2008-01-2123

Return to Mercury: An Overview of the MESSENGER Spacecraft Thermal Control System Design and Up-to-Date Flight Performance

Carl J. Ercol
The Johns Hopkins University Applied Physics Laboratory, Laurel, MD 20723-6099

ABSTRACT

Launched on August 3, 2004, MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) will be the first spacecraft to orbit the planet Mercury. Designed and built by The Johns Hopkins University Applied Physics Laboratory in conjunction with the Carnegie Institution of Washington, MESSENGER will study Mercury during a 1-year orbital phase that will begin in March 2011. Currently the spacecraft is in the middle of a 7-year cruise phase that so far has included a flyby of the Earth (August 2005), two flybys of Venus (October 2006 and June 2007), and the first of three flybys of Mercury (January 2008). The January 2008 Mercury flyby marked the first spacecraft visit since Mariner 10 (1975) and made MESSENGER the first spacecraft to encounter Mercury when near the planet's perihelion. This paper will provide an overview of the thermal design challenges for both the cruise and orbital phases, the solutions implemented to resolve those challenges, and the flight temperature and power data that verify the performance of the thermal control subsystem over the mission to date.

INTRODUCTION

The MErcury Surface, Space ENvironment, GEochemistry, and Ranging (MESSENGER) mission will characterize Mercury in detail by observing the planet from orbit for one Earth year. Although it had long been desired to follow the initial flybys with an orbital mission, early studies had deemed this infeasible, or at least prohibitively expensive, due to mass and thermal constraints. MESSENGER utilizes a trajectory (illustrated in Figure 1) that was discovered by analysts at the Jet Propulsion Laboratory (JPL) [1,2] and later refined by analysts at The Johns Hopkins University Applied Physics Laboratory (APL). This mission design has allowed a launch mass of 1107 kg (including propellant) on a Delta-II heavy while keeping the overall mission cost within the allowable cap. Flybys of Earth, Venus, and Mercury itself are interspersed with five large deterministic deep-space maneuvers (DSMs) that target the spacecraft for its Mercury orbit insertion (MOI) maneuver in 2011.

The highly varying thermal environments expected during the cruise and Mercury orbital phases were a key engineering challenge in the spacecraft design. Driven by solar distances as small as 0.30 AU, the spacecraft thermal protection is accomplished with a large sunshade that shields most spacecraft components from direct solar exposure and allows them to operate at conditions typical of other interplanetary spacecraft without much temperature variation. After launch and during the early part of the mission (known as the outer cruise phase) when the trajectory took MESSENGER multiple times to solar distances >0.95 AU (shown in Figure 2), the spacecraft was flown in a reverse attitude, exposing the core vehicle to solar heating. During this phase of the mission when solar-array output power was the least, the reversed flight configuration substantially reduced heater-power usage while passively maintaining all spacecraft temperatures well within allowable flight limits. Once in orbit at Mercury, the orbital geometry as a function of Mercury True Anomaly (MTA) was chosen via comprehensive thermal environment analysis to minimize the thermal hazards on the day side of Mercury [3] by factoring analytical thermal constraints into the mission design. During this time, MESSENGER will experience steady-state heating from the Sun and transient heating from Mercury, both of which are functions of Mercury's solar distance, which varies between 0.3 AU (perihelion) and 0.46 AU (aphelion). Because most of the spacecraft hardware is protected by the sunshade, no specialized thermal designs were required. Certain hardware [the solar panels, sunshade, phased-array and low-gain antennas, and Digital Sun Sensor (DSS) heads] is necessarily exposed to the Sun throughout the mission and was specially designed to handle the temperatures and solar flux input expected at 0.3 AU [4-7].

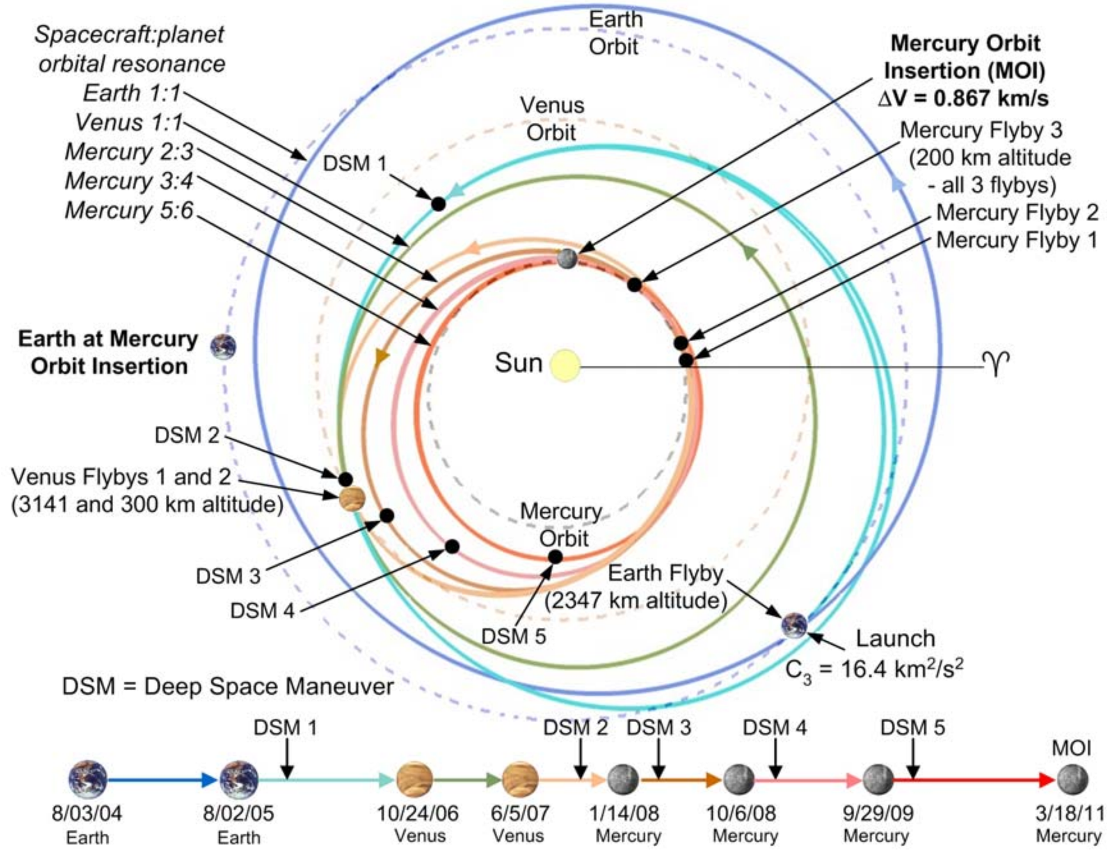

Figure 1. MESSENGER's heliocentric trajectory (viewed from north of Earth's orbital plane).

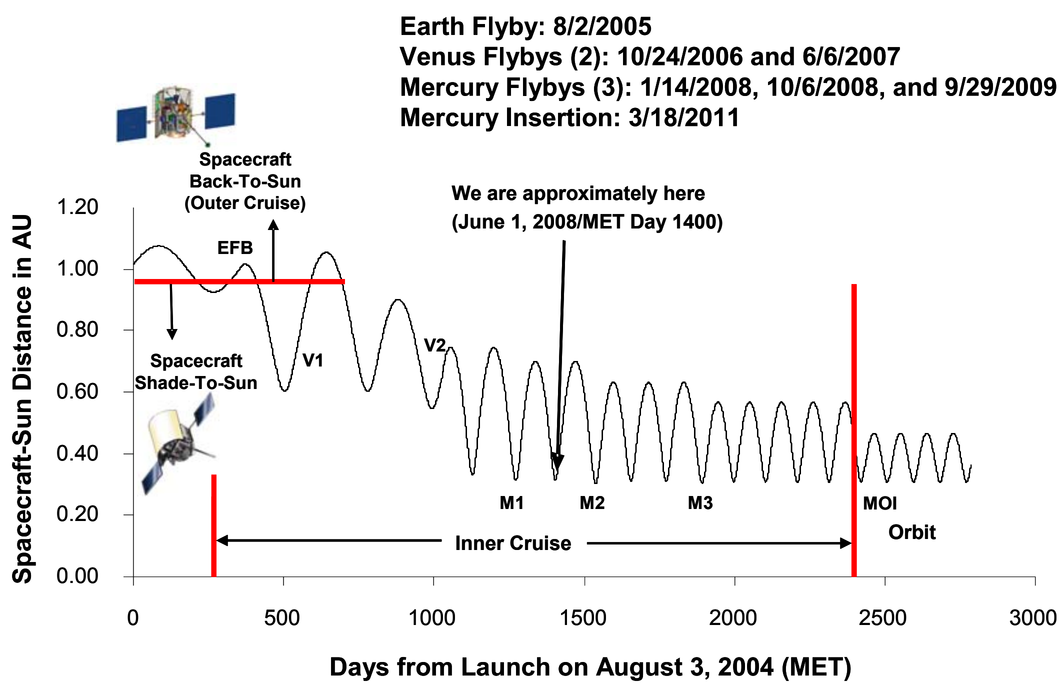

Figure 2. The spacecraft-to-Sun distance variation as a function of elapsed mission day. Key mission events are highlighted. Currently, the MESSENGER mission is approximately at the midpoint of its duration. EFB, Earth flyby; MET, mission elapsed time.

The MESSENGER mission is a collaboration between the Carnegie Institution of Washington (CIW) and APL and was selected as the seventh mission in NASA's Discovery Program, with a formal project start in January 2000. The spacecraft engineering and science instrument design evolved over the 3-year period from January 2000 to spring 2003. Assembly and integration of the spacecraft began in February 2003, and testing continued up to launch in August 2004. Flight operations are now supported from the Mission Operations Center at APL with communications through NASA's Deep Space Network (DSN) antennas.

OVERVIEW OF SPACECRAFT THERMAL DESIGN

The thermal design and operation of the MESSENGER spacecraft addressed three mission phases: outer cruise, inner cruise, and Mercury orbit. During the inner cruise and orbital phases, the thermal design relies on a ceramic-cloth sunshade to protect the vehicle from the intense solar environment encountered when inside of 0.95 AU. As shown in Figure 3, the sunshade creates a benign thermal environment when oriented with the –*Y* axis pointed toward the Sun, allowing for the use of standard electronics and electrical components and thermal blanketing materials. The solar arrays, sunshade, DSSs, and phased-array antennas, which are shown in Figures 4 and 5, are Sun-exposed components that required non-standard thermal design and specialized construction. These components have been designed to operate throughout the Mercury year and also during orbits that cross over one of Mercury's "hot poles" that face the Sun at Mercury perihelion. When at spacecraft perihelion, the sunshade, the solar arrays, the sunshade-mounted DSSs, and the sunshade-mounted antenna suite will experience as much as 11 times the solar radiation near Earth. During this time, the sunshade temperature will rise to >300°C. In certain orbits around Mercury, the spacecraft will pass between the Sun and the illuminated planet for ~30 minutes. During this period, the sunshade will protect the spacecraft from direct solar illumination, but the back of the spacecraft will be exposed to the hot Mercury surface. Components such as the battery and star trackers are positioned such that the spacecraft body blocks a substantial portion of the planet view, minimizing direct radiation from the planet surface when the spacecraft is in nominal operation. High-power spacecraft electronics that require dedicated radiators could not be packaged in a manner similar to that of the battery to reduce environmental heating from Mercury. These electronics boxes instead required specialized thermal design to allow for full, unrestricted operation during all parts of the orbital mission phase. Diode heat pipes, which are shown in Figure 6, were used in both the spacecraft and imager thermal designs to protect the attached components when radiator surfaces are exposed to day-side heating from Mercury. Thermal model results illustrating the variation of Mercury-induced heating received by MESSENGER are shown in Figure 7. During this peak heating period, the diode heat pipes will effectively stop conducting when the radiator surface becomes hot, as simulated during spacecraft-level thermal vacuum testing and shown in Figure 8. Once the planetary heating decays and the radiator cools, the heat pipe will resume conduction, restoring normal thermal control. Analysis of the near-planet environment as a function of orbit geometry and planet position was integrated into the mission design and has helped to phase the orbit plane relative to solar distance, minimizing infrared heating of the spacecraft by the planet and thus minimizing the mass required to accommodate such heating.

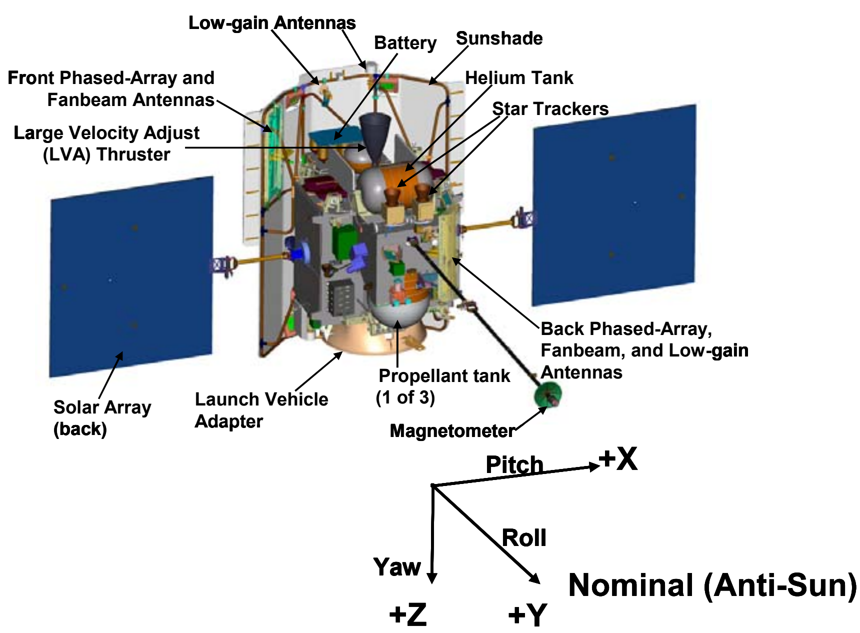

Figure 3. The MESSENGER spacecraft and flight coordinate system.

Figure 4. The MESSENGER spacecraft 2 weeks before launch. Highlighted here are the ceramic-cloth sunshade, the four DSSs with attenuating filters, the phased arrays and low-gain antennas under high-temperature radomes, and the 4.4-N pro-Sun thrusters.

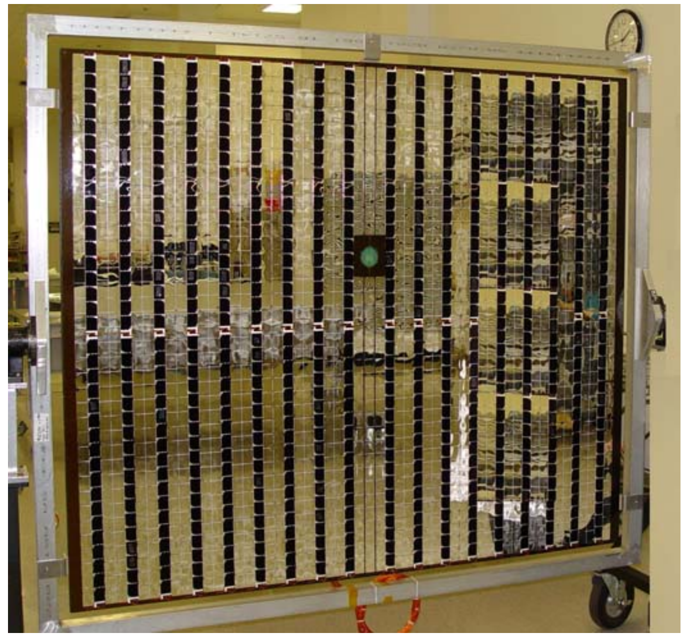

Figure 5. The MESSENGER +*X* solar array before spacecraft integration. Each solar array has a 2:1 ratio of OSRs to triple-junction gallium-arsenide solar cells.

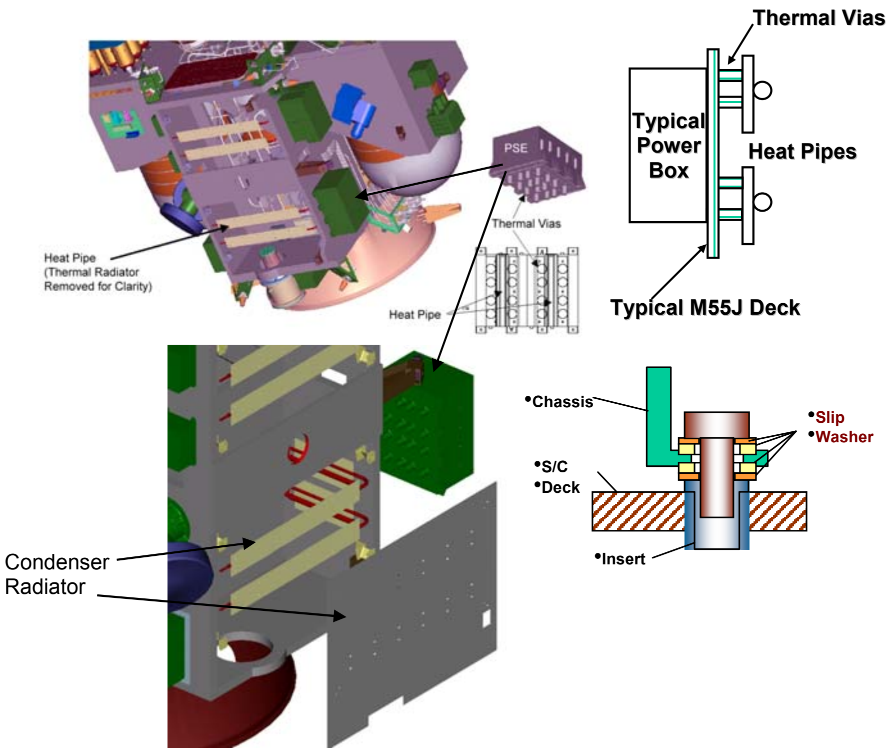

Figure 6. Driven by the Mercury orbit phase, diode heat pipes were used to protect high-power electronics. The PSE shown here required dedicated radiators. Because the MESSENGER structure was fabricated from a composite with a small coefficient of thermal expansion, a slip-mount design was used to fasten electronics boxes to the structure, and integral thermal vias connect specified electronics boxes directly to the heat pipes. S/C, Spacecraft.

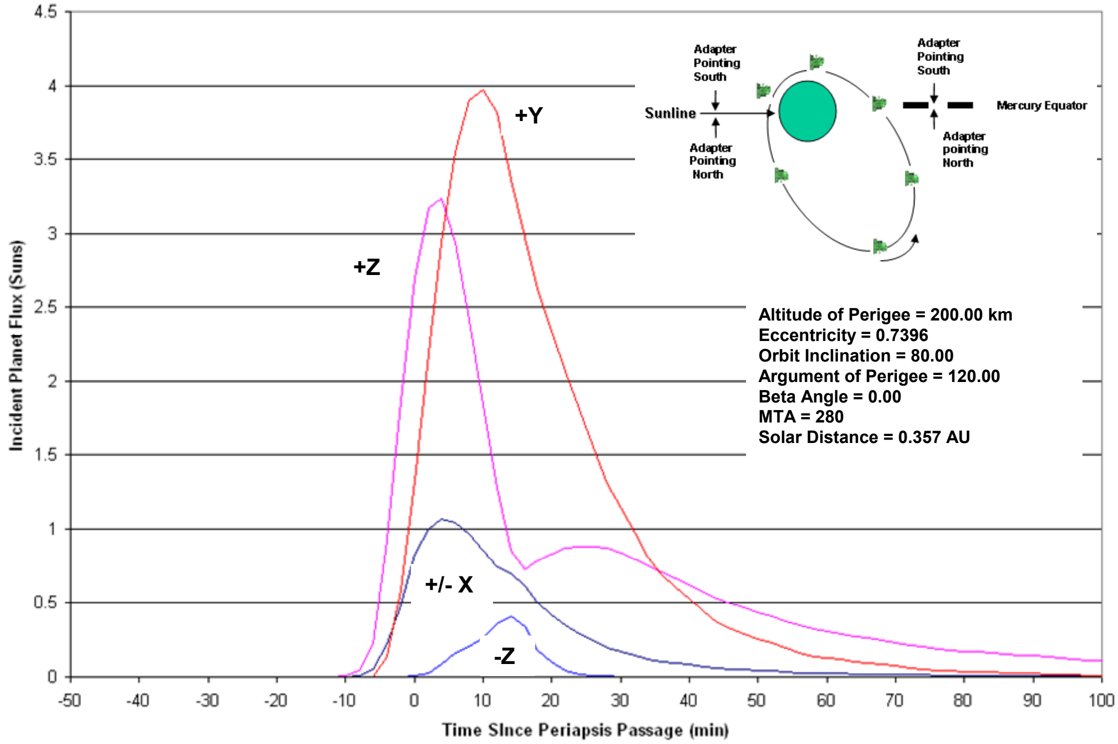

Figure 7. Incident thermal flux emitted by Mercury as seen at different locations on the spacecraft. Spacecraft thermal-control radiators are located in the *–Z* and ±*X* directions. Liquid-trap diode heat pipes constructed from aluminum and filled with ammonia were necessary to protect electronics connected to the ±*X* radiators from thermal energy back-flow when MESSENGER receives peak infrared heating. Approximately 1.2 m^2 of ±*X* radiator area was required to dissipate electronics waste heat. As shown, the +*Y* and +*Z* directions are not desirable locations for radiators.

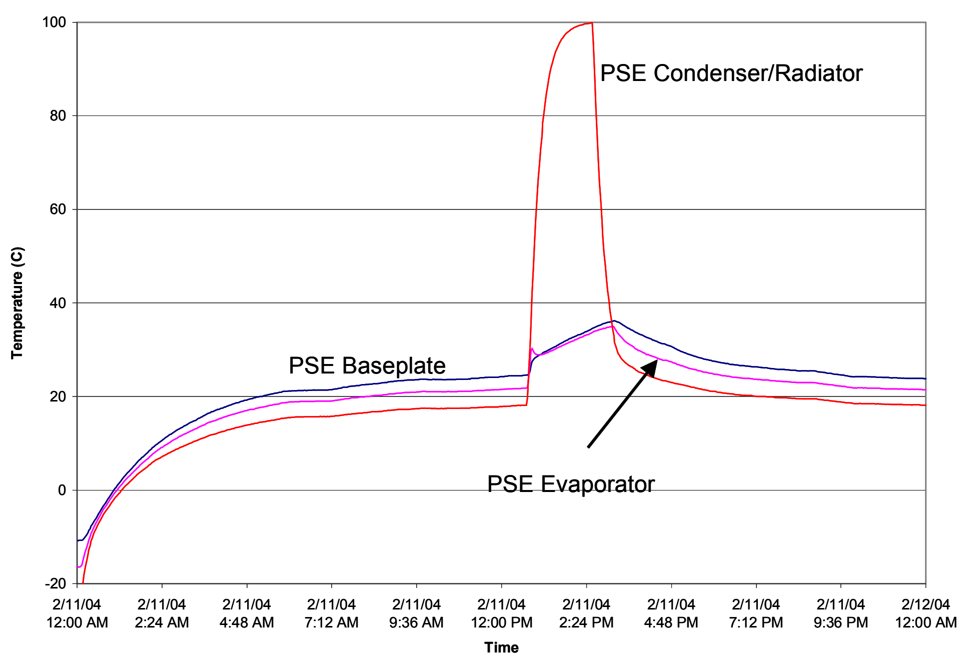

Figure 8. PSE temperature response for a simulated Mercury orbit during spacecraft thermal vacuum testing at GSFC in February 2004. This figure represents the expected behavior for boxes connected to diode heat pipes while in orbit at Mercury. Barring any spacecraft attitude anomaly, the diode heat pipes will remain dormant during the remainder of inner cruise.

The MESSENGER spacecraft and the associated Sun-illuminated component testing presented a challenging set of problems from technical, cost, and schedule aspects. It was decided early in development that the system-level thermal vacuum test would be done at the NASA Goddard Space Flight Center (GSFC) in a non-solar simulator environment. Specialized solar-simulation testing of the engineering-model solar arrays, DSS heads, radio-frequency (RF) antennas, and sunshade were performed at the NASA Glenn Research Center (GRC) Tank 6 thermal-vacuum chamber. Originally designed to simulate near-Earth solar conditions for the Solar Dynamic Power Experiment, the test setup was modified to produce an 11-Sun-equivalent environment over ~1.5 m2. This illuminated area proved large enough to test the various solar-array designs, DSS head configurations, sunshade design, and antenna components simultaneously while keeping the ratio of testing cost to components tested small.

Much emphasis has been placed on the extreme heat and high temperatures associated with a spacecraft orbiting Mercury. With the sunshade pointing toward the Sun and the spacecraft in nearly complete shadow, the spacecraft will use heater power to maintain critical temperatures. During outer cruise, when the spacecraft was outside 0.95 AU, MESSENGER was flipped (configured to fly with the sunshade pointed away from the Sun), such that the anti-Sun side was illuminated so as to reduce heater-power consumption and increase solar-array power margin. MESSENGER flew in this reversed orientation during intervals between launch and June 2006, and on the basis of measurements made during flight, the heater-power difference using this technique averaged ~220 W [8].

MISSION OVERVIEW

Precluded by Discovery mission guidelines was the use of the costly Atlas- or Titan-class boosters, which would have resulted in higher available launch mass and potentially a more direct trajectory into Mercury orbit. Instead, MESSENGER was launched on the smaller and less expensive Delta 7925H-9.5 and utilized a ballistic trajectory with Earth, Venus, and Mercury gravity-assist maneuvers in conjunction with 2250 m/s of onboard velocity adjustment (ΔV) to achieve MOI. Comprehensive analysis involving the lead mission design team at APL showed that the 2004 launch opportunities offered the best performance since 1997 and before approximately 2011 for a ballistic trajectory mission using the Delta-II launch vehicle for a payload of MESSENGER's mass. There were three opportunities for MESSENGER to launch in 2004, with varying combinations of Venus, Mercury, and Earth flybys (Table 1) [9]. From the beginning of the program, the March launch opportunity was the mission baseline trajectory due primarily to the shorter flight time and the less severe thermal environments when in orbit around Mercury. Schedule delays and additions to spacecraft testing plans forced the project to shift to the last opportunity, the current flight trajectory, increasing the flight time and moving the orbit insertion date from 2009 to 2011. Thermally, the change from the March to the August launch date translated to ~40% higher peak heating rates when orbiting Mercury due to the shift in injection point caused by the new arrival time. Instead of having the minimum sub-solar crossing orbit occur at MTA 240° (0.41 AU), the August launch has the minimum sub-solar crossing orbit at MTA 280° (0.35 AU). Because the spacecraft thermal requirements regarding orbital heating [3] were factored into the mission-design constraints, there was no major impact on the spacecraft thermal design because the worst-case conditions were always assumed and the back-up mission designs were well defined early in the program. For the near-Earth thermal conditions, all of the trajectories have mission aphelia outside of the Earth's maximum solar distance, so given steady-state thermal environments associated with near 1.0 AU solar distance, the overall impact to the spacecraft thermal design and operational strategy was negligible. The August launch, however, did increase the number of spacecraft excursions to solar distances >1.0 AU, so the number of spacecraft maneuvers, using reaction wheels and not thrusters, that rotated the shade to and from the Sun went from one to three.

Table 1. MESSENGER 2004 launch opportunities.

	Month		
	March	**May**	**July/August**
Launch dates	10-29	11-22	**30 July to 13 August**
Launch period (days)	20	12	**15**
Launch energy (km^2/s^2)	≤15.700	≤17.472	**≤16.887**
Earth flybys	0	0	**1**
Venus flybys	2	3	**2**
Mercury flybys	2	2	**3**
Deterministic ΔV (m/s)	≤2026	≤2074	**≤1991**
Total ΔV (m/s)	2300	2276	**2277**
Orbit insertion date	6 April 2009	2 July 2009	**18 March 2011**

Boldface text in the right column indicates actual launch date.

OUTER CRUISE PHASE

From launch until June 2006, there were three prolonged periods (see Figure 2) when the mission trajectory achieved solar distances >0.95 AU and the MESSENGER spacecraft was flown in the reverse-sunshade orientation. This flight configuration allowed for select illumination of the spacecraft body that reduced heater-power usage to nearly zero and allowed for large power margins while maintaining component temperatures within their allowable flight limits. This capability also allowed MESSENGER to operate easily between 0.95 and 1.08 AU and gave unconstrained flexibility on outer solar distance when the mission design team was planning for backup missions that introduced different mission trajectories and outer solar-distance excursions without complicating the spacecraft thermal or power designs. To achieve acceptable temperature control, spacecraft radiator surfaces used to maintain temperature for the battery and other critical electrical components were always kept orthogonal to the Sun, while ~25% of the spacecraft multi-layer insulation (MLI), the helium tank, and most of the thrusters were fully illuminated. Because this flight configuration was designed exclusively to minimize the effects of solar distance on the spacecraft power and thermal subsystem designs, certain operational constraints were enforced to prevent temperature excursions above the allowable flight limits for some components. Steady-state Sun elevation and yaw angles were limited to a ±5° cone centered along the spacecraft +*Y* axis (the principal axis corresponding to the magnetometer boom) so as to reduce solar exposure of spacecraft radiators and mitigate solar trapping inside of the adapter and the star tracker baffles. Because of the high spacecraft time constant, transient events of ≤2 hours requiring Sun angle excursions outside of the ±5° cone were acceptable and allowed for complete flexibility when performing spacecraft propulsive maneuvers, certain instrument calibrations, and post-launch hardware-commissioning activities.

INNER CRUISE PHASE

Once power margins were deemed to be acceptable, the inner cruise phase began on March 8, 2005, when MESSENGER was transitioned into the nominal sunshade-to-Sun orientation from the reverse-sunshade orientation that had been maintained since launch. It was apparent within the first 24 hours from the transition that the sunshade was extremely effective at insulating the spacecraft from the Sun. The nominal spacecraft power of ~220 W that was experienced when in the reverse-sunshade configuration more than doubled as dormant heaters became active (as shown in Figure 9). MESSENGER relies on heater power to keep components within allowable flight temperature limits, and because approximately one-half of spacecraft launch mass was propellant, the biggest user of heater power was the propulsion system. To manage the main-tank heater-power usage while maintaining propellant-tank temperature, software-based set-point control has been critical to spacecraft power management by allowing the desired control dead-band values to be uploaded into electrically erasable programmable read-only memory (EEPROM) storage with simple ground commands. Typical for eclipse and high-power spacecraft events, tank set points are lowered to eliminate heater-power demand, and once the event or activity is completed, the set points are raised back to the nominal values. Because the tanks by design have a very large time constant, very little change in bulk tank

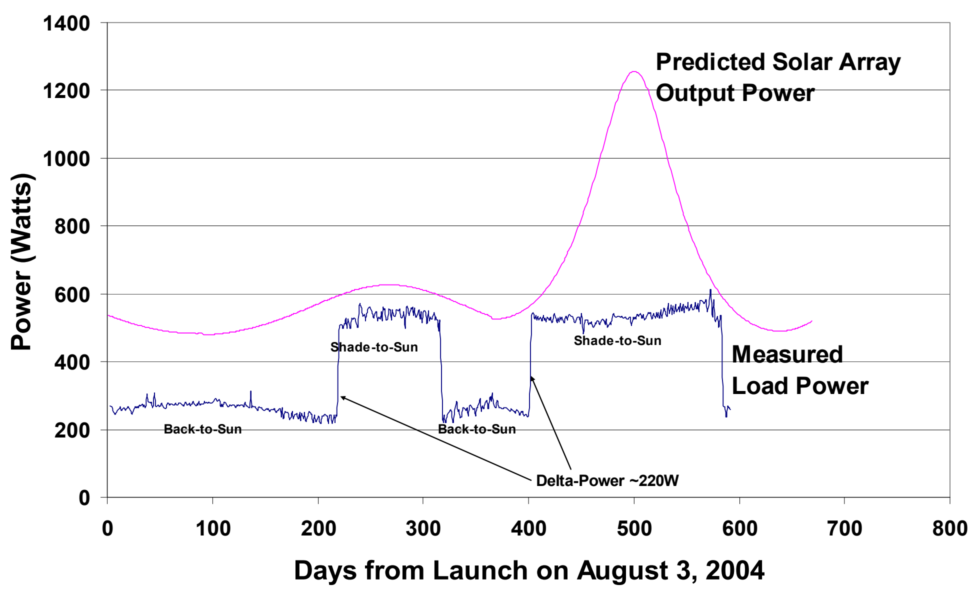

Figure 9. Power lean near Earth's solar distance, the MESSENGER spacecraft was designed to be maneuvered so as to illuminate the spacecraft body, represented by the back-to-Sun label, as a means to reduce heater-power demand. When the power margin was adequate (<0.95 AU), the spacecraft was maneuvered into the shade-to-Sun (nominal flight attitude) orientation. As shown, the power difference between the two Sun orientations was ~220 W.

temperature or pressure is experienced. By the nature of the trajectory, power margins increased daily because of the shrinking solar distance and because power not used was effectively left on the solar arrays. Once inside of 0.8 AU, the solar arrays produce more power than can be processed by the power system electronics (PSE), which means that temperature compensation for the solar arrays can be achieved by tilting them off normal to the Sun. The MESSENGER propulsion system has been maintained at near-room-temperature conditions since launch by efficiently utilizing the tank software control.

As the spacecraft nears the Sun, certain spacecraft hardware experiences temperature increases proportional to the inverse squared solar distance. Because distance changes are small on any given day, the spacecraft and associated external hardware are in essentially quasi-steady state. Therefore, to assess spacecraft performance, optical property stability, and sunshade degradation, certain spacecraft hardware that exhibits sensitivity (or lack thereof) to solar distance is closely monitored, and results are compared with those recorded when the spacecraft was at equivalent solar conditions. Analysis of flight temperature and heater-power data relative to the battery, solar arrays, sunshade-mounted components, and diode-heat-pipe radiators has been used over the duration of the mission as the spacecraft solar distance has varied. Comparison of these data has been used to quantify performance and identify any foreseeable issues that could result in unexpected temperature excursions caused by optical property shifts or thermal protection degradation.

Battery temperature control relies on two redundant heater circuits that are independently controlled by single mechanical thermostats. Heater control is augmented with 5 W of internal heat dissipation resulting from the constant trickle charge rate that maintains the battery at 100% state of charge. Each circuit has a peak power at 32.8 voltage direct current (VDC) of 32 W. The thermostats are separated by ~1° so as to avoid dual activation. Located directly behind the sunshade and shown in Figure 10, the battery experienced temperatures and heater-power duty cycles as illustrated in Figures 11 and 12. During reverse-sunshade orientation when the spacecraft was near 1.0 AU, the battery was maintained at <0°C with trickle charge and a small amount of heater power. When the spacecraft was flipped and the sunshade was oriented toward the Sun, the battery temperature was maintained at a nearly constant temperature of approximately −5°C with 32 W of heater power and 5 W of trickle charge. As the solar distance decreased from near 1.0 AU to 0.63 AU, battery temperature increased from −5°C to approximately −1°C with the primary heater at 100% duty cycle. Once inside of 0.63 AU, the battery temperature cycled between −5°C and −1°C with the duty cycle varying as a function of solar distance (Table 2). To date, MESSENGER has twice been inside of 0.33 AU, and the battery duty cycle

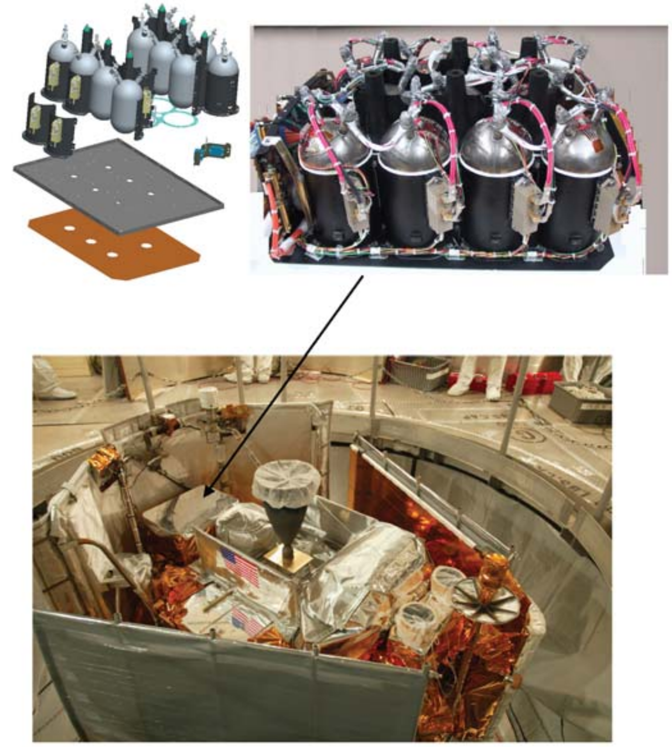

Figure 10. Packaged directly behind the sunshade on the –*Z* deck, the MESSENGER battery consists of 11 two-cell NiH2 common pressure vessels housed in a three-piece clam-shell-type mechanical design that meets all thermal-gradient requirements. Although the battery placement provides ideal protection from Mercury's day side when in orbit, the space-facing radiator used for heat rejection actually receives detectable heat input from the back of the sunshade as the spacecraft solar distance decreases. When the spacecraft reaches the mission minimum of 0.30 AU, the frontal sunshade temperature will approach 350°C. To date, the minimum spacecraft solar distance has been 0.31 AU. Also located on the –*Z* deck are redundant ASTs and redundant Solid-State Power Amplifiers (SSPAs).

Table 2. During inner cruise, the battery-heater duty cycle is affected by sunshade frontal temperature, which is a function of solar distance.

Solar Distance (AU)	Heater Duty Cycle (%)
>0.65	100
0.63	93
0.56	89
0.50	81
0.42	70
0.39	51
0.31	34

When the spacecraft was flown in the reverse-sunshade orientation during outer cruise, the battery-heater duty cycle averaged between 0 and 20%.

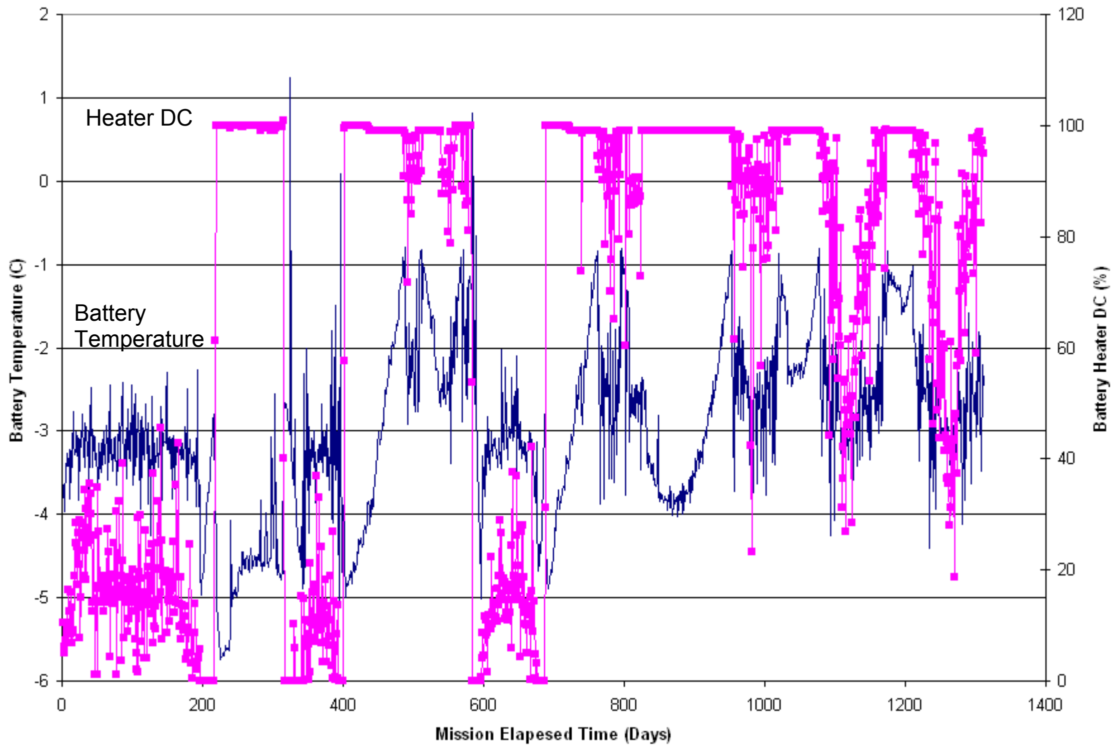

Figure 11. MESSENGER battery thermal performance as a function of elapsed mission day. Other than a few operational excursions, the bulk battery temperature (blue) has been maintained at <0°C through the mission to date by using heater power (pink). DC, Duty cycle.

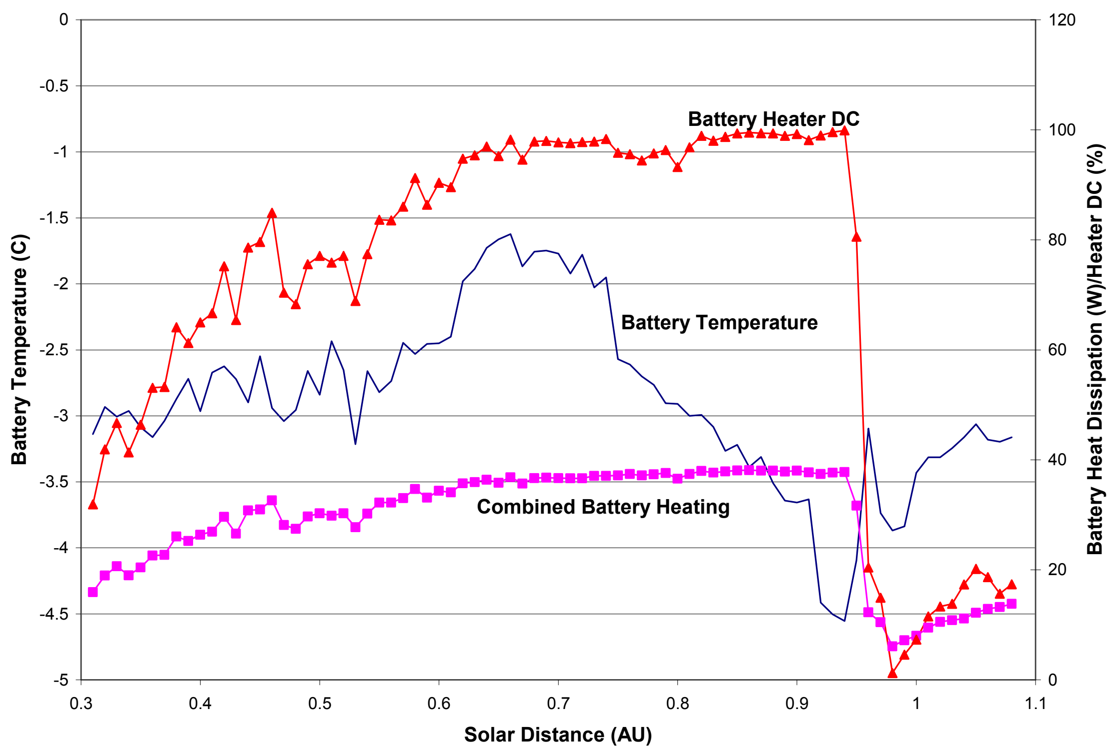

Figure 12. MESSENGER battery performance as a function of solar distance and electrical heat dissipation (trickle charging plus heater). The battery heater begins to cycle at ~0.65 AU. DC, Duty cycle.

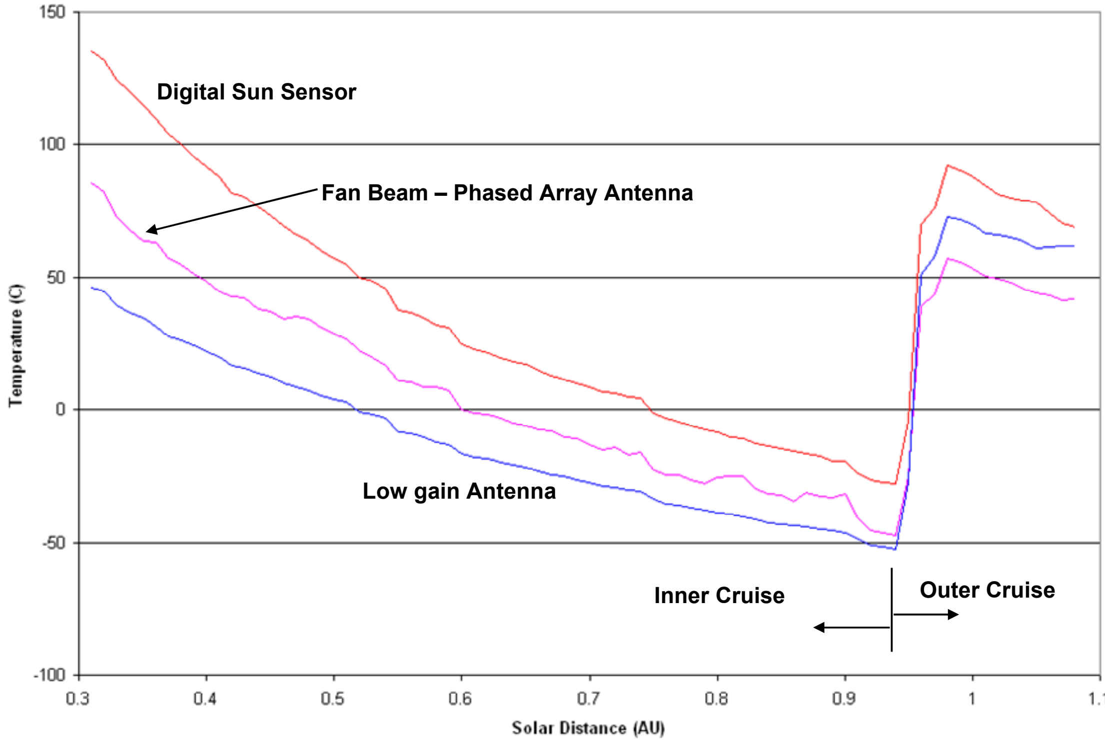

Figure 13. The sunshade-mounted components (see Figure 4) behave nominally as solar distance decreases. Comprehensive solar-simulation testing at the GRC Tank 6 facility ensured that these components were designed to operate properly operate in MESSENGER's high-solar-flux environment without any surprises. MESSENGER is the first 3-axis spacecraft to operate inside of 0.45 AU.

was measured to be between 39% and 41%. This is an indication that the thermal performance of the sunshade has not shifted dramatically, if at all, and the battery temperature remains <0°C.

Mounted on the sunshade frame are four DSSs, three low-gain antennas, and the combined phased-array and fan-beam antenna assembly. To provide the spacecraft with hemispherical and directional antenna coverage along with a Sun-safe attitude determination, these components view the Sun continuously and were thermally designed and tested to meet all operational requirements when near the Earth and when at Mercury perihelion. The temperatures of these components as functions of solar distance are shown in Figure 13. It can be seen that the temperature increase of each component follows an inverse-squared-distance law.

As with the sunshade-mounted components, the solar arrays always receive direct illumination from the Sun and had to be designed to operate and survive over a very broad range of environments. Thermal control of the solar arrays is managed by using a 2:1 packing factor ratio of optical solar reflectors (OSRs) to triple-junction gallium-arsenide solar cells combined with an ability to rotate each solar array independently. The off-normal tilting reduces the effective solar constant as the cosine of the angle of incidence. The thermal design of each panel, and hence the design driver for packing factor, allows for steady-state survivability at any Sun angle at any mission solar distance. During nominal operation, if the spacecraft is at or near solar distances >0.55 AU, the solar arrays are maintained normal to the Sun. When the spacecraft solar distance is <0.55 AU, solar-array temperature management is accomplished by tilting each array to a predetermined off-normal Sun angle that was predicted by the power thermal model. This procedure ensures that an adequate power margin exists and that each array can be thermally controlled to temperatures that are similar to those experienced by Earth-orbiting spacecraft. Figure 14 shows the measured solar-array temperature performance and the predicted power output as a function of solar distance and Sun offset angle. To date, the solar arrays have been maintained at <130°C with no adverse power effects to the spacecraft.

The performance of select electronics and propulsion components is shown as a function of solar distance in Figures 15 and 16. The spacecraft electronics are mounted to the composite structure and except for radiators and apertures are completely covered with MLI. Complete spacecraft coverage with MLI, although attractive from the perspectives of heater power and coupled thermal mass, was not feasible, so electronics boxes dissipating, on average, >20 W have dedicated radiators. These radiators effectively keep the connected electronics boxes cool and, in conjunction with MLI heat leakage, help to keep the rest of the spacecraft at benign temperatures. The PSE and Inertial

Measurement Unit (IMU) are located on the –*X* side of the spacecraft, and the two Integrated Electronics Modules (IEMs), the Power Distribution Unit (PDU), and the Solar Array Junction Box (SAJB) are located on the +*X* side of the spacecraft. The ±*X*-side-panel radiator components rise ~10°C, due to thermal coupling to the solar arrays, as the solar constant changes one order of magnitude. The Advanced Star Tracker (AST), located on the –*Z* deck, is unaffected by solar distance. Because the propulsion tanks are always under positive heater control, the heater duty cycle decreases as solar distance decreases and/or spacecraft electronics dissipation increases, keeping the main fuel tanks at nearly constant temperature.

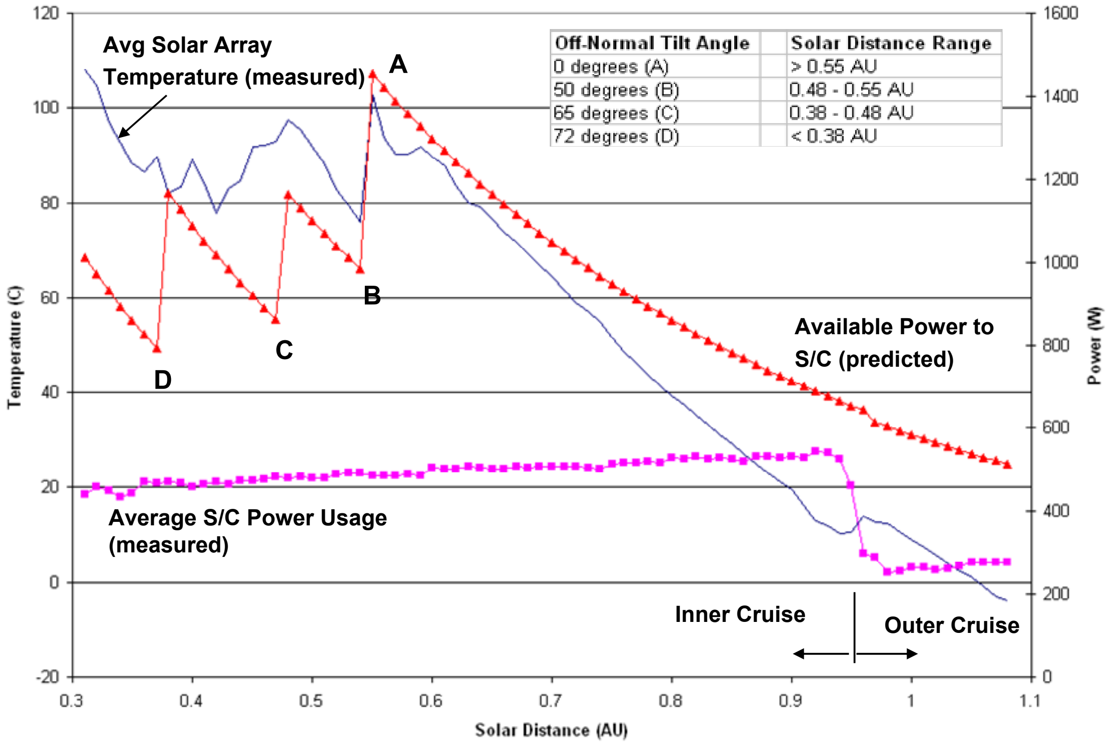

Figure 14. Measured solar-array power and temperature performance summarized as functions of solar distance. From the predicted available power, it is apparent that there is abundant power margin to accommodate transient peaks when loads, such as heaters, switch on and off. S/C, Spacecraft.

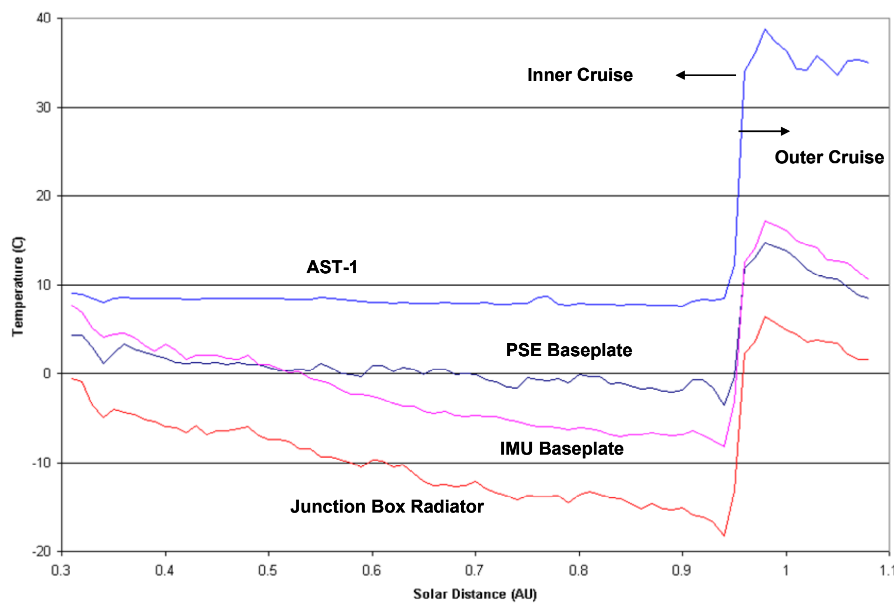

Figure 15. Temperatures of select electronics boxes controlled by external radiators that are located on the ±*X* and –*Z* sides of the spacecraft.

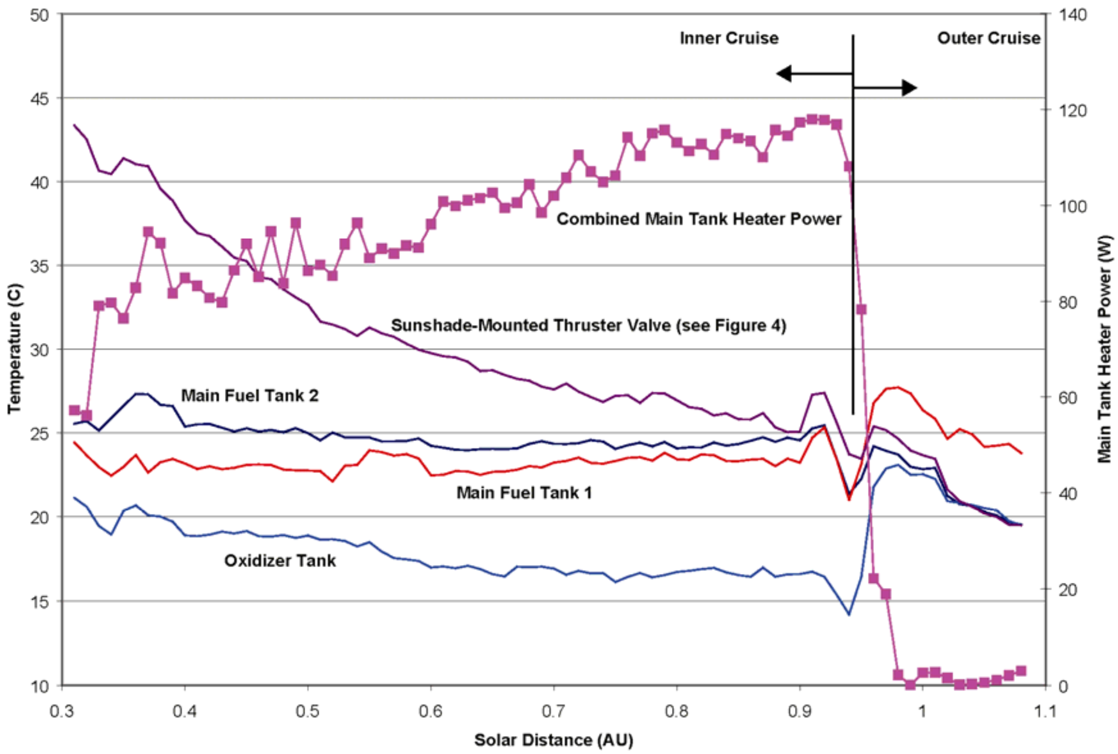

Figure 16. Temperatures of select propulsion-system components as functions of solar distance. Fuel tank 2 is located directly behind the sunshade, and fuel tank 1 is located farthest from the sunshade. The oxidized tank is located above the adapter in the center of the spacecraft. The Sun-pointing thrusters follow inverse-squared distance heating and mimic test results made during hot-case spacecraft thermal vacuum testing at GSFC. The thruster valve is radiation cooled using fuel tank 2 as the sink.

MERCURY FLYBY 1

On January 14, 2008, at 19:04:39 UTC (2:04:39 PM EST), the MESSENGER spacecraft flew ~200 km (124 miles) above Mercury's surface. This marked the first spacecraft encounter with Mercury in 33 years and the first encounter at a solar distance inside of 0.46 AU. Because thermal emissions from Mercury are a function of solar distance and position relative to the sub-solar point, spacecraft temperature response is sensitive to both the trajectory phase angle and distance from the planet surface. During Mercury flyby 1, the solar distance was 0.35 AU and the trajectory was such that the phase angle and altitude were never at minimum levels together. To achieve orbit, two more gravity-assist flybys of Mercury are necessary and are scheduled for October 6, 2008, and September 9, 2009. The thermal impacts on the spacecraft from Mercury during any of the flybys are relatively benign when compared with the environments expected during the orbital phase, as shown in Figure 17. Overall, the first Mercury Encounter went as expected from a spacecraft subsystem standpoint and was an outstanding success. All temperatures were benign, there were no power issues, the spacecraft was precisely pointed, and the science instruments collected information that was never seen before. Figure 18 summarizes the trajectory geometry during the flyby with the corresponding spacecraft temperature increase that was influenced by Mercury.

SUMMARY

Overall, the MESSENGER spacecraft thermal control design has performed very well, as illustrated by the data presented in this paper. A summary of minimum and maximum daily average component temperatures for both inner and outer cruise phases compared with the red and yellow flight limits currently stored onboard the spacecraft (Table 3) illustrates well this general assessment.

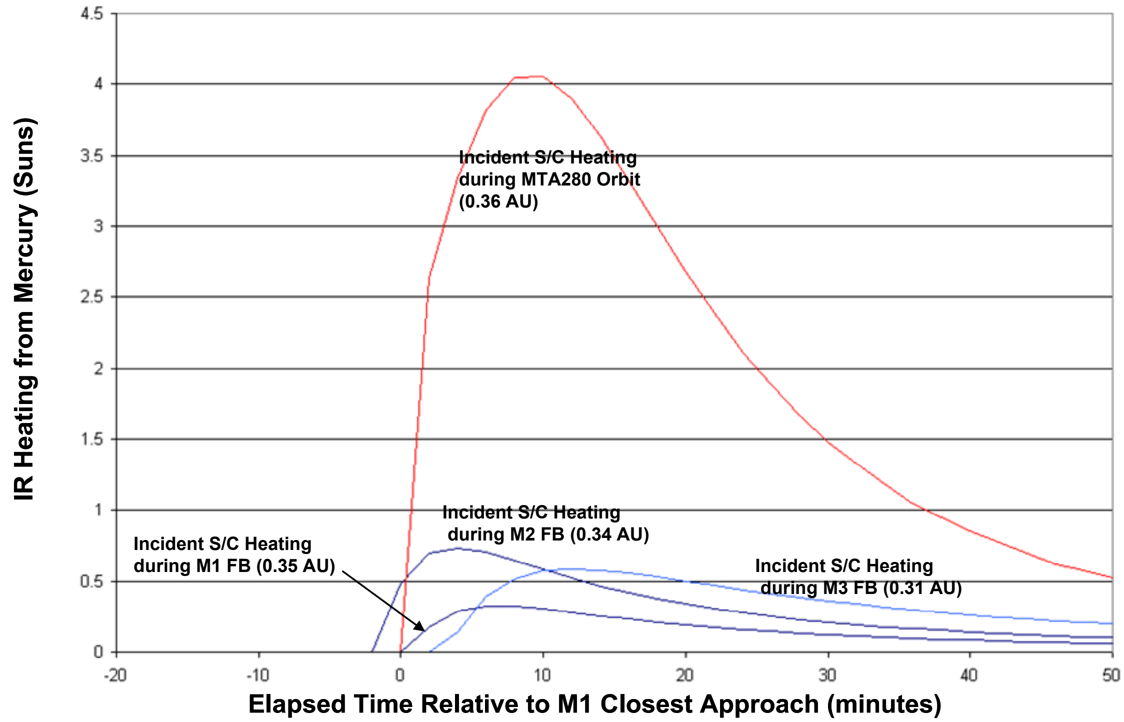

Figure 17. Comparison of the incident heating of the spacecraft from the three flybys with the incident spacecraft heating when in orbit about Mercury. Environments encountered during the orbital phase posed the most serious thermal design challenges. IR, Infrared; M1 FB, Mercury flyby 1; M2 FB, Mercury flyby 2; M3 FB, Mercury flyby 3; S/C, spacecraft.

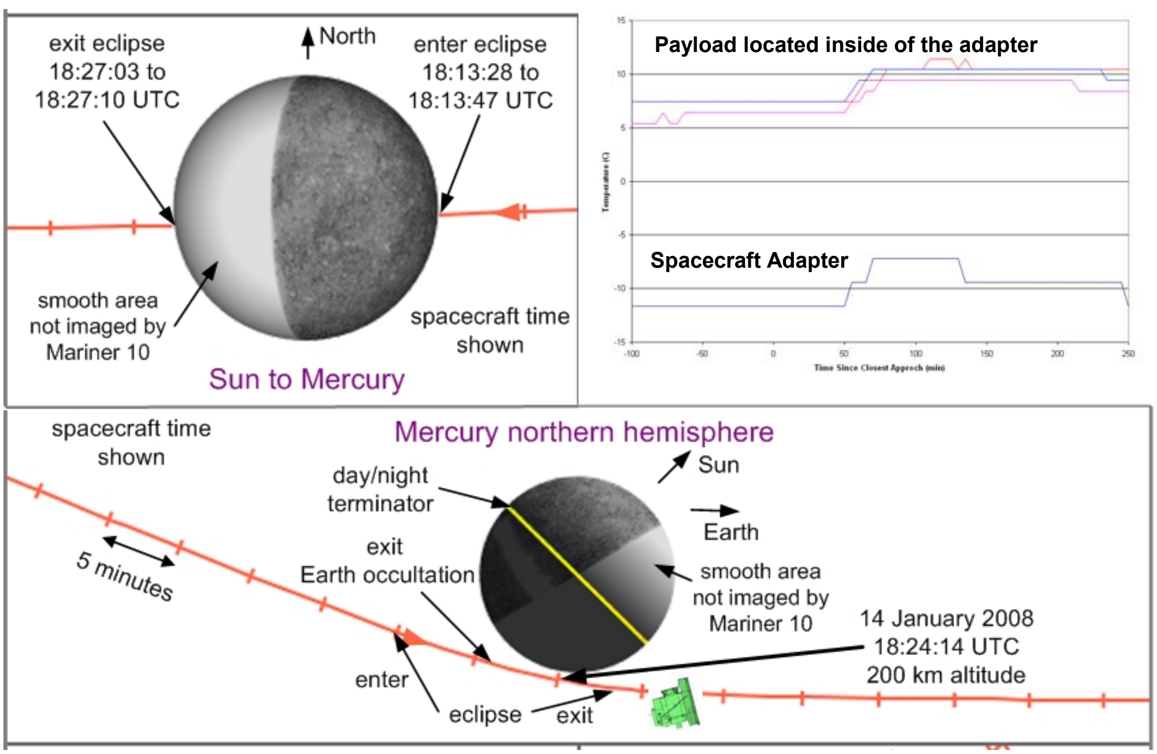

Figure 18. Mercury flyby 1 trajectory geometry. During the nearly equatorial flight path, the spacecraft maintained the adapter in a nadir orientation so as to optimize science observations with the adapter-located instruments. Infrared heating of the spacecraft began shortly after MESSENGER crossed the terminator and decayed as the distance to the planet increased as MESSENGER departed from Mercury. Onboard temperature measurements showed a peak temperature increase on the order of 10°C in the vicinity of the adapter, as shown by the temperature plot in the top right panel, as a result of heating from Mercury's hot surface as viewed by the spacecraft.

Table 3. Spacecraft component temperature performance over the mission to date.

	Flight Data System Temperature Limits (°C) (Red/Yellow)		(0.95–1.10 AU) Outer Cruise Temperature Performance (°C)		(0.31–0.95 AU) Inner Cruise Temperature Performance (°C)	
	Cold	**Hot**	**Cold**	**Hot**	**Cold**	**Hot**
Instruments						
MDIS	–48/–45	50/48	–38	–8	–40	–12
GRS	–30/–20	45/40	–15	34	–17	6
NS	–30/–27	45/40	–1	47	–25	–8
MLA	–20/–17	40/30	–13	15	–14	14
MASCS	–35/–30	50/45	–7	34	–24	12
EPS	–45/–40	45/40	–19	36	–38	–9
FIPS	–29/–24	60/55	5	33	–10	7
MXU	–25/–20	60/55	–11	23	–17	8
SAX	–55/–50	65/55	6	26	–40	12
Instrument Electronics Boxes						
MDIS/DPU Electronics	–29/–25	60/55	11	37	–10	18
GRS Electronics	–29/–25	60/55	15	47	–10	24
NS Electronics	–29/–25	60/55	5	54	–22	11
MLA Electronics	–29/–25	60/55	–7	34	–14	24
Attitude Determination and Control						
Reaction Wheels Assemblies	–15/–5	75/60	22	53	12	41
ASTs	–29/–24	50/45	14	56	–1	16
IMU	–24/–19	60/55	4	24	–9	8
Sunshade DSS Head (–*Y* Side)	–75/–65	190/140	39	100	–35	144
Non-Sunshade DSS Head (+*Y* Side)	–100/–95	190/140	0	57	–83	–4
DSS Electronics	–29/–24	60/55	17	36	4	23
Command and Data Handling						
IEM	–29/–24	60/55	–2	26	–14	4
Propulsion						
Oxidizer Tank	2/10	60/50	16	25	10	24
Main Fuel Tanks (Fuel Tanks 1 and 2)	2/10	60/50	18	32	14	31
Auxiliary Fuel Tank	2/10	60/50	15	25	17	26
Helium Tank	2/10	60/50	17	29	13	22
667 N Bi-Propellant Thruster Valves	2/10	100/70	19	23	21	29
22 N Mono-Prop Valves	2/10	100/70	17	29	23	32
4.4 N Mono-Prop Valves	2/10	100/70	18	40	22	30
4.4 N Mono-Prop Valves (Sunshade)	2/10	100/70	19	28	17	47
Power						
Plus-X/Minus-X Solar Panels	–145/–145	240/190	–4	22	9	125
PSE	–24/–14	65/55	2	23	–7	7
PDU	–24/–14	65/55	3	21	–9	11
SAJB	–24/–14	65/55	–1	15	–14	6
Battery	–10/–8	20/18	–5	1	–6	–1
SADAs	–29/–24	60/55	–2	26	–17	1
SADA Electronics	–29/–24	60/55	12	25	2	26

Table 3. ***(Continued.)***

	Flight Data System Temperature Limits (°C) (Red/Yellow)		(0.95–1.10 AU) Outer Cruise Temperature Performance (°C)		(0.31–0.95 AU) Inner Cruise Temperature Performance (°C)	
	Cold	**Hot**	**Cold**	**Hot**	**Cold**	**Hot**
RF Communications						
Front Low-Gain Antennas (–*Y*)	–90/–80	155/145	8	91	–55	46
Back Low-Gain Antennas (+*Y*)	–90/–80	155/145	–2	43	–74	14
Sunshade High-Gain Antenna Feed (–*Y*)	–90/–80	140/130	9	68	–49	89
Back High-Gain Antenna Feed (+*Y*)	–90/–80	140/130	4	42	–55	15
Transponder A/B	–30/–20	60/50	25	47	6	26
SSPA A/B	–29/–10	45/35	–7	26	–7	15

Red limits carry a minimum of 5°C margin above or below the listed hot or cold value. The maximum and minimum temperatures represent the integrated daily average for each mission day. EPS, Energetic Particle Spectrometer; FIPS, Fast Imaging Plasma Spectrometer; GRS, Gamma-Ray Spectrometer; MASCS, Mercury Atmospheric and Surface Composition Spectrometer; MDIS, Mercury Dual Imaging System; MLA, Mercury Laser Altimeter; MXU, Mercury X-Ray Unit; NS, Neutron Spectrometer; SADA, Solar-Array Drive Actuator; SAX, Solar Assembly for X-Rays.

ACKNOWLEDGMENTS

The MESSENGER mission is supported by the NASA Discovery Program under contracts to the CIW and APL. The author acknowledges Dr. Sean Solomon, the MESSENGER spacecraft Principal Investigator at CIW, and Mr. Peter Bedini, the MESSENGER spacecraft Project Manager at APL, for their support in the preparation and presentation of this paper. The author also acknowledges Mr. Sean Laughery for his help in processing the flight data used in this paper.

REFERENCES

1. C. L. Yen, "Ballistic Mercury Orbiter Mission via Venus and Mercury Gravity Assist," *J. Astronaut. Sci.*, 37, 417–432, 1989.
2. A. G. Santo *et al.*, "The MESSENGER Mission to Mercury: Spacecraft and Mission Design," *Planet. Space Sci.*, 49, 1481–1500, 2001.
3. C. J. Ercol and A. G. Santo, "Determination of Optimum Thermal Phase Angles at Mercury Perihelion for an Orbiting Spacecraft," 29th International Conference on Environmental Systems, Society of Automotive Engineers, Tech. Paper Ser., 1999-01-21123, Denver, CO, July 21–25, 1999.
4. C. J. Ercol *et al.*, "Prototype Solar Panel Development and Testing for a Mercury Orbiter Spacecraft," 35th Intersociety Energy Conversion Engineering Conference, American Institute of Aeronautics and Astronautics, Paper AIAA-2000-2881, Las Vegas, NV, July 24–28, 2000.
5. C. J. Ercol, "MESSENGER Heritage: High-Temperature Technologies for Spacecraft to the Inner Solar System," American Institute of Aeronautics and Astronautics Space 2007 Conference, Paper AIAA-2007-6188, Long Beach, CA, September 18–20, 2007.
6. P. D. Wienhold and D. F. Persons, "The Development of High-Temperature Composite Solar Array Substrate Panels for the MESSENGER Spacecraft," *SAMPE J.*, 39 (6), 6–17, 2003.
7. R. E. Wallis, J. R. Bruzzi, and P. M. Malouf, "Testing of the MESSENGER Spacecraft Phased-Array Antenna," Antenna Measurement Techniques Association 26th Meeting and Symposium, pp. 331–336, Stone Mountain, GA, October 2004.
8. C. J. Ercol *et al.*, "Power Subsystem Thermal Design and Early Mission Performance," 4th International Energy Conversion Engineering Conference and Exhibit, Paper AIAA-4144, San Diego, CA, June 26–29, 2006.
9. R. L. Vaughn *et al.*, "Return to Mercury: The MESSENGER Spacecraft and Mission," Institute of Electrical and Electronics Engineers Aerospace Conference, IEEEAC Paper 1562, Big Sky, MT, March 4–11, 2006.

CONTACT INFORMATION

To whom correspondence should be addressed: Carl J. Ercol, Lead MESSENGER Thermal Engineer, Space Department, The Johns Hopkins University Applied Physics Laboratory, 11100 Johns Hopkins Road, Laurel, MD 20723-6099.