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A Summary of the Cassini Spacecraft Thermal Performance from Launch Through Early Cruise

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ABSTRACT

Cassini, NASA's mission to investigate the Saturnian system was launched successfully on October 15, 1997. The cruise period from launch until Saturn arrival takes the spacecraft through a wide range of solar/thermal environments (0.67 astronomical units [AU] to 10 AU). The thermal control approach, which consists of thermal design features and operational constraints, must therefore maintain hardware temperature limits throughout this wide range of environments.

The off-sun exposure flight experience with interplanetary spacecraft at relatively close heliocentric distance is very limited. Cassini's ability to perform off-sun maneuvers relies heavily on the large thermal capacitance of the spacecraft's central body and the relatively short off-sun durations required for these maneuvers. The post launch execution of the first trajectory correction maneuver (TCM-1) was the first opportunity to validate the spacecraft off-sun capability and to enhance the thermal math model simulation capability.

INTRODUCTION

SCOPE - This purpose of this paper is to present the spacecraft thermal performance from launch through early cruise (10/15/97 through 2/24/98). This period is characterized by engineering activities, limited instrument maintenance and one TCM. The off-sun exposure flight experience with deep space interplanetary spacecraft at relatively close heliocentric distance is very limited. The spacecraft nominally points the High Gain Antenna (HGA) to the sun so that areas beneath the HGA are shaded while in the inner solar system (<5 AU). The Cassini mission design requires that the spacecraft be able to perform trajectory correction maneuvers with the HGA pointed away from the sun for limited durations. An integrated system level thermal balance test was performed prior to launch but off-sun attitude simulation was not feasible because of the size of the spacecraft and cost constraints. An off-sun solar characterization was performed in conjunction with TCM-1 when the nominally shaded spacecraft components were exposed to direct solar irradiance for a predetermined dwell period. A comparison of flight data with predictions will be presented. Special attention will be focused on the in-flight off-sun maneuvers since ground testing for these maneuvers was not performed. In addition, operational changes resulting from in-flight lessons learned will be discussed.

MISSION DESCRIPTION AND TRAJECTORY - The Cassini spacecraft was launched successfully on October 15, 1997. Since the energy of the Titan IV-B and Centaur launch vehicles was not sufficient for a direct injection trajectory, planetary gravity assists from Venus (twice), Earth, and Jupiter will enable the spacecraft to reach Saturn by July 2004 (see figure 1). The spacecraft heliocentric distance will vary from 1 AU at launch, to 0.67 AU at the first perihelia, to 10.07 AU at Saturn. During its cruise to Saturn, the three axis stabilized spacecraft will normally point its HGA towards the Sun. However, during TCM's the spacecraft is turned away from Sun point to accommodate delta V vectors that are not aligned with the solar pointing vector.

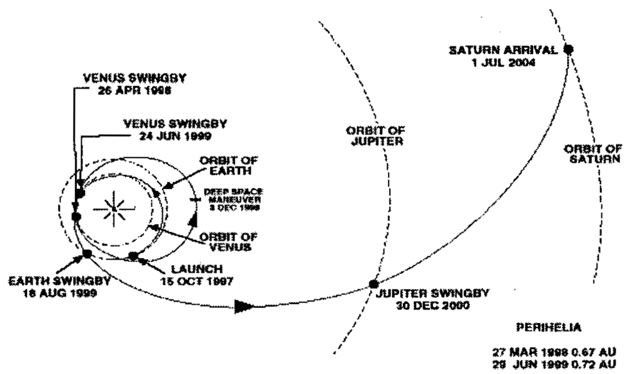

Figure 1: Cassini Mission Trajectory

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

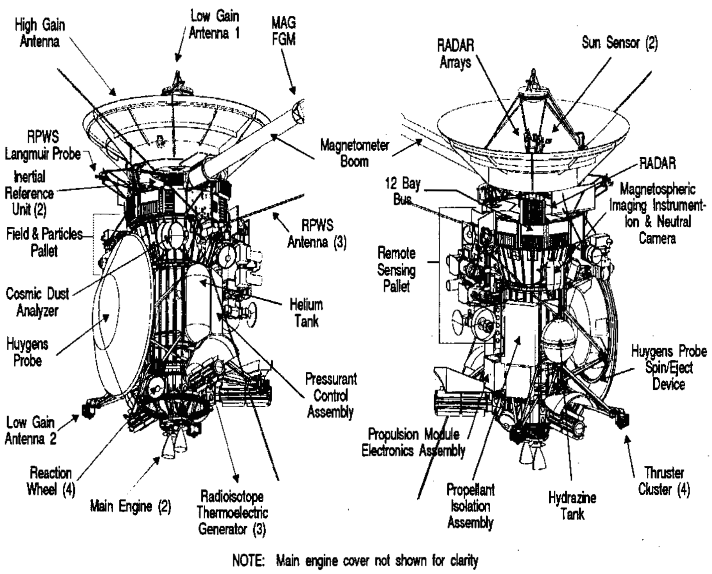

Figure 2: S/C configuration

The science instruments are mounted throughout the spacecraft, most notably on the Huygens Probe, the remote sensing pallet (RSP), and the fields and particles pallet (FPP). The Flux Gate Magnetometer (FGM) and the Vector/Scalar Helium Magnetometer (V/SHM) are located on a deployable boom which is mounted to the Bus. A pivoting Cosmic Dust Analyzer (CDA) and a radio and plasma wave (RPWS) instrument are attached to the USSA.

SPACECRAFT THERMAL CONTROL DESCRIPTION

THE CHALLENGE - The requirement to satisfy mission objectives at Saturn (10.07 AU) as well as receive inner solar system planetary gravity assists (0.67 AU) results in a large variation in heliocentric distance. In order to provide mission trajectory design flexibility (thus optimizing propellant consumption) the spacecraft must tolerate off-sun maneuvers throughout the heliocentric range. The spin stabilized Galileo spacecraft had a trajectory that was comparable to Cassini's but its ability to implement changes to its velocity vector while sun-pointed meant that it did not have to contend with solar exposure due to off-sun maneuvers at small heliocentric distances [1]. The three axes stabilized twin Voyager spacecraft did require off-sun maneuvers but none were required inside of 1.0 AU [2]. The formidable challenge for Cassini was met with thermal design features and operational constraints.

THERMAL DESIGN FEATURES - The highlights of the spacecraft thermal design are illustrated in figure 3. Details of the Multi-Layer Insulation (MLI) blankets are not shown for clarity. The thermal control implementation minimizes the sensitivity to the widely varying environments. The HGA serves as a shade and its structure serves to conductively isolate it from the Bus while the spacecraft is sun pointed. During maneuvers, the Huygens Probe is used as a shade which protects most of the Orbiter's most thermally sensitive hardware.

Electrical heater power requirements were minimized by the use of Radioisotope Heating Units (RHU's) and Variable Radioisotope Heating Units (VRHU's). In addition, the heat generated by the three Radioisotope Thermoelectric Generators (RTG's) was used to heat the PMS. The spacecraft's thermal design and implementation are have been previously documented [3 and 4].

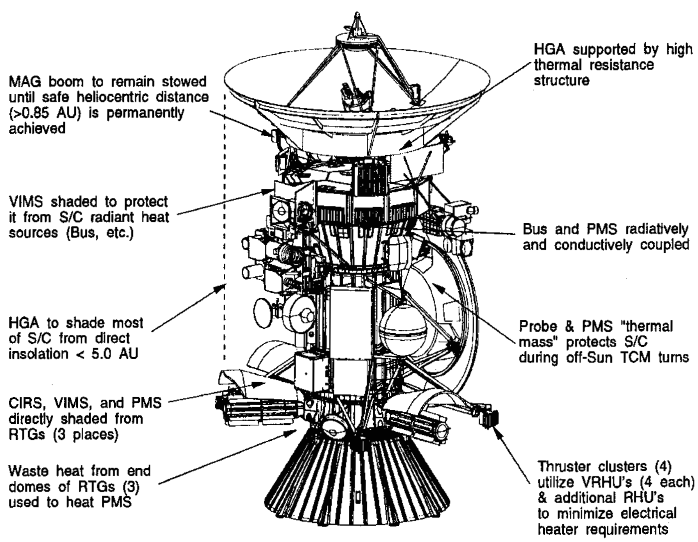

Figure 3: System-level thermal design schematic

OPERATIONAL CONSTRAINTS and REQUIREMENTS - The allowable flight temperature (AFT) limits are specified in project documentation [5]. In addition, the AFT documentation specifies how inner solar system off-sun maneuvers should be executed (see figure 4). All maneuvers inside of 5.0 AU are performed in the X-Z plane by turning the -Z axis toward the +X (-yaw turn) axis to always place the Probe side of the spacecraft in the Sun. The maximum duration limit of maneuvers inside of 1.0 AU is specified by:

$$\text{Duration} = Y_1 * (AU^2/0.61^2)$$

where Y_1 = 30 minutes

AU = desired maneuver heliocentric distance

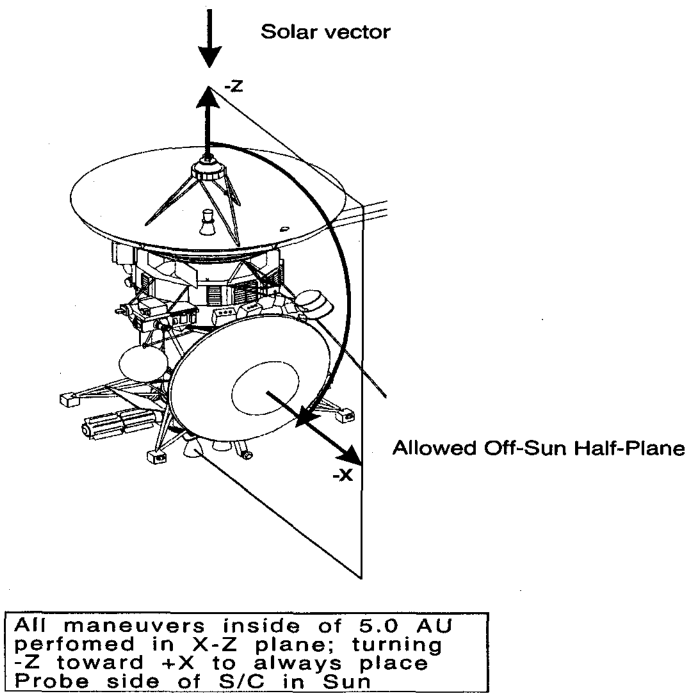

Figure 4: Maneuver Execution Requirement

The duration of maneuvers between 1.0 AU and 5.0 AU is specified by:

Duration = Y_2*AU^2

where Y_2 = 4 hours if off-sun angle is less than or equal to 60 degrees

= 1.35 hours of the off-sun angle is greater than 60 degrees

AU = desired maneuver heliocentric distance

There are no constraints beyond 5.0 AU. This maneuver strategy protects the most vulnerable assemblies from exposing radiators and apertures to direct solar irradiance. Additional operability constraints, known as flight rules, are also contained in project documentation [6]. These constraints apply to modes of instrument and engineering subsystem operations, heater usage and boresight pointing.

SPACECRAFT SYSTEM LEVEL FAULT PROTECTION - There are three system level fault protection (SFP) algorithms that contribute to the spacecraft's thermal control approach. The first is the Autonomous Thermal Control (ATC) algorithm. ATC is essentially a computer controlled thermostat. The algorithm enables the spacecraft's on-board computers to monitor the temperatures of specified assemblies (up to 12 assemblies) and compare them to on-board thresholds. ATC then responds by issuing a power on or power off command to the assemblies' heater(s) depending on which threshold has been exceeded. The second is the Emergency Overtemperature Algorithm (EOTA). EOTA was implemented to provide some protection against inadvertent off-sun solar exposure. The algorithm enables the spacecraft's on-board computers to monitor the temperatures of specified assemblies (up to 12 assemblies) and compare them to on-board thresholds. The specified assemblies are those that respond quickly to environmental heating and thus provide quick notification of anomalous spacecraft pointing. When EOTA monitors indicate temperatures are exceeding the specified thresholds, the algorithm requests a response from the third SFP algorithm. The third algorithm is called spacecraft SAFING. SAFING sends a request to the attitude control subsystem to point the spacecraft to the sun and also reconfigures the spacecraft to a thermally safe power profile.

CASSINI SPACECRAFT SYSTEM LEVEL THERMAL MATH MODEL (SCTMM)

SCTMM REQUIREMENTS - The SCTMM was developed as an operational tool for use in mission planning and anomaly reconstruction. The SCTMM was to provide ±5°C agreement for assemblies with relatively small AFT ranges (e.g. Bus Bays) and ±10°C for assemblies with relatively large AFT ranges (e.g. HGA areas and RTG's). The SCTMM simulates, as a function of time, environmental heating, electrical power dissipation, and RTG and RHU power and thermal decay [7].

SCTMM DEVELOPMENT APPROACH - The SCTMM was developed by reducing and integrating existing subsystem thermal design models. The SCTMM consists of all relevant spacecraft hardware with Space as the only boundary condition. One of the objectives of the Cassini Spacecraft Thermal Balance test was to correlate the SCTMM with spacecraft performance [3 and 4].

SCTMM POST LAUNCH CORRELATION & OFF-SUN CHARACTERIZATION - The first four weeks after launch, when the trajectory remained at about 1 AU, provided a significant amount of data while at the sun pointed attitude. The design of the first trajectory correction maneuver included a spacecraft dwell at the delta V vector attitude for the maximum duration allowed by the design requirements rather than what was needed to achieve the delta V. These data were then used to improve the SCTMM, validate compliance with flight functional requirements and generated thermal performance predictions for perihelion conditions.

FLIGHT DATA AND PREDICTION COMPARISON

SUN POINTED (HGA TO SUN) ORIENTATION- When the spacecraft is in the HGA to Sun orientation, most assemblies are shaded by the HGA. There are 194 temperature transducers on the spacecraft that are monitored but this paper will focus only on those assemblies that are continuously exposed to solar irradiance (HGA and HGA mounted assemblies) while sun pointed and those that start out in the HGA shade and then become illuminated when maneuvers are executed. A summary of sun pointed flight transducer and SCTMM temperatures for the heliocentric distances where TCM-1 was performed and at the present date (2/24/98), is shown in Table 1. The SCTMM data shown for the 1.01 AU case (TCM-1) was generated after model improvements were made following the off-sun thermal characterization. The improved SCTMM was then used to generate predictions for the 0.73 AU case (2/24/98). It should also be noted that the SCTMM now meets the documented simulation accuracy requirements [7].

Table 1: Sun Pointed Temperature Comparison

Assembly	AFT's, °C	1.01 AU S/C, °C	SCTMM °C	0.73 AU S/C, °C	SCTMM °C
HGA Reflector	-199/125	-45	-45	10	10
HGA X-Bd FSS	-208/129	15	15	73	72
LGA-1	-206/81	-2	-4	56	55
Sun Sensor 1	-90/80	-12	-12	28	26
Sun Sensor 2	-90/80	-10	-12	26	24
MAG FGM	-30/80	26	30	31	32
MAG VSHM	-30/55	8	6	8	6
IRU A	-5/45	29	31	31	31
IRU B	-20/45	16	13	18	14
BAY 5	5/50	25	26	26	27
BAY 6	5/50	21	24	24	25
BAY 7	5/50	18	20	21	21
BAY 8	5/50	24	24	25	25
BAY 9	5/50	22	19	24	21
FPP Structure	N/A	7	8	10	9
INMS Electronics	-30/60	3	5	5	6
INMS Sampling Area	-102/60	-4	-4	-2	-4
CAPS DPU	-20/40	-1	-1	-3	0
CAPS IMSCVR	-20/40	-6	-6	-10	-6
MIMI CHEMS	-25/40	13	12	15	12
MIMI LEMMS ROT	-25/40	-3	-3	0	-2
MIMI LEMMS NROT	N/A	3	2	5	2
CDA EMB	-50/40	2	0	-8	-9
CDA HRD	-30/40	-7	-6	-18	-19
CDA NROT	-20/40	10	11	7	7
RPWS Antenna Assy	-15/60	16	21	17	22
PROBE RFE	-20/60	-6	-4	-2	0
PROBE PCDU	-40/70	11	10	14	13
PROBE Spin Eject Device	-80/70	-54	-59	-50	-58
Thruster Clusters 1	20/60	40	40	41	44
Thruster Clusters 2	20/60	39	39	41	44
Thruster Clusters 3	20/60	40	40	42	44
Thruster Clusters 4	20/60	39	40	40	44
REA-A Oxidizer Valve	5/45	33	33	33	35
REA-A Fuel Valve	0/45	17	19	18	21
REA-A Chamber	-1/39	5	5	7	7
REA-B Oxidizer Valve	5/100	34	32	36	33
REA-B Fuel Valve	-10/100	17	18	18	20
REA-B Chamber	N/A	5	4	5	5
LGA-2	-80/140	3	3	5	3
RTG 1 (avg. of 3 sensors)	NA/260	248	240	247	247
RTG 2 (avg. of 2 sensors)	NA/260	243	245	243	245
RTG 3 (avg. of 3 sensors)	NA/260	247	240	247	240

OFF-SUN (PROBE TO SUN) ORIENTATION - The spacecraft off-sun maneuver thermal response can be grouped into four classes.

1. Surfaces that are continuously exposed to solar irradiance and respond by cooling as the -Z axes turns away from the sun. In some cases the cooling is followed by warming if the off-sun angle is large enough that now the +Z surfaces of HGA mounted assemblies are exposed. Examples include the HGA, Low Gain Antenna 1 (LGA-1) and the Sun Sensors Heads 1 and 2 (SSH1 and SSH2).

2. Surfaces that are nominally shaded by the HGA, on the -X axis hemisphere, and respond by warming as the -X axes turns to the sun. Examples include the Bus Bays 5 through 9 and the Fields and Particles Pallet.

3. Surfaces that are nominally shaded by the HGA, on the spacecraft aft end (+Z direction), and respond to a combination of warming influences as the -X axes turns to the sun, the power profile changes, and thruster or main engine burn occurs. Examples include thruster clusters, the main engine oxidizer and fuel valves and the combustion chamber.

4. Surfaces that remain shaded while sun-pointed and remain shaded during maneuver execution (+X axes) respond only to changes in power profile. Examples include all the assemblies mounted to the RSP. This class of response is not the focus of this paper and will not be presented or discussed.

The total off-sun duration allowed by the thermal design requirements for TCM-1 was 1 hour 22 minutes and 38 seconds at an off-sun angle of 70.6 degrees while at a heliocentric distance of 1.01 A.U. The duration includes the time it takes to turn to and from off-sun attitude (yaw turns). The spacecraft thermal performance during TCM-1 is captured in Figures 5 through 28 (for representative sunlit surfaces). The off-sun flight data clearly indicate that the maneuver approach was sound and no thermal limits were threatened. Post maneuver correlation of the SCTMM yielded acceptable equilibrium and transient agreement. The SCTMM calculated temperature profiles are also shown in Figures 5 through 28.

The class 1 response can be seen in Figures 5 through 9. Turning off-sun cooled the HGA and LGA-1 assembly given that the nominally sun pointed HGA is now viewing deep space and solar exposure is now edge on to the dish. The SSH1 (whose aperture is in the -Z direction and radiator is in the -Z direction) on the -X side first cools as the spacecraft begins the turn off-sun and then warms as the its radiator, cabling, and MLI wrap are exposed to the sun at this off-sun angle. The SSH2 on the +X side shows only the effect of cooling, since its radiator is shaded by the spacecraft during maneuver execution.

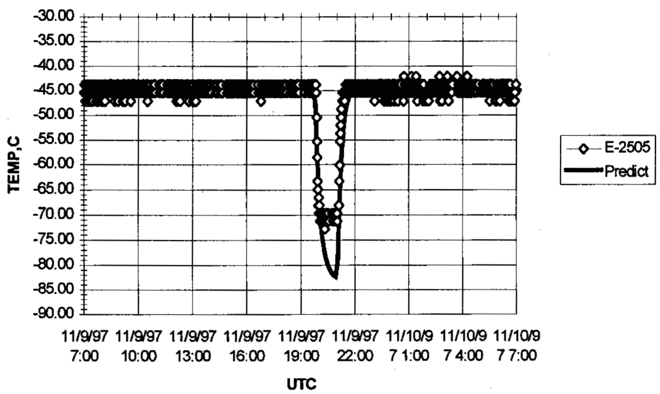

Figure 5: HGA Reflector Rear Surface

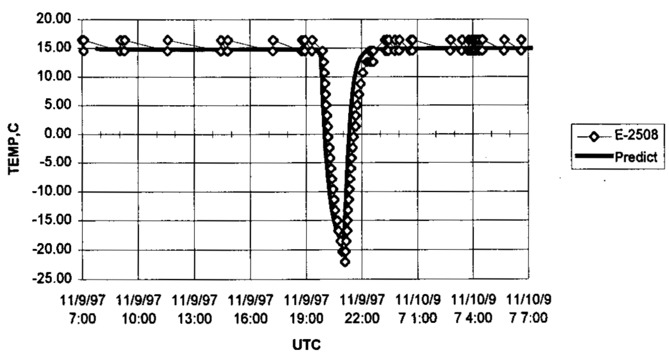

Figure 6: HGA X-Band FSS

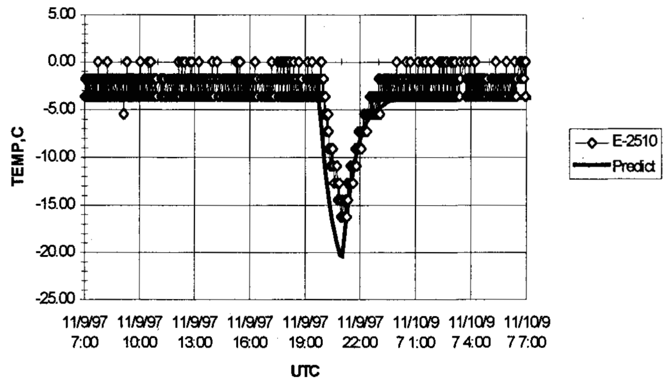

Figure 7: LGA Gain Antenna 1

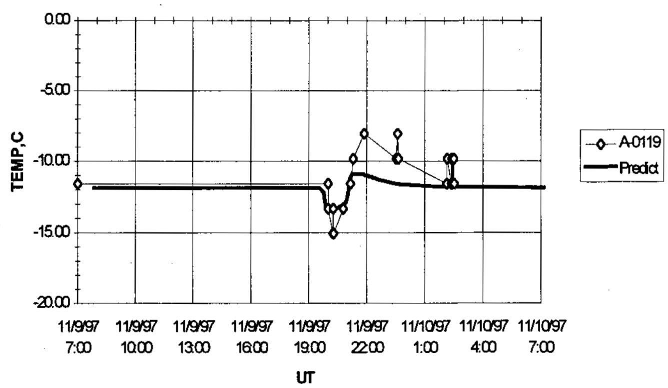

Figure 8: Sun Sensor Head 1

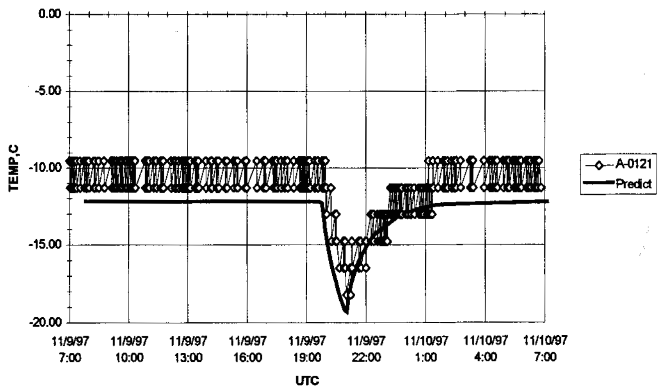

Figure 9: Sun Sensor Head 2

The class 2 response to solar heating can be seen in Figures 10 through 23. All of these assemblies are in the -X hemisphere which becomes exposed to the sun during the maneuver. The Inertial Reference Unit B (IRU B, Figure 10), above Bay 5, Bays 5 through 9 (Figures 11-15), Probe Radio Front End (RFE, Figure 16), above Bay 6, all show a clear response to solar heating. The response of the fields and particles pallet and the FPP mounted instruments can be seen in figures 17 through 21. The CDA is mounted to the USSA but is also in the -X hemisphere. Its response is shown in Figure 22. The LGA-2 response is shown in figure 23.

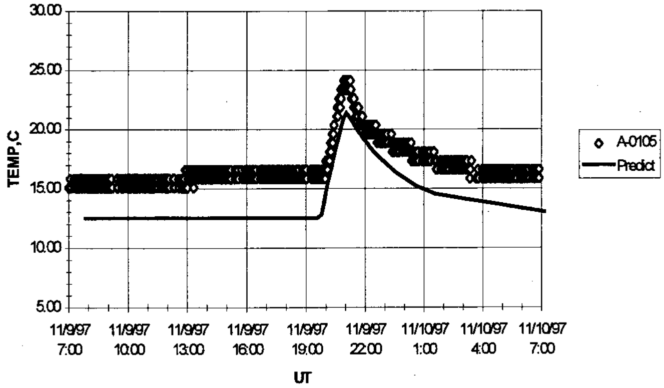

Figure 10: Inertial Reference Unit B

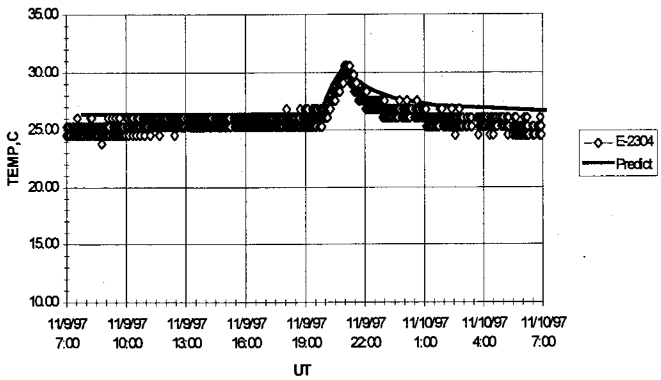

Figure 11: BAY 5

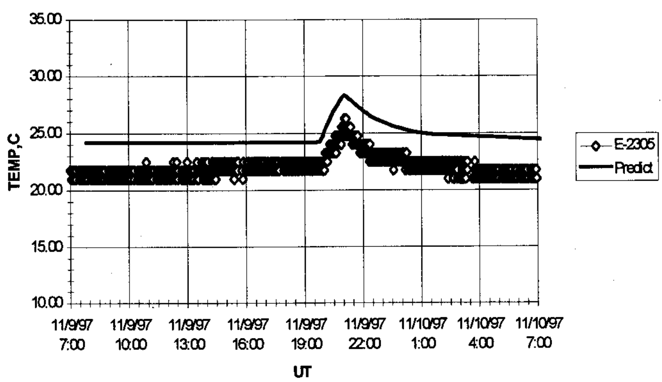

Figure 12: BAY 6

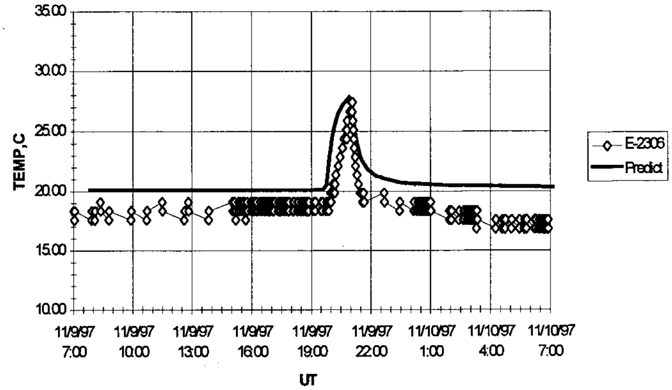

Figure 13: BAY 7

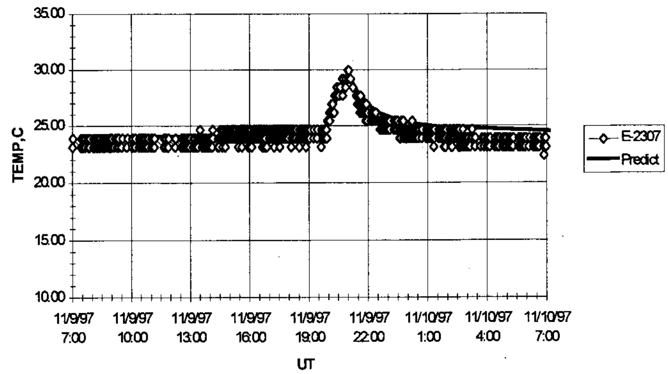

Figure 14: BAY 8

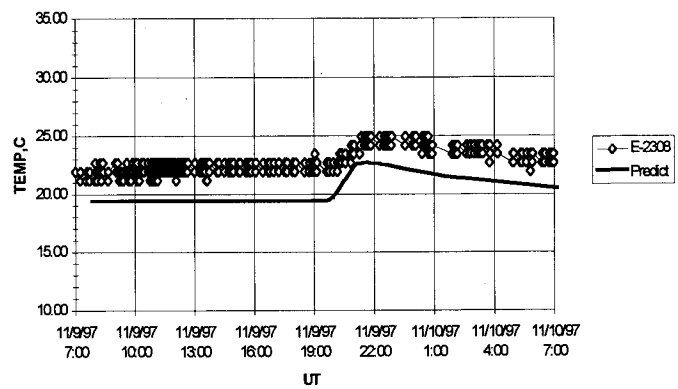

Figure 15: Bay 9

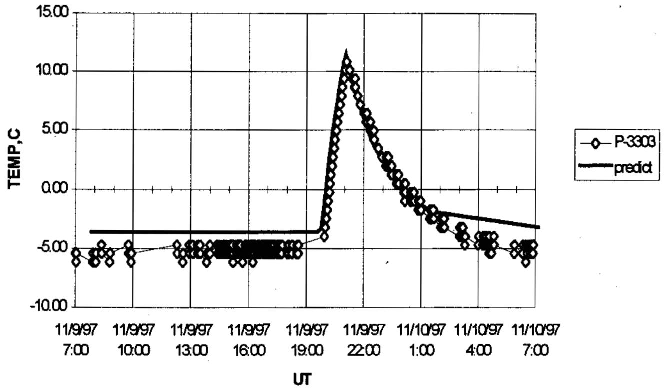

Figure 16: Probe RFE

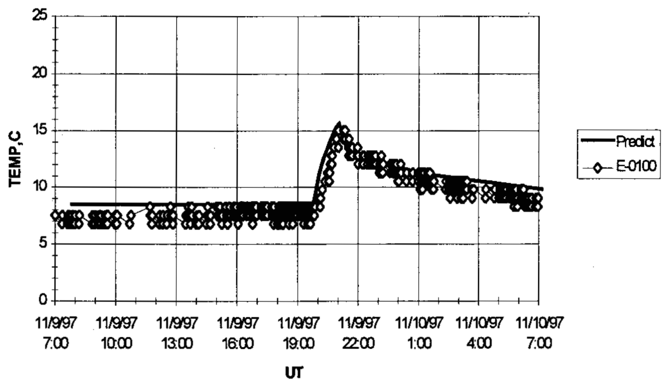

Figure 17: Fields & Particles Pallet Structure

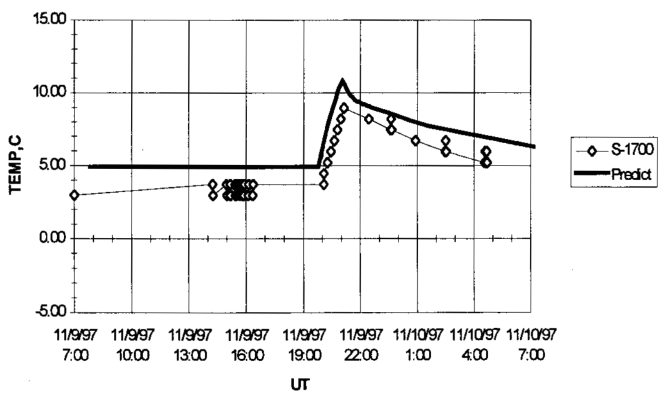

Figure 18: Ion & Neutral Mass Spectrometer

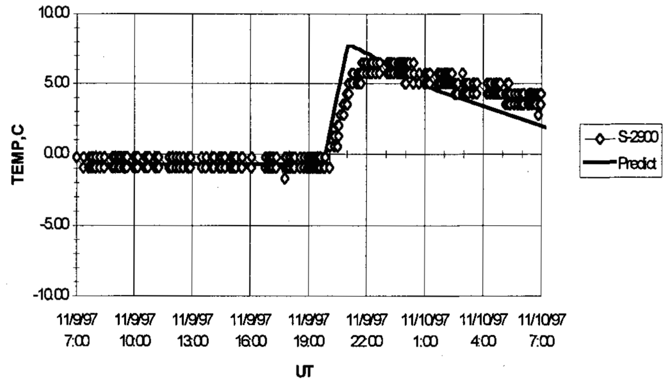

Figure 19: Cassini Plasma Spectrometer

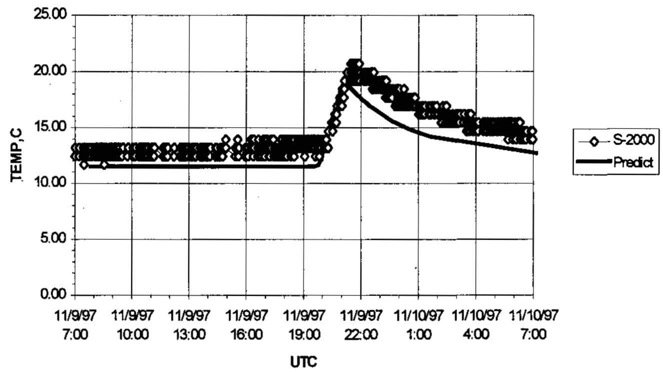

Figure 20: MIMI CHEMS

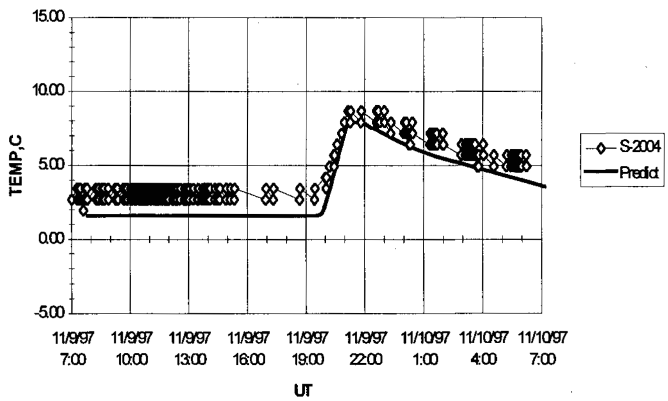

Figure 21: MIMI LEMMS

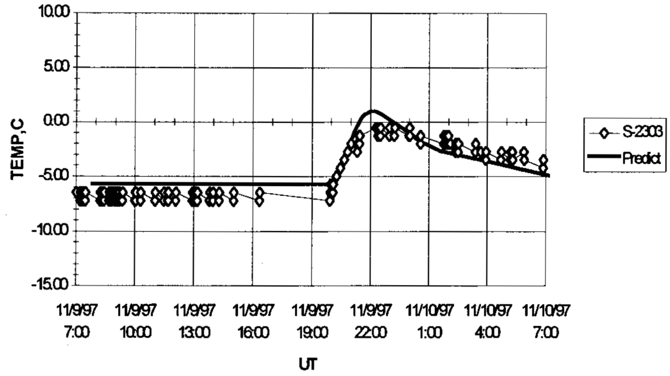

Figure 22: Cosmic Dust Analyzer

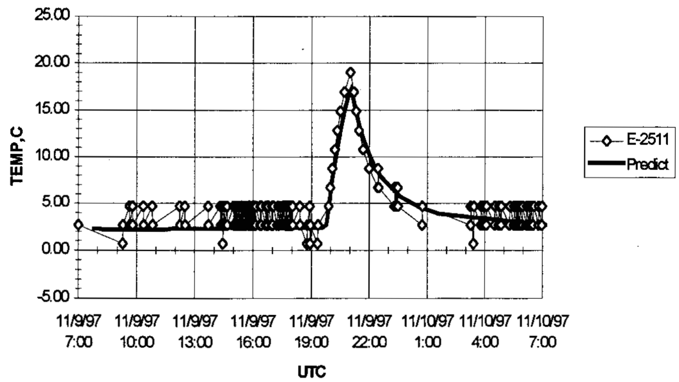

Figure 23: Low Gain Antenna 2

The class 3 response can be seen in Figures 24 through 28. Figures 24 and 25 display the temperature trends on the Reaction Control Subsystem (RCS) Thruster Cluster Housings. The trends displayed by Cluster Housings 2 and 3 (in the -X hemisphere) show a somewhat singular temperature spike which includes the thermal response to both turn related burns and solar heating during the off-sun period. The Thruster Cluster Housings 1 and 4 (in the +X hemisphere) were shaded during the TCM-1 maneuver attitude and thus are not included here. Figures 26 through 28 display main engine A temperature trends. All three show a warming of the engine assembly due to the influence of the Rocket Engine Assembly A (REA-A) engine mounting plate heater that conditions the main engine prior to the burn. Subsequent transient spikes in temperature are seen as a result of the main engine burn, being most pronounced for the REA-A Chamber. At the 70.6 degree off-sun angle, the main engine assemblies were shaded by the stowed main engine cover. Solar exposure to the main engine assemblies can be expected for off-sun angles greater than 75 degrees.

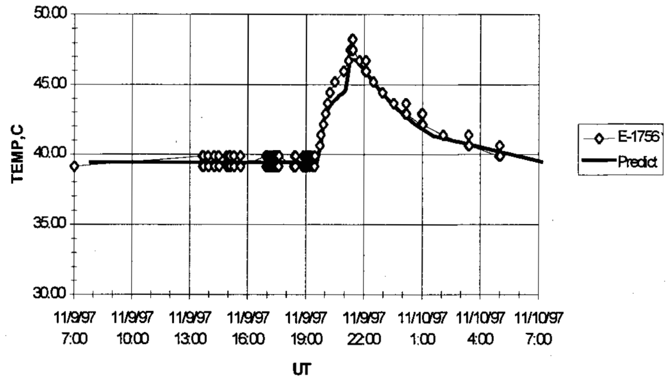

Figure 24: Thruster Cluster 2

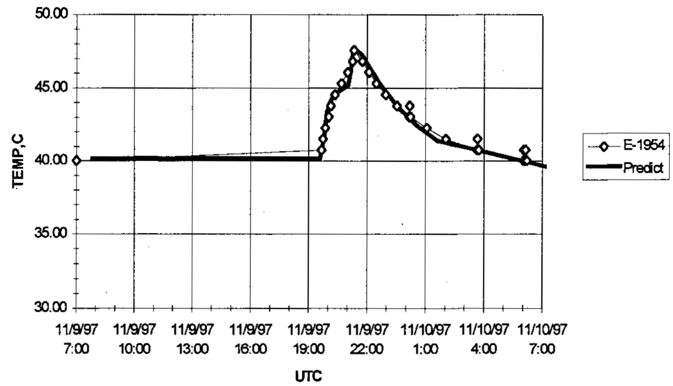

Figure 25: Thruster Cluster 2

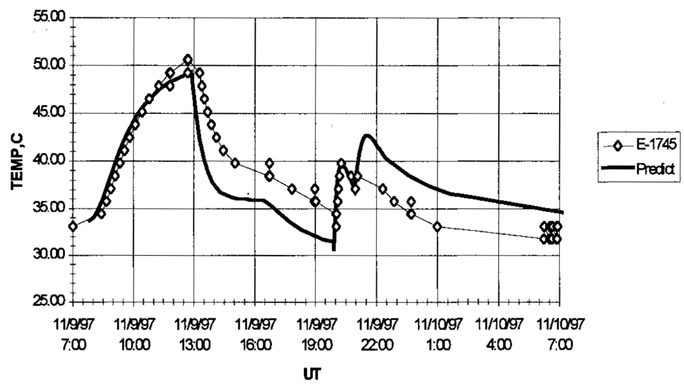

Figure 26: REA-A Oxidizer Valve

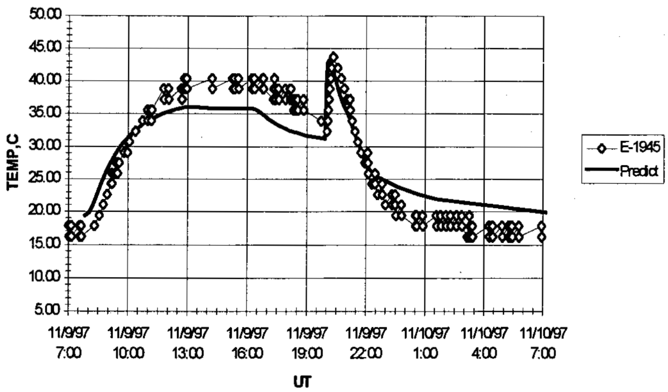

Figure 27: REA-A Fuel Valve

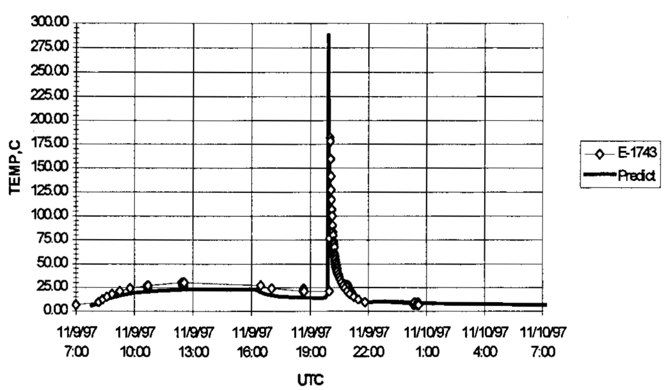

Figure 28: REA-A Combustion Chamber

The Probe and PMCB are very massive and, as expected, their transient responses during off-sun maneuvers are very small, if not completely negligible. This fact is, of course, the driving reason for implementing the maneuver execution approach that allows solar heating on the -X hemisphere of the spacecraft.

THERMAL PERFORMANCE DEVIATIONS AND LESSONS LEARNED - The only performance deviation that has occurred is the magnitude of the heating of the main engine A oxidizer valve, which occurred while the spacecraft was sun-pointed. The temperature of the main engine A oxidizer valve exceeded it's high "At Ignition" temperature limit of 45°C (Figure 26). While not a requirement violation at the time, the temperature had to be reduced to meet "At Ignition" requirement before the main engine could be fired. It peaked at 51°C before a real-time command was sent to turn off the primary oxidizer valve heaters. This action corrected the problem and the TCM-1 was completed without further incidents. The deviation was due to powering on the main engine mounting plate heater while the oxidizer valve heaters were on. A review of Solar Thermal Vacuum Test (STV) data reveals information supporting these events as a nominal response [8]. The scenario tested had the Main Engine Assembly (MEA) hardware at equilibrium with the MEA Cover open and the PMCB in a worst case cold cruise condition. The test case was meant to conservatively verify that the engine mounting plate heater could elevate the initially cold chamber temperature to its "at ignition" range. This was verified. However, the additional power from the engine mounting plate heater did not adversely affect any other main engine hardware because the PMCB was deliberately in its worst case cold cruise condition with temperatures at the lower end of their requirement range. The STV data indicate that the engine mounting plate heater elevates the oxidizer and fuel valve temperatures approximately 25°C above their initial temperature. This effect in flight was unanticipated but should have been expected based on this data.

The strategy with respect to the use of main engine oxidizer valve heaters and engine mounting plate heaters has been updated and will be implemented for the next main engine TCM. Associated with this new strategy, Flight Rules that govern the operation of these heaters have been updated.

CONCLUSION

The off-sun and sun pointed thermal performance of spacecraft subsystems to date has been exceptional with respect to both expectations and requirement compliance. The maneuver approach has been validated and comfortable margins are predicted for perihelion conditions. The only performance deviation (temperature of the main engine A oxidizer valve for TCM-1) that has surfaced is of an operational nature and was subsequently verified as expected response when solar thermal vacuum test data was revisited. A new heater strategy has been implemented that maintains the main engine oxidizer valve temperatures within the "At Ignition" requirements. This strategy required a change in the use of the main engine oxidizer valve heaters and engine mounting plate heaters. Associated with this new strategy, Flight Rules that govern the operation of these heaters have been updated. Flight performance to date has also validated the use of the SCTMM and has provided data for enhancing its prediction capability.

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