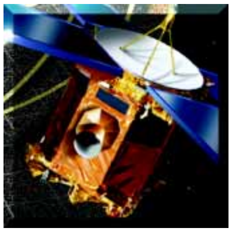

Thermal Design of NEAR

Carl J. Ercol and Stephen J. Krein

Because of its externally mounted sensors, tight power budget, and widely varying mission conditions, the Near Earth Asteroid Rendezvous (NEAR) spacecraft presented a challenging thermal design opportunity. This article describes the design approach, problems encountered in the design process, and final thermal design. Spacecraft testing and early mission performance are also discussed.
(Keywords: Heater, Multilayer insulation, Radiator, Thermal, Thermostat.)

INTRODUCTION

The thermal design challenge posed by the Near Earth Asteroid Rendezvous (NEAR) spacecraft stemmed from its externally mounted sensors and widely varying mission environments. The mission thermal environment varies as a function of both spacecraft attitude and solar distance. During the extended cruise mode, the solar distance varies considerably, inducing sizable fluctuations in the solar heating inputs to the spacecraft. In addition, the spacecraft attitude can deviate from the default solar-normal orientation provided that the attitude is maintained within the boundaries required to enable communication with the Earth-based ground station. Further compounding the design complexity is the wide variation in internal spacecraft power dissipations expected throughout the mission. The internal power dissipation is directly linked to the overall system objectives at any given mission phase, thus the thermal load can vary from minimal housekeeping power during the semi-dormant cruise mode to maximum power during science data acquisition at the asteroid. This article describes the design approach, the resulting thermal design, and the thermal model and associated analyses including prelaunch, ascent and transfer, cruise mode, and asteroid mode mission phases for the NEAR spacecraft. The results of thermal vacuum testing and post-test model correlations efforts are included, as are early mission temperature telemetry data.

SPACECRAFT DESCRIPTION

The spacecraft in the deployed flight configuration is shown in Fig. 1. To minimize the cost and schedule penalties inherent in configuration complexity, the design is relatively simple. The spacecraft structure is composed of forward and aft aluminum honeycomb decks connected together with eight aluminum honeycomb side panels. Mounted on the outside of the forward deck is a 1.5-m X-band dish antenna and four fixed solar panels. The Magnetometer mounts on top of the high-gain antenna feed and will conduct the first close-up search for coherent bodywide magnetism at an asteroid. Incorporated in the design is a complete calibration of the Magnetometer as well as the effect on

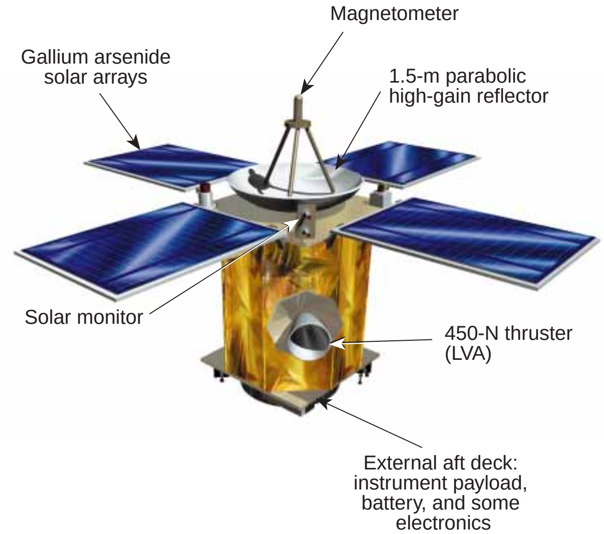

Figure 1. NEAR spacecraft. (LVA = large velocity adjustment.)

the instrument of spacecraft magnetic interference. The remainder of the NEAR instruments, including the Near-Infrared Spectrometer, the Multispectral Imager, the X-Ray/Gamma-Ray Spectrometer, and the Laser Rangefinder are mounted on the outside of the aft and forward decks. The instruments are all fixed relative to the spacecraft and point in a common direction.[1]

In general, the science payload requires a stable thermal environment for the instrument sensors and support electronics to perform at their optimum capacity. The degree of required temperature stability and allowable temperature variation differs among the instruments. Because the mounting interfaces (spacecraft decks) experience potentially wide temperature variations throughout the mission, the majority of the instruments must be thermally decoupled from the decks. With the exception of the X-ray/Gamma-Ray Spectrometer and the argon gas–filled solar monitor, all of the instruments incorporate thermal isolation provisions in their mounting schemes.

Mounted on the inside of the forward deck and the inside and outside surfaces of the aft deck are the spacecraft bus electronics and instrument data processing electronics. With the exception of pre-existing vendor-supplied packages, all electronics boxes were designed to minimize mass. Detailed electronics modeling to specify only essential heat sinking, and an integrated packaging approach that incorporated magnesium alloys for chassis designs, helped to achieve this objective. All mass savings directly translated into additional propellant (loaded before launch), which provided extended spacecraft maneuvering capabilities for science data acquisition both during cruise mode and at the asteroid.

The interior of the spacecraft contains the propulsion module. The propulsion module is located in the plane of the center of mass. It contains the propellant tanks, eleven monopropellant thrusters grouped into six different pods, and the 450-N bipropellant thruster. The location of the propulsion tanks is selected to maintain the spacecraft's center of mass along the vector of the 450-N thruster throughout the mission as the bipropellant is depleted.

MISSION ENVIRONMENTS

The two primary factors affecting the thermal design are spacecraft attitude and solar distance. The spacecraft attitude is dictated by the mission communication and navigation requirements that define the mission trajectory and optimal geometry. An important attribute of the selected mission geometry is that the Sun–spacecraft–Earth angle is always less than 40°, except for the first two months after the launch and the first two months after Earth flyby. The constrained Sun-angle pointing limits solar illumination of the aft deck instruments and thus reduces the variations in their thermal environment, which simplified the instrument design process. Because the instruments are fixed, instrument pointing must be achieved by rotating the spacecraft around the axis of the high-gain antenna while the spacecraft orbits the asteroid in a plane normal to the asteroid–Earth line. The solar loading on the spacecraft can therefore be defined in terms of a cone angle (θ) defined from the Sun line to the spacecraft $+z$ axis (Fig. 2), and a clock angle (ϕ) that defines the side panel illumination area as the Sun moves around the spacecraft $+z$ axis. As shown in Fig. 2, θ can vary from 0 to 45° during cruise mode, and from 0 to 30° at the asteroid. Angle ϕ can vary from 0 to 360° provided that θ is greater than 0°. The intensity of the solar loading varies as a function of the spacecraft distance from the Sun as depicted in Fig. 3.[2]

The mission trajectory brings the spacecraft within close proximity to the Earth during two separate periods. The first is immediately following the launch during the ascent and transfer phase. The primary heat inputs to the spacecraft during this phase are from free molecular heating following fairing jettison, and from heat soak-back from the booster third-stage engine dome. The relatively brief exposure to environmental inputs from the Earth ensures that the resulting heat inputs are negligible. The second period of Earth exposure occurs during the Earth swingby nearly 2 years after launch. During the Earth swingby, the NEAR spacecraft will experience transient heating effects from the Earth's albedo and from emitted infrared radiation.

THERMAL DESIGN

The thermal design of the spacecraft and instruments is simple and robust using a passive philosophy

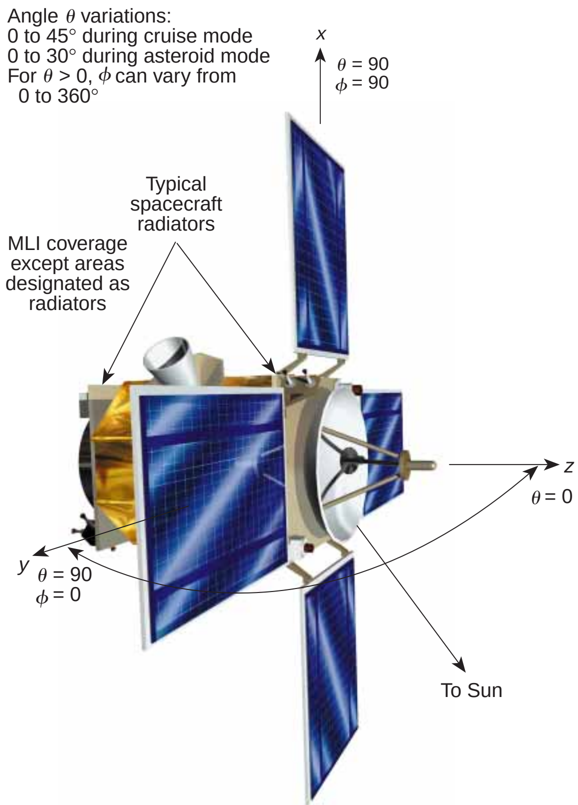

Figure 2. Spacecraft incident sun angles. (MLI = multilayer insulation.)

Figure 4. Photograph showing radiators and multilayer insulation.

of radiators, multilayer insulation (MLI), optical coatings, heaters and thermostats, and thermal isolation to maintain temperatures within qualification specifications. The spacecraft radiators are located on the main body side panels as shown in Fig. 4. Fifteen layer MLI blankets cover the entire spacecraft with the exception of designated radiators, RF surfaces, and the solar panels and their associated hinge mechanisms. The MLI outer layer is either 0.025- or 0.076-mm-thick aluminized kapton, and the radiators are covered with 0.127-mm-thick perforated silver Teflon. The instruments, battery,

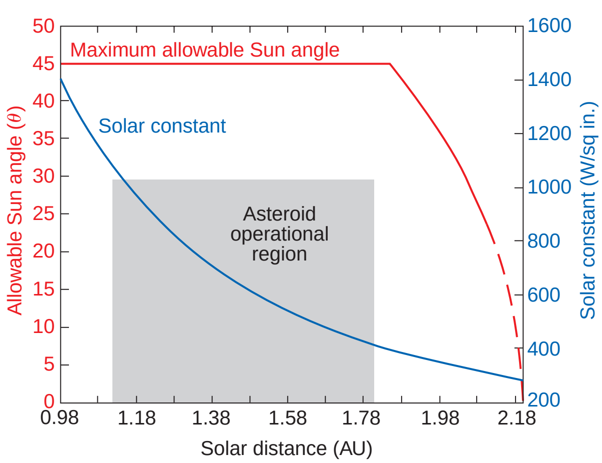

Figure 3. Mission solar intensity variation.

and propulsion subsystem are thermally isolated from the spacecraft. Thermal control for these components is accomplished by radiators, heaters, and mechanical thermostats. The inside of the spacecraft and all electronics are coated with high-emittance black paint. The spacecraft radiators are designed to maintain the internal cavity temperatures within component design limits under worst hot and cold scenarios. The radiators have a relatively small conduction coupling to the structure because of the side panel face skin thickness of 0.127 mm per face. The heat generated by the electronics is spread by the decks and radiated to the radiators and side panels. Electronics mounted directly to the spacecraft decks have power densities at or below 0.039 W/cm^2. Redundant solid-state power amplifiers (SSPA), each with a power density in excess of 0.194 W/cm^2, are colocated on a 3.175-mm-thick AlBeMet 162 heat sink–radiator, which is mounted diagonally opposed to the large velocity adjustment (LVA). Survival heater circuits were incorporated to maintain the SSPA temperatures above cold survival limits during SSPA non-operating conditions. Table 1 lists the worst-case hot and cold power dissipations for electronics and instruments.

Because spacecraft power is limited during the months surrounding spacecraft aphelion, it was necessary to allocate heater power with great discretion. Isolated instruments, the battery, and the propulsion

Table 1. NEAR power dissipation levels.

Component	Heat dissipation (W) Inner cruise (max power)	Outer cruise (min power)	Asteroid mode (min/max)
Predicted heater circuit powers[a]			
Spacecraft operational[b]	5.9/69.3	26.6.2/69.3	22.6/69.3
Instrument survival[c]	11.6/40.1	30.6/40.1	2.4/40.1
Instrument operational[c]	0.0/42.8	0.0/42.8	23.9/42.8
Propulsion system[d]	32.1/67.6	53.5/67.6	45.5/67.6
Internal shunts			
Aft deck	22.4	0.0[e]	0.0/22.4
Forward deck	0.0	0.0[e]	0.0/22.4
Attitude control			
RWA 1	5.5	3.5	3.5/5.5
RWA 2	5.5	3.5	3.5/5.5
AIU	10.0	9.0	9.0/10.0
IMU	27.0	22.0	22.0/27.0
Star tracker	7.8	6.5	6.5/7.8
Telemetry			
CTP 1	9.4	6.0	6.0/9.4
CTP 2	6.0	4.0	4.0/6.0
Solid-state recorder 1	0.0	0.0	0.0/5.0
Solid-state recorder 2	6.0	5.0	5.0/6.0
Flight computer	10.0	8.0	8.0/10.0
Power switching	1.0	1.0	1.0/1.0
Communications			
SSPA 1 or 2	31.0	24.0	24.0/31.0
SSPA converter 1 or 2	9.0	8.0	8.0/9.0
Receiver/exciter 1	9.5	6.5	6.5/9.0
Receiver/exciter 2	6.5	6.5	6.5/6.5
CDU 1	1.4	1.4	1.4/1.4
CDU 2	1.4	1.4	1.4/1.4
TCU 1	0.0	0.0	0.0/0.0
TCU 2	3.6	3.6	3.6/3.6
Power			
PSE	35.0	4.0	6.0/24.0
Battery	4.3	3.0	3.0/4.3
Instrument electronics			
Magnetometer electronics	0.0	0.0	0.5/0.5
NIS/Magnetometer DPU	0.0	0.0	7.0/7.0
MSI DPU	0.0	0.0	6.0/6.0
XGRS DPU	0.0	0.0	10.0/10.0
XGRS electronics	0.0	0.0	12.0/15.0
NLR electronics	0.0	0.0	6.0/8.0
NLR converter	0.0	0.0	4.0/6.0
HVCE	0.0	0.0	2.0/2.0
Instrument sensors			
Magnetometer	0.0	0.0	0.05/0.05
MSI FP detector	0.0	0.0	1.7/1.7
NIS IGA detector	0.0	0.0	0.7/0.7
NIS GE detector	0.0	0.0	0.5/0.5
NLR transmitter	0.0	0.0	0.6/0.6
NLR receiver	0.0	0.0	1.1/1.1
X-ray sensor	0.0	0.0	0.5/0.5
Gamma-ray sensor	0.0	0.0	0.07/0.07
Solar monitor	0.0	0.0	0.5/0.5

[a]Represents heater circuit (max predicted average power/measured peak power).
[b]HVCE, battery, star tracker, command and data handling, and RF panel heaters.
[c]All instrument heater power.
[d]Propulsion system includes thruster valves, LVA flange, tanks, and latch valve panel; 15.8 W of line heaters are not included.
[e]Analysis also with shunt powered.
Note: RWA = reaction wheel assembly, AIU = attitude interfere unit, IMU = inertia measurement unit, CTP = command and telemetry processor, SSPA = solid-state power amplifier, TCU = telemetry conditioning unit, NIS = Near-Infrared Spectrometer, XGRS = X-Ray/Gamma-Ray Spectrometer, MSI = Multispectral Imager NLR = NEAR Laser Rangefinder, HVCE = high-voltage control electronics, FP = focal plain, GE = germanium, LVA = large velocity adjustment.

subsystem all require survival and to some extent operational heaters. The spacecraft has a fixed bus voltage of 33.28 V DC that is independent of the battery when the battery is not discharging. The only expected battery discharge occurred during liftoff and early mission operations that included a 35-min eclipse as the spacecraft exited the Earth's shadow. Since the bus voltage is fixed and any battery discharge would indicate a power deficiency, it was decided to design the heaters for full power operations at 30.00 V DC. This gives an 80% duty cycle at peak bus voltage conditions and keeps the peak heater power at reasonable levels. All operational and survival heater circuits are redundant. Primary and secondary bus thermostat set points are offset to preclude any simultaneous heater operation during a given mode. Tables 2 and 3 list thermostat control ranges and capacities for NEAR survival and operational heater circuits, respectively. For NEAR shunt heaters, both forward and aft as well as primary and redundant, the capacity is 22.4 W @ 33.5 V. All shunt heaters are activated by ground command.

The spacecraft structure maintains the temperatures of electronics mounted to the decks. This maintenance is accomplished purely by electronic dissipations. To help increase thermal margins, 44 W of shunt power can be switched either into the spacecraft or to the backs of the solar panels when desired. During near-Earth solar conditions, the shunts are switched to the array. At conditions represented by aphelion, both shunts will be switched inboard. The spacecraft will use excess array power to enhance inboard temperature conditions. The thermal design does not depend on the auxiliary shunt power during cold conditions, which was verified during spacecraft TV testing.

Schedule and operational requirements dictated that the instruments, battery, and propulsion subsystem be thermally isolated from the spacecraft structure. The thermally isolated interface for these components

Table 2. Thermostat control ranges and capacities for NEAR survival heater circuits.

	Primary survival heater bus			Secondary survival heater bus		
	Thermostat control range		Heater	Thermostat control range		Heater
Component	Close temp. (°C)	Open temp. (°C)	capacity (W @ 33.5V)	Close temp (°C)	Open temp (°C)	capacity (W @ 33.5 V)
Visible image (MSI)						
FPD electronic housing	–41	–38	4.4	–44	–41	4.4
Optics housing	–35	–30	2.5	–41	–33	2.5
Infrared spectrograph (NIS)						
Main chassis	–40	–35	4.7	–44	–39	4.7
Detectors	–42	–38	6.1	–44	–41	6.0
Laser altimeter (NLR)						
Transmitter housing	15	17	3.6	14	16	3.6
Receiver housing	–25	–22	4.0	–29	–24	4.1
X-ray spectrometer						
Argon gas tubes	–17	–11	6.6	–28	–25	6.6
Stepper motor	–36	–31	0.6	–44	–40	0.6
Gamma-ray sensor	–11	–5	2.0	–17	–9	2.0
Solar monitor						
Argon gas tube	–17	–9	2.2	–28	–25	2.2
High-res. monitor	–25	–21	2.6	–29	–25	2.6
Magnetometer	Heater control based on uploadable set point, typically 0°C		0.8	N/A	N/A	N/A
HVCE chassis	–25	–22	2.7	–29	–25	2.7
Star tracker	–28	–23	6.4	–36	–32	6.5
Battery	1	5	11.2	0	5	11.2
RF panel	–26	–21	28.1	–29	–25	28.1
Command and data handling	0	6	20.8	–15	–12	20.8

Note: MSI = Multispectral Imager, FPD = focal plane detector, NIS = Near-Infrared Spectrometer, NLR = NEAR Laser Rangefinder, HVCE = high-voltage control electronics. NIS main chassis and detector survival heaters, as well as command and data handling survival heaters, contain multiple circuits.

Table 3. Thermostat control ranges and capacities for NEAR operational heater circuits.

Component	Primary operational heater bus: Thermostat control range, Close temp. (°C)	Primary: Thermostat control range, Open temp. (°C)	Primary: Heater capacity (W @ 33.5V)	Secondary operational heater bus: Thermostat control range, Close temp (°C)	Secondary: Thermostat control range, Open temp (°C)	Secondary: Heater capacity (W @ 33.5 V)
Visible image (MSI)						
Optics housing forward	16	25	6.0	15	25	6.1
Optics housing aft	16	25	5.6	15	25	5.6
Infrared spectrograph (NIS)	–16	–10	17.1	–18	–10	16.9
Laser altimeter	17	24	10.0	15	24	10.0
Gamma-ray sensor	Heater control based on uploadable set point, typically 20°C		4.1	N/A	N/A	N/A

Note: MSI = Multispectral Imager, NIS = Near-Infrared Spectrometer. NIS operational heaters contain multiple circuits.

minimized thermal interchange between them and the spacecraft. Therefore, the thermal design for the components could be developed in parallel with the spacecraft thermal design effort, assuming minimal heat soak-back and heat-leak effects from the spacecraft. In addition, the temperature requirements for these components are more stringent than those for the majority of the spacecraft bus components. The spacecraft structure and bus components can function properly over a fairly wide temperature range (–29 to +55°C); however, the instruments, battery, and propulsion module must be maintained within a tighter temperature range to ensure acceptable performance. The thermally isolated mounting configuration for these components allowed a fine-tuned thermal control scheme that provided optimized temperature control independent of the interface temperature. The propulsion subsystem was designed and built by GenCorp Aerojet in parallel with the spacecraft. Based on operational temperature requirements and power limitations, the propulsion subsystem was thermally isolated. This thermal isolation reduced the amount of heater power necessary to maintain proper tank and thruster temperatures at expected deep space environmental conditions when power is very limited. The propulsion subsystem's thermal design was flight-qualified to the cold design interface conditions in a thermal vacuum environment before delivery for integration, and the degree of thermal isolation at the mounting interface proved to be adequate.

The spacecraft thermal design has to accommodate the firing of the 450-N LVA thruster. Power subsystem and communication link requirements dictate the buried configuration of the LVA. Figure 1 shows the LVA placement on the spacecraft. Since the hottest portion of the thruster is buried inside the spacecraft, a gold-plated CRES heat shield is used to protect internal components from thermal back loading during engine firings. The solar arrays are also affected by plume impingement. However, since the firings occur at solar distances greater than 1.75 astronomical units (AU) where the reduced solar intensity (Fig. 3) results in lower array temperatures, no special plume protection was necessary for the arrays.

The high-gain antenna, thermally isolated from the spacecraft using MLI and low-conduction titanium feet, is a 1.5-m parabolic honeycomb dish. The facesheet material is a cyanate ester resin system using 0.356-mm-thick XN50 fibers with a Nomex (fire resistant) core. The reflector essentially points at the Sun unless the spacecraft is pointing to the Earth for a communication link. Naturally flat black, it was decided to coat the reflector's Sun-facing surface with highly diffuse, low-solar-absorptance MS-74 white paint. Although MS-74 bonds well to organic materials such as aluminum, it did not bond to the cyanate ester resin system. The MS-74 coating delaminated in ambient conditions before any thermal vacuum testing could be performed. Therefore, the reflector was stripped of all MS-74 coating and primer, thoroughly cleaned with a Scotch Brite scrub sponge and acetone, and then baked out at 75°C for 36 h (pressures were stable within 24 h). After researching white coatings with the desired thermo-optical properties that displayed compatibility with the cyanate ester resin system, it was decided to recoat the reflector with ITTRI S13GP/LO-1 white paint. The S13GP/LO-1 bonded well to the reflector and survived six cycles of spacecraft-level thermal vacuum testing between –90 and +50°C without exhibiting any evidence of delamination.

ANALYSIS

The NEAR mission is essentially steady state relative to environments and electrical loads. Transients occur only during spacecraft attitude changes, heater duty cycling, electrical load shedding in case of an anomalous power shortage (the spacecraft is designed to never draw power from the battery except for launch and the first 40 min of the mission), and the main engine (LVA) firing. Steady-state analysis cases were run for θ angles (Fig. 2) of 0, 20, and 45° for cruise mode and 0, 20, and 30° for asteroid mode with ϕ varying from 0 to 360° in 45° increments at each θ angle. The results of the analysis were used to characterize the thermal response of the spacecraft and the instruments as a function of Sun angle, electrical power, and solar distance. The hot analysis uses end-of-life material optical properties with varying steady-state Sun angles. The mission mode, either cruise or asteroid, defines solar distance, 0.98 and 1.13 AU, respectively, as well as expected worst-case electrical loads. Cold analysis uses beginning-of-life material optical properties with the Sun angle fixed normal to the solar panels (θ equal to zero). The Sun must be maintained at this angle to provide appropriate power for the spacecraft at solar distances of 2.2 and 1.75 AU, respective cruise and asteroid mode operation. Thorough analysis under worst hot and cold solar and electrical conditions optimized placement and sizing of all heaters and radiators. This operation was important because uncertainty in analysis could have led to increasing heater capacity due to questionable thermal margins.

Cruise Mode Operation

The NEAR mission is broken into two functional modes, cruise and asteroid. Cruise mode lasts nearly 3 years, from launch until acquisition of the asteroid mapping orbit. During cruise mode the instruments are essentially nonpowered, and the spacecraft will experience its maximum and minimum solar distances and Sun-angle variations. Normally, the spacecraft $+z$ axis is Sun pointing. During communication links, the Earth–spacecraft–Sun angle is limited to 45° with no projection limitation in the x–y plane. This condition could last as long as 24 h. Steady-state analysis was done to determine thermal impacts on instruments as a result of solar aperture loading and the introduction of solar flux on the once-shadowed spacecraft radiators. Power was maintained at worst-case 0.98 AU Sun-pointing conditions (Table 1). The power dissipation of the power system electronics could theoretically vary from 35 W peak to 23 W at a slew angle of 45° and 0.98 AU. Analysis showed small variations in temperature due to changes in Sun angle, although the worst hot case does occur at a Sun angle of 45°. Analysis also showed that infrared back loading from the solar arrays helps to balance the effects of Sun-angle variation on spacecraft temperatures.

The worst cold case is expected during cruise mode at 2.2 AU. The spacecraft must maintain its $+z$ axis pointing to the Sun so that the solar arrays have normal illumination to provide sufficient power for spacecraft operations. The electrical loads are about the same as near-Earth cruise except the power system electronics only dissipate about 4 W, and the solar constant has fallen off by about 80%. Large influential external surfaces, specifically the high-gain antenna and solar panels, have bulk temperatures below –60°C. The spacecraft was designed to maintain all temperatures 10°C inside cold qualification limits when powered to steady-state loads (listed in Table 1) with no internal shunt heater augmentation. When augmenting with shunts, predictions show a bulk spacecraft temperature increase of about 9°C. Based on early mission power generation, excess solar array power is predicted during aphelion; therefore, available internal shunt power is expected.

Asteroid Mode Operation

Asteroid mode occurs after the spacecraft has been inserted into the asteroid mapping orbit (following cruise mode), and lasts until all of the spacecraft propellant has been depleted (approximately 1 year). Asteroid mode operation was considered a driver for the hot case design only. During hot asteroid operation, the spacecraft and asteroid are 1.13 AU from the Sun. Instruments are fully powered using the expected worst hot case dissipations. The Sun–spacecraft–Earth angle is limited to 30°, which is driven by instrument imaging requirements. Spacecraft–Sun angle variations were analyzed in the same fashion as hot cruise mode except asteroid mode powers and the appropriate solar constant were used. It was apparent from the analysis that spacecraft temperatures were not extremely sensitive to Sun angle, although component temperatures did respond to impressed solar loading on nearby radiators. Instrument heater power analysis was performed at the furthest asteroid solar distance (1.75 AU) to verify that operational heater power was adequate with spacecraft interface temperatures at their coldest aphelion levels. To help cold-bias the analysis, only single instruments were powered to reduce the spacecraft deck heat loads and corresponding interface temperatures.

Transient Analysis

Special transient analyses were performed to characterize LVA firings, instrument operational scenarios, and expected post-launch cool-down temperature profiles. Transient analysis was also used to predict heater duty cycles and to verify the steady-state temperature results using the same dissipative and environmental conditions.

LVA Firing

During the mission, two major delta maneuvers are scheduled to adjust the trajectory of the spacecraft. The first is at mission aphelion, and the second is at asteroid rendezvous. The aphelion burn has a duration of approximately 5 min. Thermal effects from the burn are induced by spacecraft load management and plume impingement on the $+x$ and $+y$ solar arrays. Load management will be used if a low-solar-array current condition exists. Electrical loads, such as the tank heaters, would be switched off to gain power margin for propulsion subsystem operation. The analysis predicted benign temperature changes due to load management, and it is therefore not a concern. Plume impingement on two of the four solar arrays will cause local panel temperatures to rise from –60 to +75°C. Power analysis was performed to assess the impacts of the resulting asymmetric panel temperatures. The results of the analysis show only small perturbations in power generation because of asymmetric temperature transients. Internally, the spacecraft is cold enough to mitigate the thermal soak-back effects from the LVA heat shield.

The rendezvous burn has a duration of approximately 18 min. Thermal impacts are induced by plume impingement on the $+x$ and $+y$ solar arrays and thermal soak-back from the heat shield to the inside of the spacecraft. Figure 4 shows the effects of an 18 min LVA firing on the $+x$ and $+y$ solar arrays and the expected solar array panel temperature distribution. The peak heat flux was calculated to be 0.248 W/cm^2 at a location about 45.7 cm from the hinge edge.[3] The plume heating analysis was supplied by GenCorp Aerojet using the method of characteristics solution computer code RAMP2. The analysis assumed that the plume "sticks" to the panel, thereby yielding the highest heating rate. Worst-case predicted panel temperatures are on the order of 100°C as shown in Fig. 5. Although the predicted maximum temperature is outside the hot operational qualification limit, it was decided not to test the arrays any hotter than 80°C based on the results of an in-depth risk analysis. Thermal back loading from the heat shield causes a worst-case local temperature rise of 29.5°C on the terminal board, with the peak temperature staying below the qualification limit of 60°C. Therefore, no thermal concerns are associated with LVA engine firing.

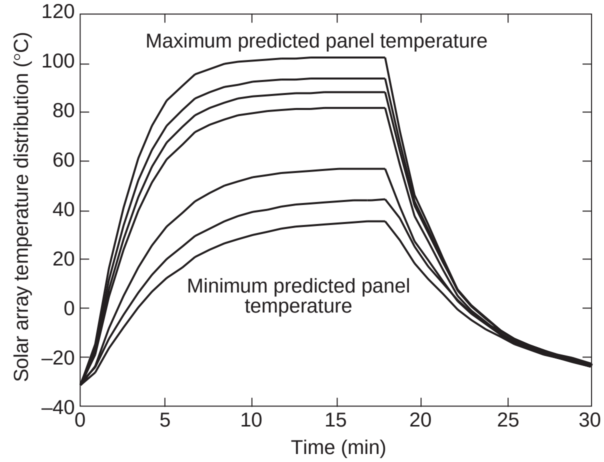

Figure 5. LVA plume heating analysis. (LVA = large velocity adjustment.)

Post-Launch Analysis

Transient analysis was also used to determine spacecraft temperatures and heater duty cycles during the launch sequence. During the first 40 min of the mission, the spacecraft was operating on battery power and the spacecraft heaters were disabled to reduce the risk of over-discharge. The transient launch analysis had predicted that the first heater activation would not occur for at least 2 h following launch, so it was decided to disable all instrument, propellant tank, and battery heater circuits at launch.

TESTING

The compressed design and development timeline for the spacecraft and instruments left little schedule margin for redesign efforts to fix problems discovered during final thermal acceptance testing. Therefore, to reduce technical and schedule risks, a series of engineering-level thermal vacuum tests were used to verify essential interfaces and performance characteristics during the development cycle. These tests were conducted on engineering prototypes or hardware models for each facility instrument or sensor. Data and insight gained from these tests were used to incorporate minor design adjustments that improved operational functionality and performance while increasing the probability for a successful science mission. Performance data gained from these tests, in conjunction with operational performance measured during each component's stand-alone final qualification testing, provided confidence in the thermal design. This testing approach also ensured that each component was verified before the spacecraft-level thermal vacuum test, thus simplifying the system test by eliminating the need for individual component verifications.

The spacecraft underwent thermal vacuum testing from 1 November 1995 through 20 November 1995 at NASA GSFC in Chamber 290. The test objectives were twofold: First, verify the thermal design at specified equilibrium conditions and use measured temperature and power data to correlate and refine the flight thermal mathematical model. Second, thermally cycle the spacecraft through various test temperature ranges while exercising electrical components. The thermal cycling was intended to stress spacecraft components and connectors while accelerating incipient failures in marginal designs.[4]

Seven thermal balance cases were simulated: three at outer cruise (2.2 AU), three at inner cruise (0.98 AU), and one at near-asteroid (1.13 AU). Results from the balance testing showed nominal performance for the thermal control subsystem. No temperatures were out of limits, and all heater circuits and flight telemetry sensors were checked and verified for proper operation. The flight thermal mathematical model, which was modified to accommodate the test configuration, yielded accurate predictions and required only minor adjustments to match test temperatures to within 5°C. The thermal vacuum balance test was a complete success.

Four hot and cold functional plateaus were completed during the 20 days of testing. A total of 157 h of cold soak and 172 h of hot soak were recorded. No functional anomalies for flight hardware or software were discovered; however, a few ground support equipment connectors exhibited unexpected behavior during forward shroud transitioning from –100 to +50°C. No corrective action was required during the test. It was determined at the conclusion of the test that these connectors were loose, and the extreme temperature swings caused unexpected electrical noise as a result of relative motion in the connector.

EARLY MISSION PERFORMANCE

Early mission temperature telemetry data show the thermal control subsystem operating as expected. All temperatures are well within the required limits. Flight thermal model predictions have compared favorably to the flight temperature data when using measured power and simulated Sun-pointing solar conditions. To this date, all spacecraft subsystems have performed nominally.

CONCLUSIONS

The spacecraft thermal design has shown, via analysis, test, and early mission data, that acceptable thermal performance can be expected for all phases of the NEAR mission. The system and mission requirements, both specified and implied, shaped the final thermal design of both the spacecraft bus and the facility instruments. The design, which incorporates only passive thermal control hardware augmented by heater circuits, maintains all components within acceptable temperature ranges without adversely affecting the overall spacecraft system performance.

The use of innovative analysis and post-processing software in conjunction with the Gaski SINDA and SINDA/FLUINT thermal solution programs allowed rapid and accurate inputs to proposed design and mission changes. These timely responses were critical in supporting the compressed design and development schedule that the NEAR program dictated. Selective thermal testing of prototypes for key interfaces and mission critical items also provided early verification for thermal designs and the resulting confidence gained from proven functionality.

The results presented illustrate the robustness of the thermal design and confirm that simple, flight-proven methods of spacecraft and component thermal control can be used effectively for complex spacecraft and missions.

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