse louver) [31.

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 while the S/C is sun-pointed. Although the transient off-sun condition during TCM'S is the worst hot thermal environment for some hardware, the cost and schedule impacts associated with the test fixture and a larger test facility made this testing unattractive. As an alternative, off-sun testing was conducted during subsystem thermal development testing for solar-sensitive items [3]. 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 flight-like environment including AFT range; 3) demonstration of functional margin when beyond AFT 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 flight 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

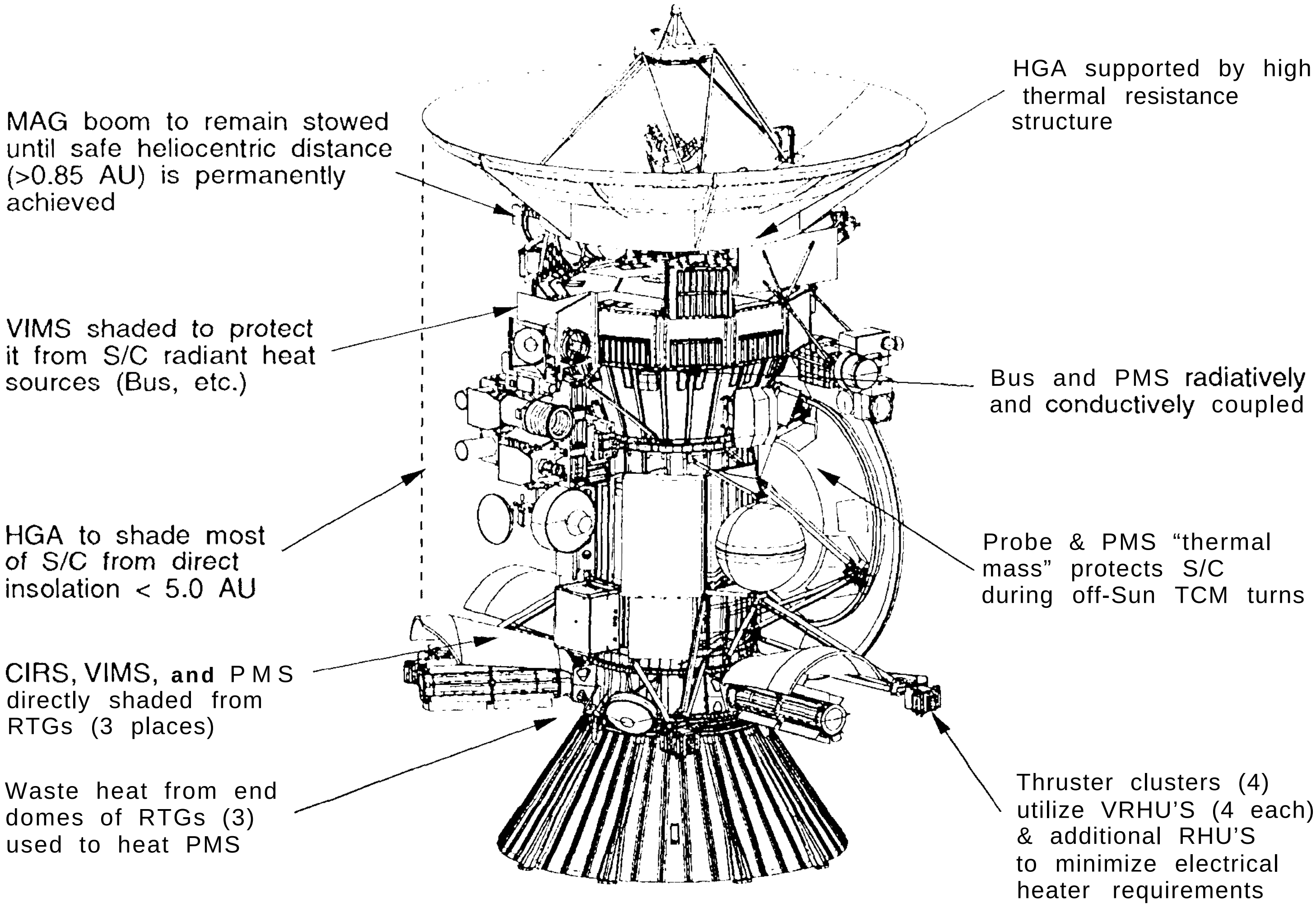


Figure 3: System-level thermal design schematic

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

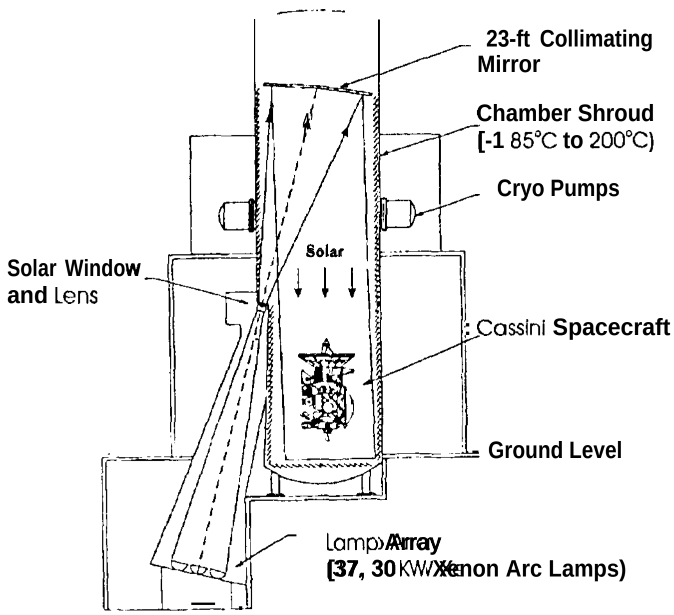


Figure 4: Space simulator chamber schematic

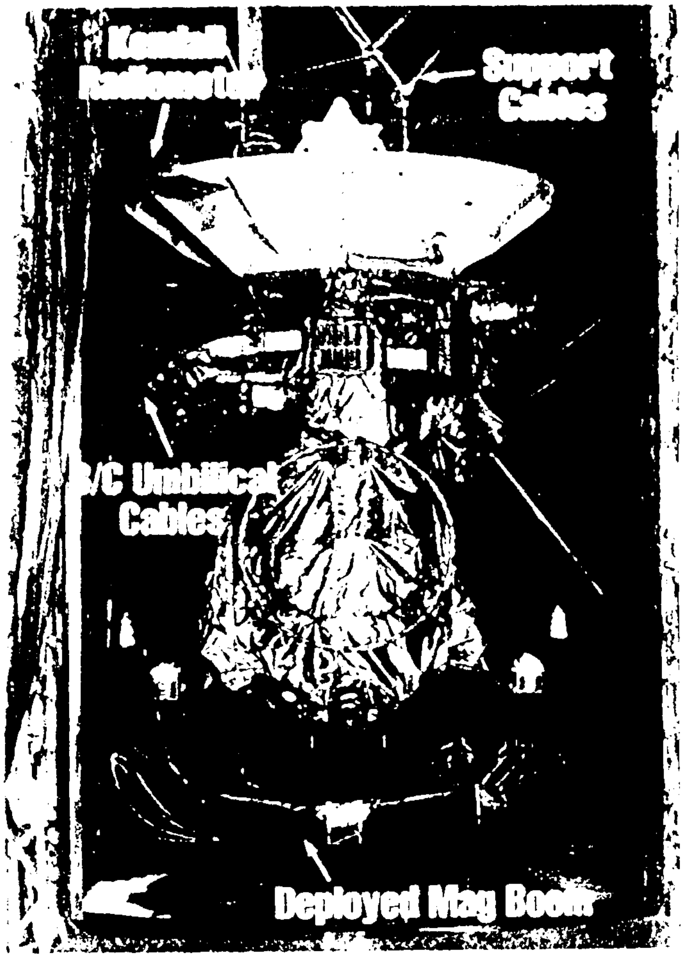


Figure 5: Phase one S/C configuration

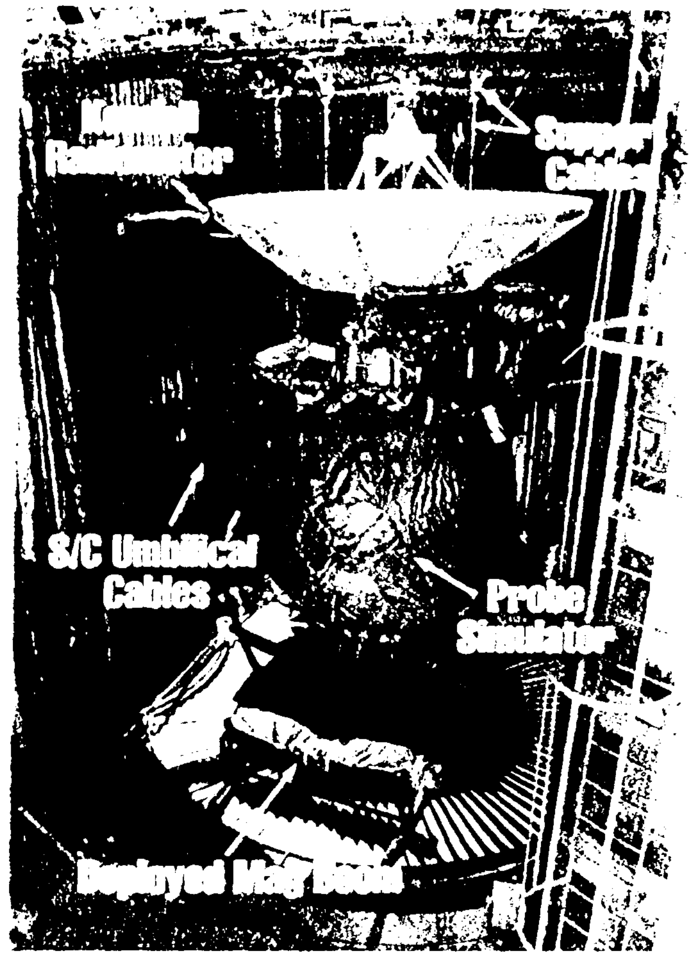


Figure 6: Phase two S/C configuration

recommended that twenty-five lamps be used for the thermal balance test and 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 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 portions of the HGA experienced temperatures in excess of the maximum AFT limits during the bake-out without any ill effects. The thermal balance test would use three distinct 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.

TEST CONFIGURATION The S/C test configuration in the space simulator chamber is 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 four stainless steel cables at the bottom. The six suspension cables carry the weight of the S/C while the four anchoring cables prevent the S/C from swinging in the event of an earthquake.

The flight 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 SIC. This installation permitted the solar illumination of the boom. he 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 could be cooled below -188°C during functional testing. The cold target was cooled to or below -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 SIC when the

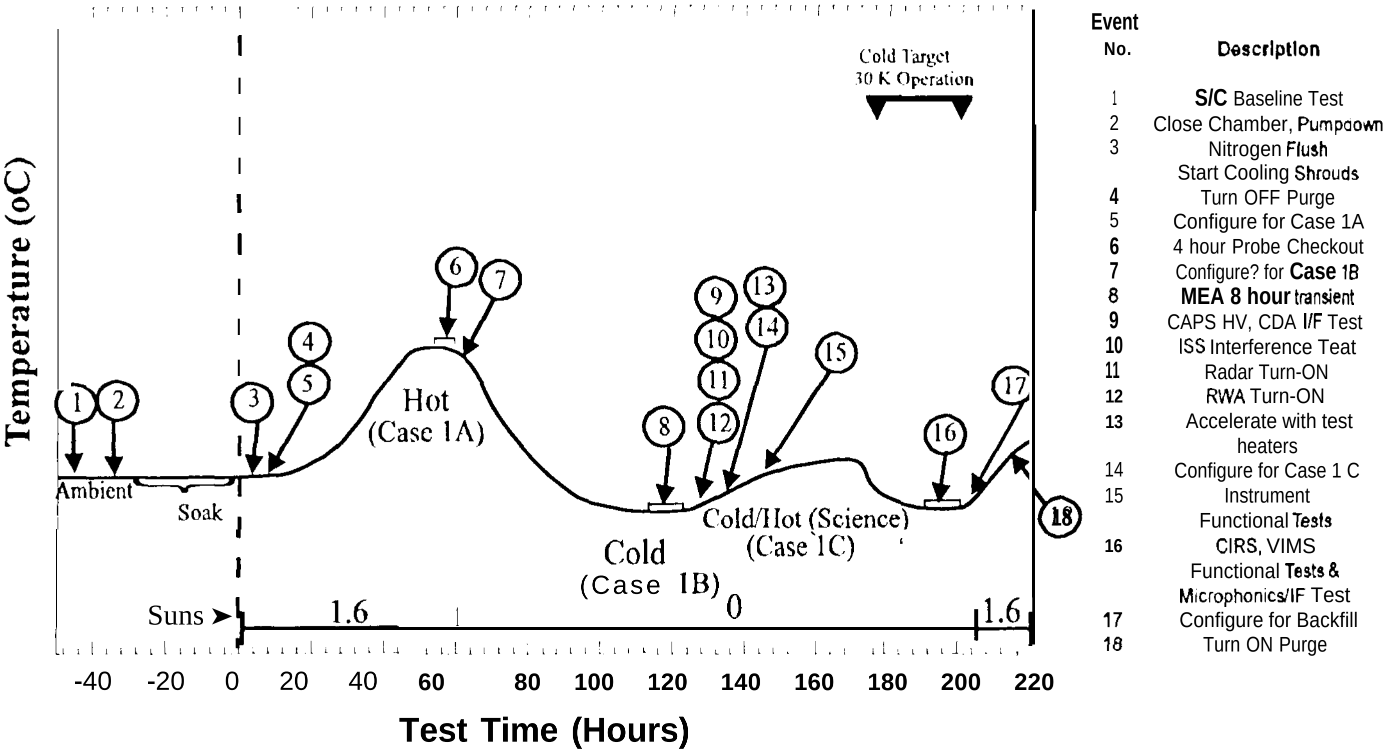


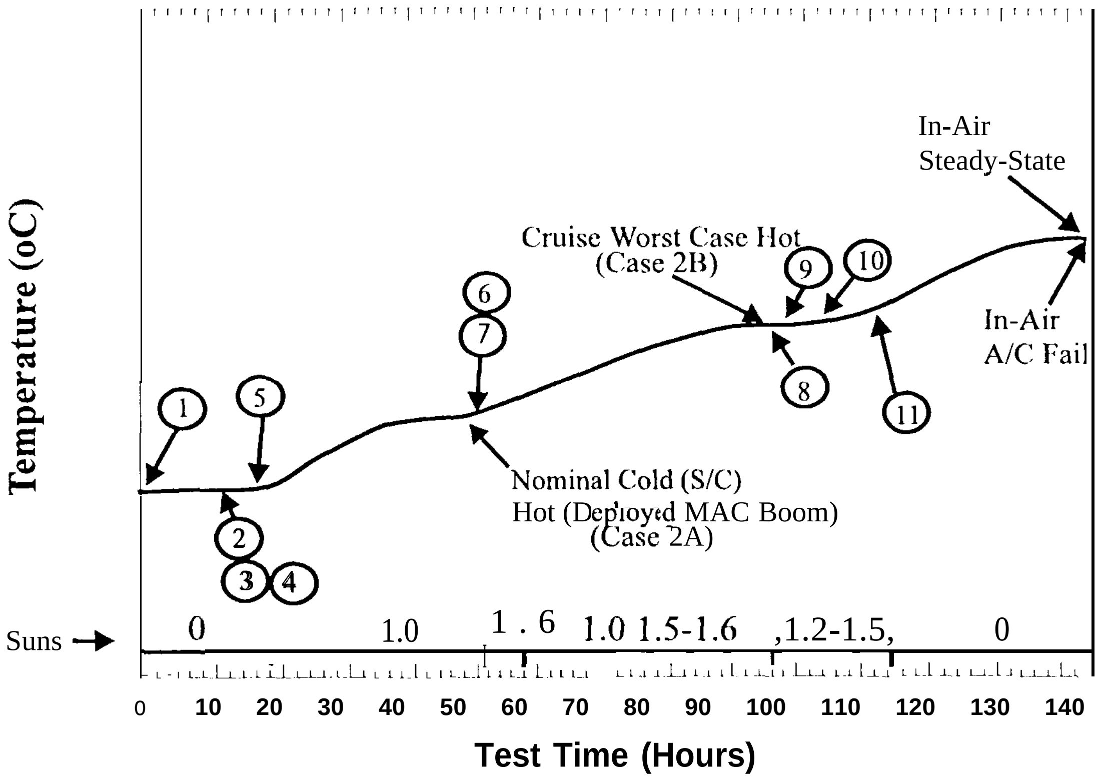
Figure 7: Phase one timeline

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 attachment 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. The MIMI-LEMMS engineering model was used because of the uncertainty that its rotation bearings could withstand operation in a gravity field. Because of schedule difficulties, INMS and CAPS provided engineering models for the test. As previously mentioned, the deployed magnetometer boom was developmental hardware.

INSTRUMENTATION - A total of 519 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 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 191 platinum resistance thermometers (PRT's) onboard the S/C; this represents the total number of flight PRT's except those located on the Huygens Probe and RTG's.

There was a total of 75 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 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



Event No.	Description
1	Close Chamber
2	Start Cooling Shrouds
3	Accelerate with test heaters
4	Turn OFF Purge
5	Configure for Case 2A
6	Accelerate with test heaters
7	Configure for Case 2B
8	PMS Latch Valve Test
9	MEA Cover Actuation
10	Configure for Back-fill
11	Turn ON Purge

Figure 8: Phase two timeline

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 - Typically, there are four driving cases: cold and hot operation (both at Saturn); and cold and hot non-operation (at Saturn and perihelion, respectively). In this instance, the hot non-operation case (perihelion) is not truly a driving case since it is examined for compliance after the thermal designs accommodate the other driving cases. In addition, a cold "sleep" case was necessary. "Sleep" mode is the lowest power operational state where portions of the instrument are powered off for power sharing purposes. The instrument should be able to quickly transition into the fully operational state from "sleep." In general, the "sleep" mode dissipates less instrument power than cold operation, and operational AFT limits must be maintained.

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 SIC 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 "sleep" 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 the 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. Probe maintenance after equilibrium reached for hot non-operation). 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 SIC configuration represents Saturn tour (i. e., Probe simulator removed and the MEA cover open, see figure 5). Hot operation,

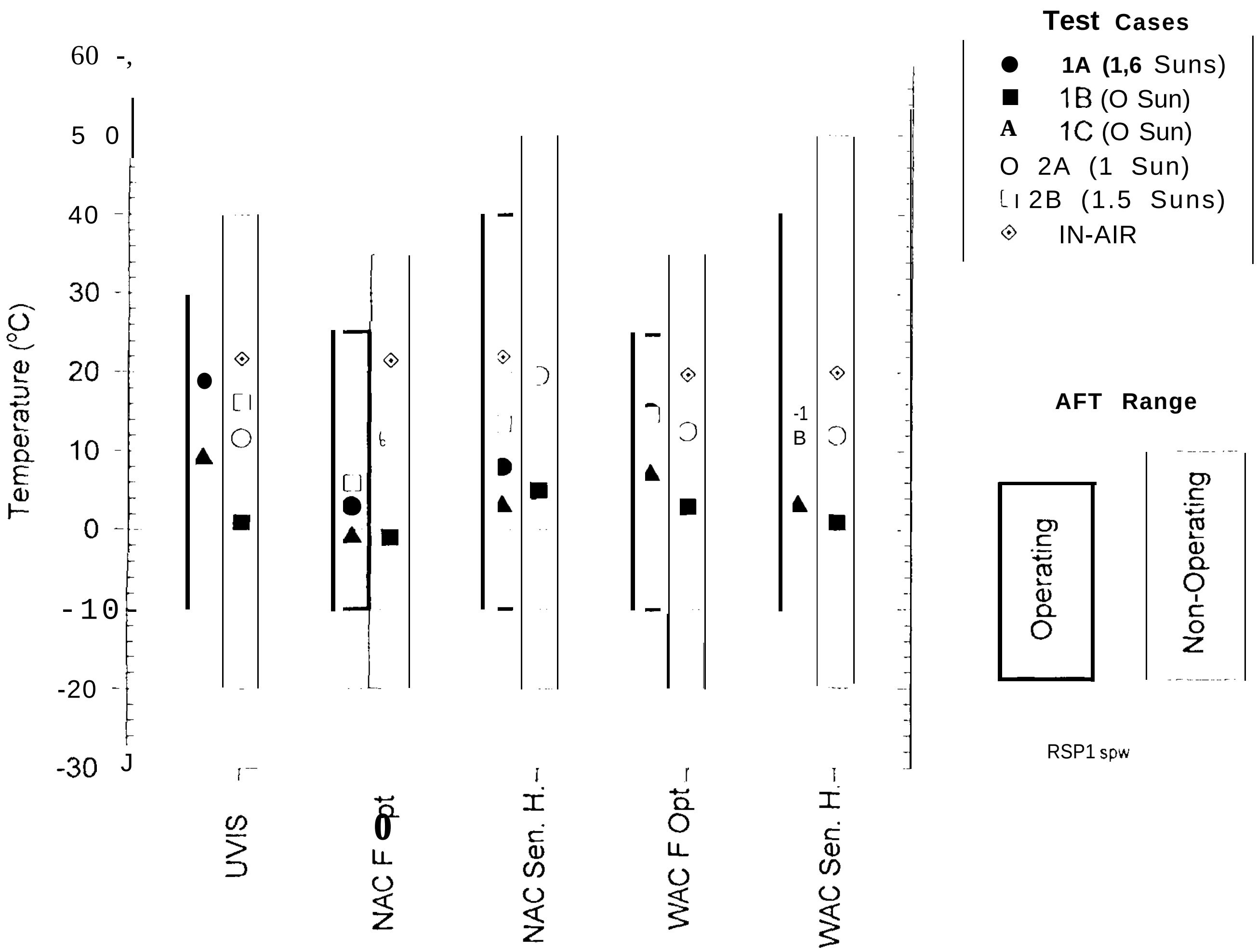


Figure 9: RSP test results (part 1 of 2)

hot sleep, and cold non-operation testing was performed. Probe maintenance and cold target operation were performed after steady-state was declared for hot sleep (corresponded to a hot PMCB) and hot operation, respectively. 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, Hot non-operation and cold sleep testing was expected. A transient Probe relay test was expected after cold the sleep steady-state. The use of a black Kapton "sock" on the deployed magnetometer boom thermal design was also tested. For last phase, an *ad hoc* one, 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.

Figures 7 and 8 show the actual test timelines for phases one and two.

TEST RESULTS - The discussion focuses on the science instruments and there is no mention of engineering subsystems. The engineering subsystems are addressed elsewhere [41. The phase summaries present the important results and the discussion of the ramifications of those results are presented afterward. The RSP, FPP, and appendage science test results are summarized in figures 9 through 12.

Phase One Summary - In phase one, the instrument thermal designs maintained temperatures within operating AFT limits for the hot sleep case with the exception of the CIRS OA which was in a decontamination state. The MIMI-LEMMS flight temperature transducer measurements were erratic during this phase. The instrument was rotated to clear the slip ring contact, and readings became more credible. After the hot sleep, the transient Probe maintenance was demonstrated to be thermally acceptable. Most of the instrument temperatures were well within non-operating allowable flight temperature limits during the cold non-operating case. However, the CDA main and HRD electronics violated non-operating AFT limits by approximately 5°C. In addition,

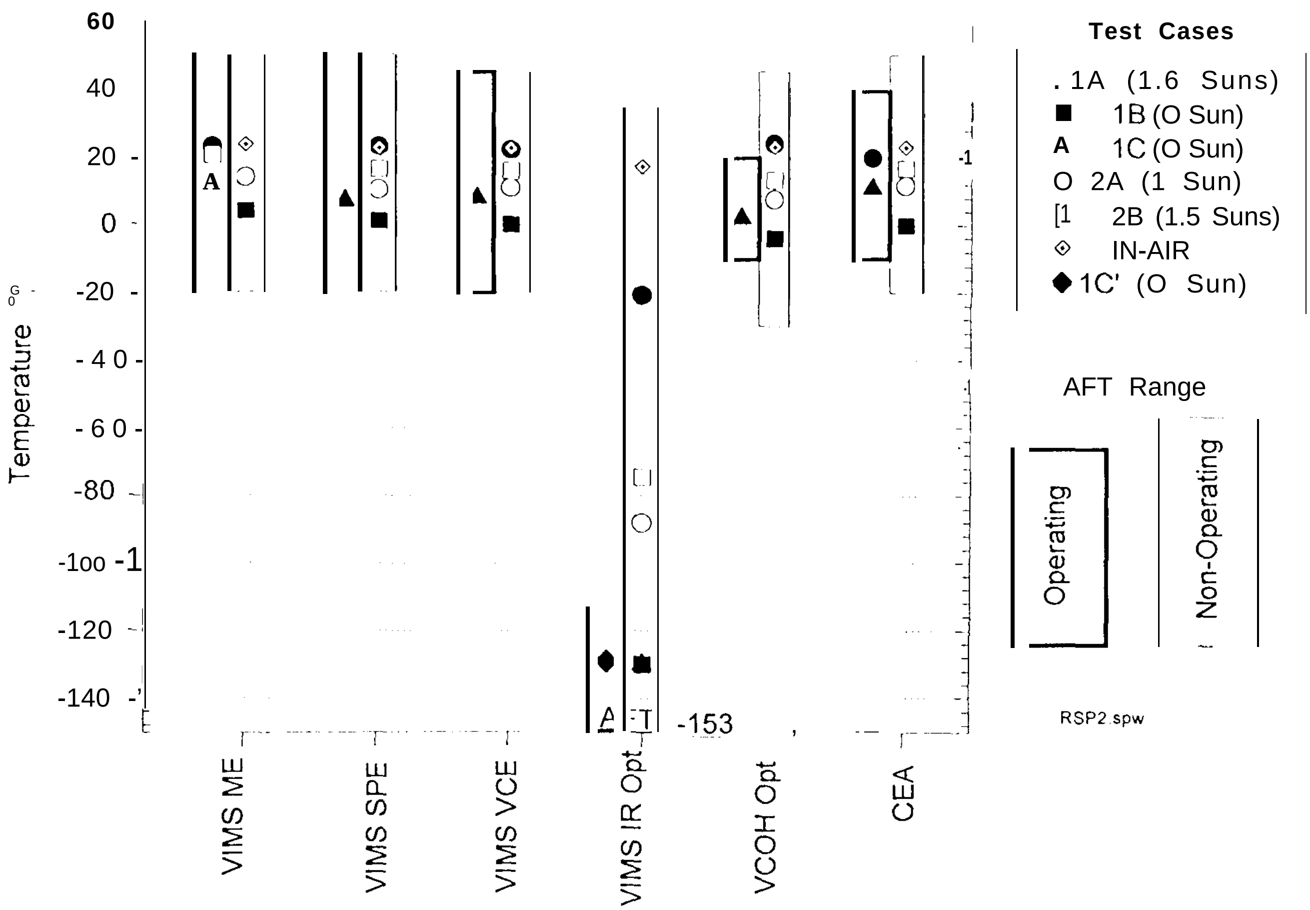


Figure 10: RSP test results (part 2 of 2)

confirmation was obtained that the RPWS MSC required three RHU's. S/C telemetry indicated that the PSA temperature was below the minimum non-operating AFT limit of 5°C. However, a test TC adjacent to the flight transducer showed a 7°C margin above the cited limit. Upon a post-test investigation, it was believed that there was electrical interference from the operating PSA's, which caused the erroneous flight transducer readings. During the hot operation case, operational AFT limit violations for the same CDA hardware persisted from the previous cold non-operating case since the operating and non-operating AFT limits were identical. The CIRS OA and VIMS-IR focal planes were exempted since the cold target was not yet operational. After steady-state was declared for this case, the cold target was flooded with liquid helium. Although the cold target achieved its required operating temperature of less than -243°C, the CIRS OA and VIMS-IR focal planes did not attain their respective operating temperature limits, -1 98°C and -1 95°C, respectively, Post-test analysis indicated that since the CIRS OA and VI MS-IR focal plane radiators did not have a 100% view of the cold target, parasitic heat loads from the S/C and support hardware were sufficient to prevent the instrument focal planes from attaining their operating temperatures.

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 the CIRS OA, UVIS, and VIMS-IR apertures to protect them from any further possible molecular contamination. However, the introduction of theses covers resulted in a non-flight configuration for the phase two test cases. The cold sleep case was eliminated since the results from hot sleep were as expected from analysis and developmental testing data. Consequently, risk in forgoing this design case was minor because the analytical models could be credibly used to verify temperature requirement compliance. From the project viewpoint, the deletion of the cold sleep case was advantageous since the protoflight HGA would not be cycled through the extreme cold temperatures again. The remaining pertinent cases involved: the deployed magnetometer boom design verification, the sizing of the CAPS supplemental heater power, and re-testing the FPP hot operation. The magnetometer boom deployment is expected just before the Earth flyby, and

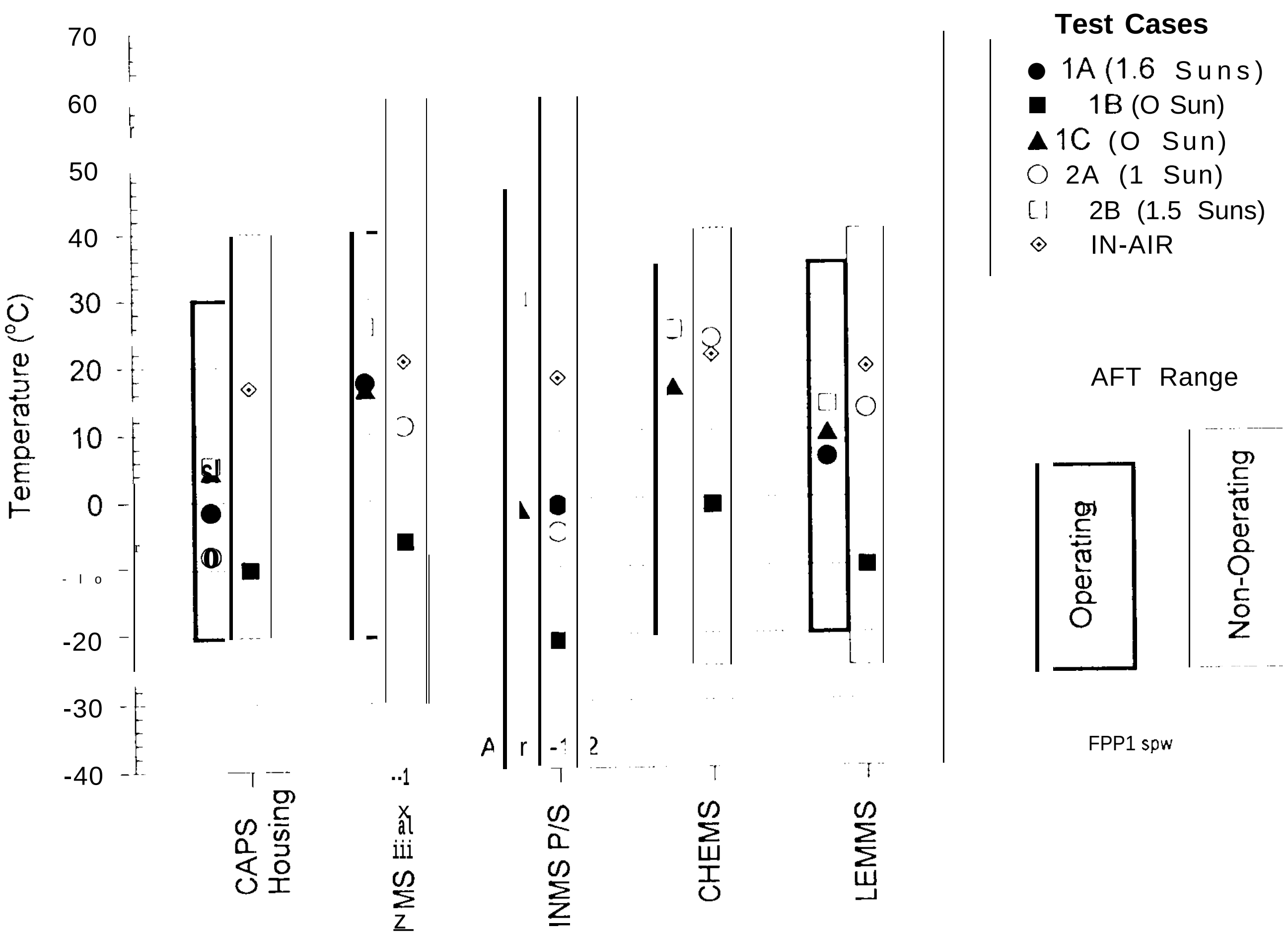


Figure 11: FPP test results

the solar irradiance is 1.0 sun. Concurrently, a case for determining the nominal cold PMCB temperature at 1.0 au was under consideration. In turn, a cold PMCB condition would present the appropriate conditions for the CAPS supplemental heater power sizing. Hence, a test case for the deployed magnetometer boom and the CAPS supplemental heater was established. The INMS team indicated that the instrument was not placed into its maximum operating power in phase one. This instrument drives the FPP hot operation design so an FPP hot operation re-test was prescribed. An important case for the PMCB was to assess the effect of the attached Probe simulator and closed MEA cover for the hot condition (1.6 suns). This presented the best opportunity for the FPP re-test since the bus would be in a hot condition, and so another test case was identified. In order to reduce test schedule, the 1.0 sun case was conducted before the 1.6 sun case. In general, the expected latter case temperatures would be warmer than the former, and test acceleration techniques such as utilizing test heaters to drive hardware to their expected equilibrium state could be implemented. As a result of replanning phase two, 54 hours of testing were eliminated.

In phase two, the magnetometers demonstrated that they are within operating AFT limits even when the black Kapton "sock" is insolated by the sun. Preliminary post-test analysis of the magnetometer boom structure indicated that the boom should be below the maximum deployment AFT limit of 5°C. The CAPS supplemental heater power sizing was within pre-test predictions, and since an engineering model was used, this sizing was timely enough to be implemented on the flight unit. The FPP instruments demonstrated compliance with operating AFT limits during the hot operation re-test with sufficient temperature margins.

In-Air Summary - Since the in-air testing was primarily intended to characterize the PMCB, this environment was thermally benign for the instruments. Therefore, there were no maximum AFT limit violations.

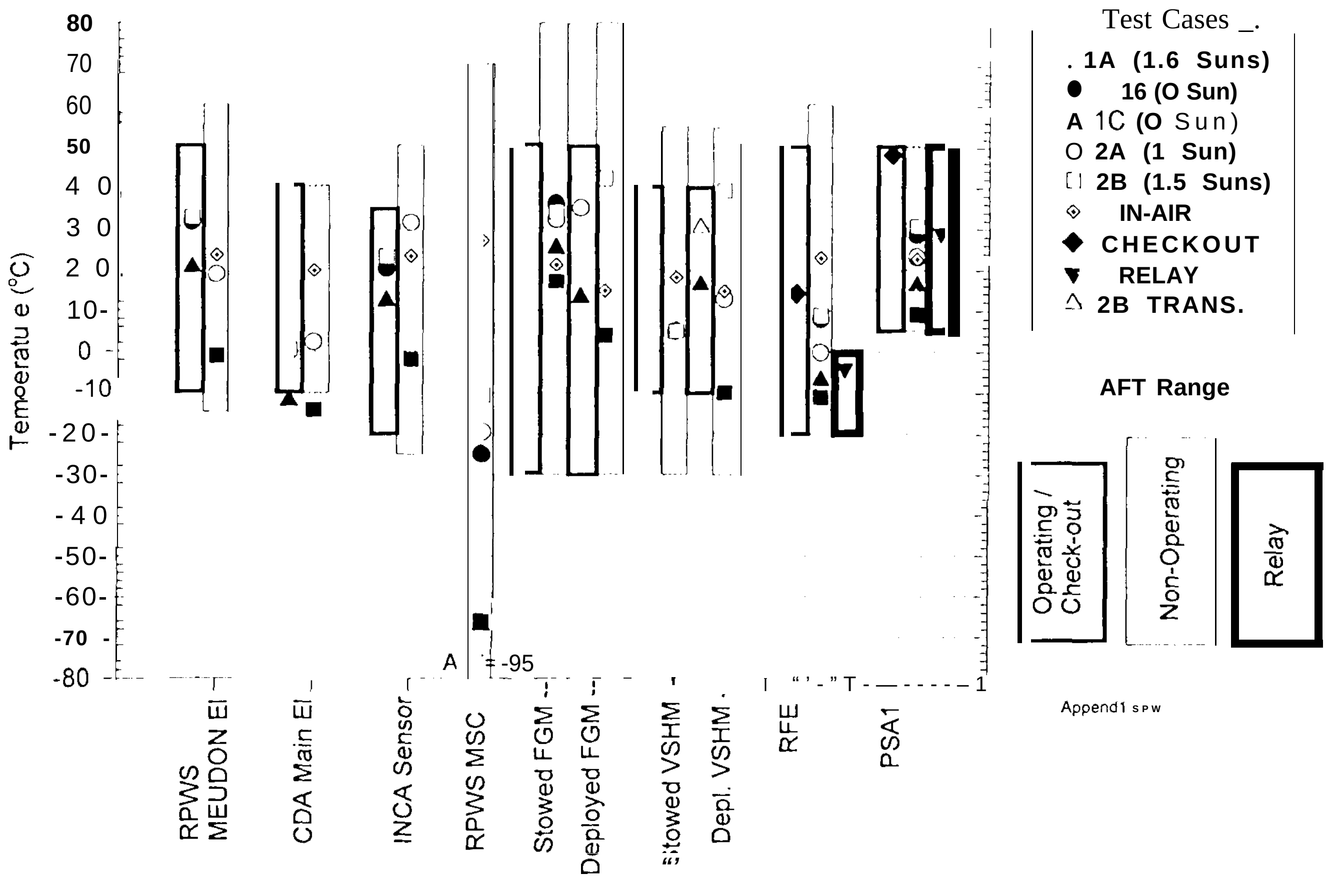


Figure 12: Main body-mounted science test results

DESIGN MODIFICATIONS - For those instruments that JPL had the bulk thermal design responsibility, there were no AFT limit violations. Hence, no design modifications are being considered. The CAPS supplemental heater was expected to be sized in this test since the CAPS power state for this heater usage was not precisely known.

The CDA instrument team had the entire bulk thermal design responsibility for its instrument. There were minimum operating and non-operating AFT limit violations for the main and HRD electronics. The team is in the process of reducing the minimum AFT limits based on test data to levels where there will not be any limit violations and will not compromise the qualification test margin.

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.

CONCLUSIONS

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, science instruments were largely unaffected since the HGA is conductively and somewhat radiatively isolated from the instruments.

Measurements from the MIMI-LEMMS and the PSA flight temperature transducers were not always in agreement with test TC'S and steps to characterize these discrepancies have been taken.

The S/C has been shipped to Cape Canaveral and preparation activities have commenced for an October 1997 launch.

ACKNOWLEDGEMENTS

A test of this magnitude could not have been conducted without the diligent efforts of many key personnel. Recognition is due to the cognizant thermal design engineers who conceptualized, analyzed, and insured implementation of their instrument designs: A. Avila, S. Clark, R. Garcia, R. Giampaoli, P. Hoffman, W. Ledeboer, E. Lin, J. Lyra, G. Kinsella, E. Kwack, K. Novak, and D. Porter. Additional thermal design test support was provided by E. Beltran, L. Chan, K. Juanero, S-C. Lee, P. Shakkottai, and B. Zeldin. All the thermal test engineers are deeply indebted to the responsive test facility support from K. Johnson, B. Lane, and S. Young and the excellent data acquisition support from T. Fisher, A. Levi, D. Perry, E. Bailey, and W. Villapando. In addition, recognition is due to P. Hardy and his team for mechanical integration support. Science instrument interface support was ably provided by K. Breitenbach and M. Carney. Thermal blanketing services were ably provided by J. Stultz, P. Hoffman, and M. Duran. The test was efficiently conducted by H. Doupe and J. Webster. D. Hanks was responsible for the test procedure. W. Moore and A. Accord managed the entire assembly and test operations. Finally, R. Reeve and J. Stultz managed the majority of the thermal design development, and it is to their credit that the test was successful.

The work described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

REFERENCES

[1] Reeve, R. "Thermal Redesign of the Galileo Spacecraft for a VEEGA trajectory," *Journal of Spacecraft and Rockets,* Volume 28, Number 2, Pp. 130-138, American Institute of Aeronautics and Astronautics, Washington, D. C., March-April 1991.

[2] Tsuyuki, G. "Functional Requirements, Cassini Orbiter, Design Criteria for Temperature Control," JPL Internal Document 699-205-3-210, Revision C, September 1996.

[3] Tsuyuki, G. and Stultz, J. "A Summary of the Cassini Thermal Development Test Program," SAE Paper 961576, July 1996.

[4] Mireles, V. and Tsuyuki, G. "A Summary of the Cassini System-Level Thermal Balance Test: Engineering Subsystems," SAE Paper 97 ES-278, July 1997.