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The MESSENGER Spacecraft Power Subsystem Thermal Design and Early Mission Performance

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The MErcury Surface, Space ENvironment, GEochemistry, and Ranging (MESSENGER) spacecraft, launched on August 3, 2004, is a NASA spacecraft that will orbit the planet Mercury for a one-year mission. The spacecraft launch mass limitation, combined with the large solar distance variations, impose severe requirements on the spacecraft power and thermal subsystems. The spacecraft is three-axis stabilized. A sunshade protects the spacecraft from the high intensity solar flux. The attitude control maintains the sunshade pointed to the Sun. The solar panels, which are outside the thermal shield, are designed to survive normal Sun incidence at 0.3 AU. The solar panels consist of alternating rows of triple junction cells placed between Optical Solar Reflectors (OSRs). Solar panels thermal control is performed by tilting the panels with increasing solar flux. To minimize the mass of the spacecraft, the structure is made of composite materials. Spacecraft electronic boxes that are high power dissipaters are designed with special thermal vias that conduct the heat directly to diode heat pipes, which transport the heat of the box to thermal radiator panels on the side of the spacecraft behind the sunshade. The MESSENGER spacecraft is on a trajectory to enter Mercury orbit in 2011. The spacecraft is performing as designed.

I. Mission Description

MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging), shown in Fig. 1, is a NASA Discovery Program spacecraft designed and built by The Johns Hopkins University Applied Physics Laboratory (APL). It will orbit the planet Mercury for one Earth year. During the orbital phase of the mission, MESSENGER will complete global mapping and the detailed characterization of the exosphere, magnetosphere, surface, and planet interior [1]. The instruments consist of the following: The Mercury Dual Imaging System (MDIS) is a narrow-angle imager and a wide-angle multispectral imager that map landforms and surface spectral variations. The Mercury

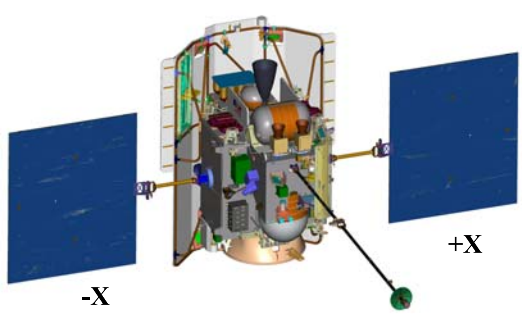

Figure 1. The MESSENGER Spacecraft.

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Laser Altimeter (MLA) produces measurements of surface topography. The Gamma-Ray and Neutron Spectrometer (GRNS) maps elemental abundances in Mercury's crust. The Neutron Spectrometer sensor on GRNS provides hydrogen sensitivity in ices at the poles. The Mercury Atmospheric and Surface Composition Spectrometer (MASCS) includes an ultraviolet-visible spectrometer that measures abundances of atmospheric gases and a visible-infrared spectrograph that detects minerals in surface materials. The Magnetometer (MAG), attached to a 3.6-m boom, maps Mercury's magnetic field and searches for regions of magnetized crystal rocks. The X-Ray Spectrometer (XRS) maps elemental abundances of crustal materials. The Energetic Particle and Plasma Spectrometer (EPPS) measures the composition, spatial distribution, energy, and time-variability of charged particles within and surrounding Mercury's magnetosphere.

MESSENGER was launched on a Delta 7925H-9.5 launch vehicle on August 3, 2004. The spacecraft trajectory requires two gravity-assist flybys from Venus and three from Mercury. The spacecraft-to-Sun distance varies between 1.075 and 0.3 Astronomical Units (AU). The solar illumination, which varies inversely with the square of the Sun distance, has a severe impact on the thermal design of the solar array panels and spacecraft.

II. Power and Thermal Design Overview

A sunshade protects the spacecraft from the high intensity solar illumination. The spacecraft is kept behind the sunshade, made of 3M Co. Nextel ceramic cloth material. The spacecraft attitude control points the sunshade toward the Sun at all times when the spacecraft-Sun distance is less than approximately 0.95 AU. To reduce the spacecraft and instrument heater-power requirements in the mission, the spacecraft is turned around such that the spacecraft body is pointed toward the Sun, thus increasing the solar flux on the body of the spacecraft and reducing the thermal power requirements. The spacecraft is flipped, pointing the sunshade toward the Sun, at approximately 0.95 AU to protect the spacecraft from overheating. The solar panels, which are outside the sunshade, are designed to survive solar illumination at normal incidence at the 0.3-AU mission perihelion.

The large solar distance variations impose severe requirements on the solar array design. The operational solar array maximum-power-point voltage is expected to vary between 45 and 100 V. A Peak Power Tracker (PPT) topology with strong heritage to the APL-designed Thermosphere, Ionosphere, Mesosphere, Energetics and Dynamics (TIMED) spacecraft power system was selected [2]. This architecture isolates the battery and the power bus from the variations of the solar-array voltage and current characteristics and maximizes the solar-array power output over the highly varied solar-array operating conditions of the mission.

The MESSENGER power system consists of the Power System Electronics (PSE), the Power Distribution Unit (PDU), the Solar Array Junction Box (SAJB), the battery, the two solar array (S/A) panels, and the Solar Array Drive Assembly (SADA). The spacecraft loads are connected directly to the single 22-cell, 23-Ah nickel hydrogen (NiH_2) battery, in which every two cells are in one Common Pressure Vessel (CPV). The nominal bus voltage is 28 V and can vary between 22 and 35 V depending on the state of charge of the battery. A simplified block diagram of the power system is shown in Fig. 2.

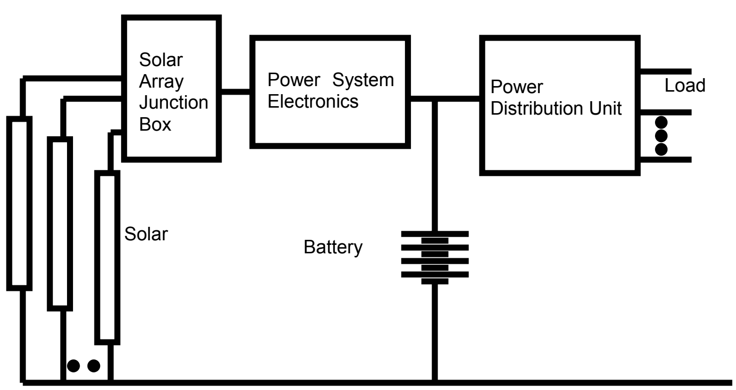

Figure 2. Simplified Power System Block Diagram

Triple-junction solar cells with efficiencies of 28% were used on the solar array. The solar cell strings were placed between Optical Solar Reflector (OSR) mirrors with a cell-to-OSR ratio of 1:2 to reduce the panel absorbance.

The average spacecraft load power during flight, with the sunshade pointed in the anti-Sun direction, is 280 W. The load with the sunshade pointed to the Sun is 550 W. The orbital load during Sun time is 595 W.

Solar panel thermal control is performed by tilting the panels away from normal incidence with increased solar intensity. In case of an attitude control anomaly near Mercury, the solar array temperature may reach 275°C. All material and processes used in the solar panels were designed and tested to survive the worst-case predicted temperatures. The array strings are isolated with decoupling diodes that are placed in the Solar Array Junction Box inside the spacecraft to protect them from the expected high temperatures. The two solar panels are maintained normal to the Sun until the panel temperature reaches a preset value (maximum 150°C). The panels are rotated by the SADA to limit the temperature to the preset value but still provide the required spacecraft power. The panels are rotated toward normal incidence as the panel temperature drops below the limit value and more power is required. The two S/A wings are rotated to the same incident Sun-angles so that they will operate at the same temperature. One peak power tracker with eight converter modules is used.

To minimize the mass of the spacecraft, the structure is made of graphite-composite materials. However, due to the poor thermal conductivity of the composite material, it became necessary to modify the thermal design of the boxes that are high power dissipaters in order to eliminate issues of heat transport to the radiating plate. MESSENGER power system electronic boxes are designed with special thermal vias that conduct box waste heat directly into diode heat pipes that are connected to radiator panels oriented orthogonal to the direction of the Sun. Figure 3 shows the typical interface between a power system box and the spacecraft.

In addition to the thermal design implication of the composite structure, electrically conductive tapes were added for electrical grounding and electromagnetic interference (EMI) control due to the low electrical conductivity of the composite structure material.

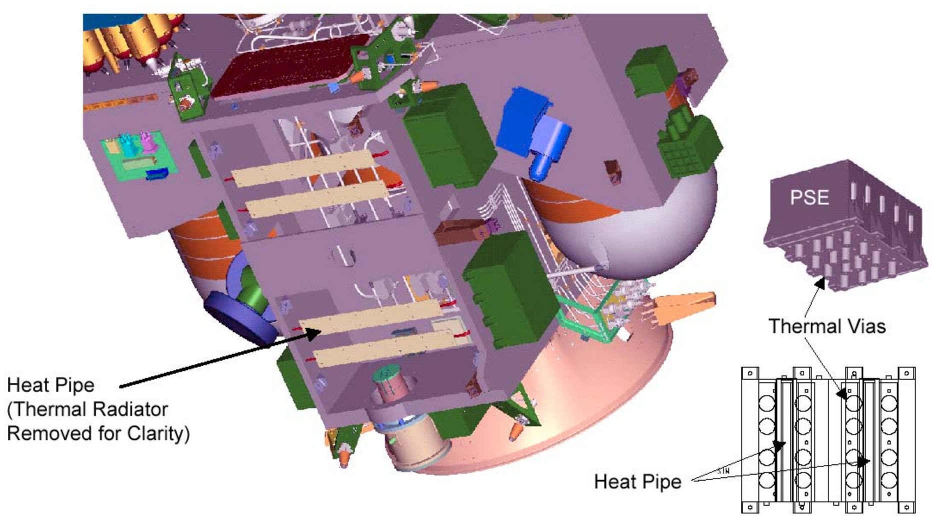

Figure 3. Typical Thermal Interface for a Power Box

III. The Sunshade

MESSENGER has a low-risk passive thermal control design that is dominated by the sunshade. The sunshade is made of aluminized Nextel 312-AF 10 ceramic cloth and multi-layer aluminized Kapton. The sunshade will be heated to a maximum temperature of 325°C when at Mercury perihelion. However, the sunshade creates a benign thermal environment for the main spacecraft bus, allowing the use of standard electronics, components, and thermal blanketing materials. Extensive testing was performed on the sunshade material including long duration 11-Sun exposure at NASA Glenn Research Center [3]. During spacecraft thermal vacuum testing the “Sun-side” of the sunshade was soaked at 355°C for an accumulated 18 days (~432 hours) for different hot spacecraft test phases. All the spacecraft component temperatures remained within typical spacecraft limits. Figure 4 shows MESSENGER behind the sunshade just prior to launch.

Figure 4. MESSENGER Sunshade

The sunshade structure performs two functions: (1) It supports the high-temperature multi-layer insulation (MLI) blanket that protects and insulates the core spacecraft from direct solar radiation when inside of 0.95 AU, and (2) it supports active components consisting of four Digital Sun sensors, one of the phased-array and three low-gain antenna assemblies, one X-ray solar monitor, and two thrusters. These components, except for the phased-array antenna, are “off-the-shelf” technologies that were modified by the specialized cost effective thermal designs. Standard Adcole Digital Sun sensors were modified with an enlarged housing to improve heat rejection. High-temperature adhesives, similar to those used on MESSENGER solar arrays, replaced the lower-temperature adhesives used for their standard designs. But the key new thermal design feature of the Sun sensor is a special solar attenuating filter that is placed over each head to reduce the incident solar intensity by one order of magnitude. All Sun-facing antennas are covered with custom-made high-temperature and RF-transparent Nextel radomes. The thermal control of the solar panels that must be exposed to the solar illumination is provided by the chosen ratio of OSRs to solar cells and appropriate rotation of the panels relative to the Sun line as described in the solar array section. All of these Sun–illuminated components are thermally isolated from themselves and the rest of the spacecraft.

During certain orbits about Mercury, the spacecraft will be between the Sun and the hot planet for approximately 30 minutes. During this period the sunshade will protect the Sun-facing side of the spacecraft but the unshaded back of the spacecraft will be in direct view of 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 the effects of direct radiation from the thermal environment. Planet-viewing instruments such as MDIS required a very specialized thermal design to allow full operation during this hot transient period. Diode heat pipes are employed in both the spacecraft and imager thermal designs to protect attached components when radiator surfaces are exposed to the thermal radiation emitted by Mercury. The diode heat pipes effectively stop conducting when the radiator surface begins to get hot, and return to conduction when the radiator surface cools restoring normal thermal control. The PSE, PDU, and SAJB are all temperature controlled via diode heat pipes connected to aluminum honeycomb radiator panels that are located on the +/- X sides of the spacecraft. Figure 5 demonstrates, with temperature data taken during a planet crossing simulation performed during spacecraft thermal vacuum (TV) testing, the effectiveness of the diode heat pipe design as applied to the PSE.

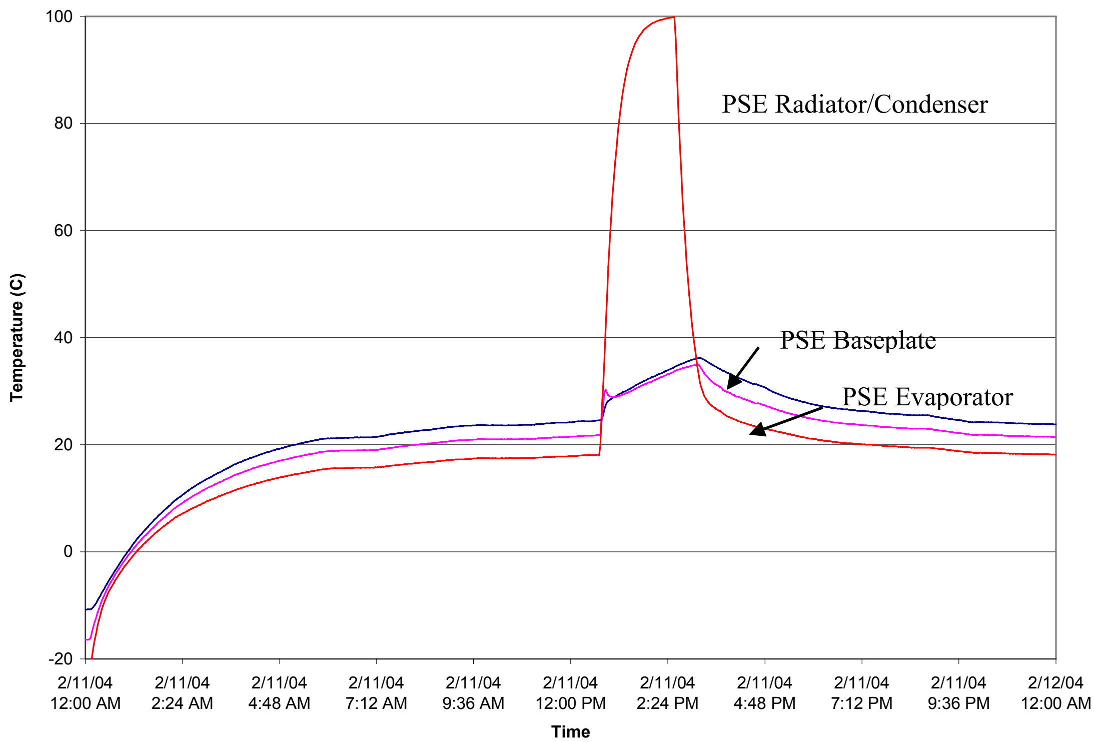

Figure 5. PSE Orbital Simulation during Spacecraft Thermal Vacuum Testing

Special emphasis in the thermal design of the spacecraft components was placed on the extreme heat and high temperatures associated with an orbit about Mercury. This emphasis led to high heater power demand during the portions of the spacecraft trajectory when the sunshade is pointed to the Sun but there is no heat from Mercury's surface. In order to reduce heater power consumption and allow the use of a smaller size solar array while maintaining good solar array power margin, a mission critical design feature of MESSENGER is the ability of the spacecraft to be “flipped” such that the anti-Sun side can be illuminated (configured to fly with the sunshade pointed away from the Sun). This capability has allowed MESSENGER to operate easily between 0.95 and 1.08 AU and allowed unconstrained outer solar distance flexibility when the mission design team was planning for back-up missions that introduced different mission trajectories and outer solar distance excursions without complicating the spacecraft thermal or power designs.

IV. Solar Array

The MESSENGER solar array consists of two deployed single-panel wings. Each panel is 1.54 m wide and 1.75 m long. The panel substrates are 18-mm-thick aluminum honeycomb with RS-3/K13C2U composite face sheets. The face sheets are 0.6 mm thick with local 0.5-mm doublers. The graphite-cyanate-ester materials on the panel face sheets were chosen for their high thermal conductivity, but their mechanical strength is relatively low. Doublers and triplers are therefore required in areas of high stress due to the large panel cantilever in the stowed configuration. The panels articulate using a single-axis solar array drive actuator.

The panel front cell side is insulated with 0.05-mm Kapton, co-cured with the graphite fiber face sheet. The back face sheet is covered with co-cured square-shaped aluminized Kapton in 30.5 cm x 30.5 cm sheets. The aluminized Kapton is used to lower the absorbance of the backside of the solar panel to a level comparable to the solar cell-OSR side. This ensures that the solar panels can survive solar illumination with normal incidence to either side at the closest approach to the Sun.

The solar cells are 0.14-mm-thick, 3 cm by 4 cm triple-junction cells with a minimum efficiency of 28%, from EMCORE Corporation. The cell lay-down was carried out by Northrop Grumman Space Technology (NGST).

All electrical interconnections, including cell repairs, are welded. High-temperature wire is used. The staking of the wires is done with CV 1142-2 RTV. The wires are routed along the titanium boom to connectors at the SADAs. Both the wires and the boom are wrapped with multi-layer insulation. The solar panel temperatures are sensed using platinum wire sensors, PT103, placed beneath the solar-cell-side face sheet in small bored cavities.

To demonstrate the survivability and validate the thermal analysis [3,4], qualification panels successfully completed a series of high-temperature tests including infrared heater and high-intensity illuminated high-temperature tests in vacuum at the Tank 6 facility at NASA Glenn Research Center (GRC).

Test panels with solar cells and OSRs were fabricated by four U.S. cell lay-down manufacturers. The panels were tested in vacuum over a temperature range from -130°C to 270°C at APL. They were tested at GRC at 11-Sun intensity illumination. The panels were also life cycled successfully over the range from -130°C to 150°C in a nitrogen environment at the Aerospace Corporation in Los Angeles. The rate of change in temperature was 80 to 100°C/minute to simulate the thermal shocks expected at Mercury eclipse exit.

A flight qualification panel was built and tested. It was cycled in vacuum over the range -140 to +275°C. a special infrared (IR) coupled thermal box, named the H-Box and shown in Figure 6, was built at APL to control accurately and uniformly the temperature of the flight panels during the high-temperature TV chamber testing. The H-Box is capable of testing two MESSENGER size flight solar panels at one time. The box can achieve -150 to +330°C and contains liquid nitrogen (LN_2) cooling. It is combined with heaters for accurate solar panel temperature control. The panel cooling in the H-Box is independent of the TV chamber cold wall. The peak power required by the H-Box heater is around 50 KW.

The flight panels were cycled in vacuum in the H-Box over the -140°C to 240°C maximum operational temperature range. The flight panels were not exposed to temperatures over the substrate materials' glassing temperatures, above 240°C, due to concerns over possible weakening of the substrate mechanical strength required for launch.

Figure 6. H-Box Used for High-temperature Tests of the Solar Panels

The mass of the two solar array panels is 34.12 kg. Figure 7 shows the two panels mounted on the spacecraft in the launch configuration.

Figure 7. MESSENGER Spacecraft with Solar Panels in Launch Configuration

V. Battery

The MESSENGER spacecraft battery was designed and built at APL. The nickel hydrogen CPV cells were manufactured by Eagle Picher Space Energy Products Division. The 11-cell CPV cells are housed in a compact three-piece machined aluminum frame that is a clam-shell-type design. The battery frame is thermally isolated from the spacecraft deck and conductively coupled to a space-facing silver-Teflon covered radiator that has an area of approximately 0.13 m^2. Spacecraft structural requirements dictated the frame wall thicknesses, and corresponding thermal analysis verified, as did battery thermal vacuum testing, that this design would meet the maximum cell-to-cell gradient requirement of < 3.0°C.

The thermal design of the battery maintains operation between -5 and +10ºC during most of the mission. However, the temperature may reach 20ºC during discharge in the sub-solar crossing of certain Mercury orbits. Originally packaged on the top deck and farthest from the sunshade, at the current location of the helium tank, thermal simulation of orbit phase operation showed that the battery transient temperature excursion during sub-solar and near sub-solar orbits would exceed the battery's maximum allowable temperature limit. As a direct result of this thermal analysis the battery and helium tank were switched, and the battery was packaged directly behind the sunshade and above main fuel tank #2, as shown in Figure 8. This location minimized planetary heating when in orbit but still allowed for battery temperature maintenance to be dominated by the thermostatically controlled primary heater circuit, as evident by the battery temperature measurements since launch shown in Figure 9. An interesting artifact of Figure 9 is the battery temperature as a function of solar distance. Between mission elapsed time (MET) day ~400 and MET day ~500 the solar distance decreased from 0.95 to 0.61 AU. During this time the battery temperature rose due to the residual heating from the back of the sunshade in conjunction with the approximately 30 W of battery heater power and about 6 W of battery trickle charge. Calculations based on a quasi-steady state battery temperature indicate that before the heater switched off on MET day 485 the input to the battery radiator from the sunshade was on the order of 4 W. Since the minimum mission solar distance occurs when at Mercury perihelion, 0.3 AU, the worst case sunshade heating environment introduced to the battery is expected to be less than 15 W based upon the flight data analyzed during the 0.61-AU case discussed. Since the battery is using around 30 W of heater power, the solar effect on the sunshade due to Mercury perihelion would cause the battery heater to operate at a 50% (instead of 100% at 1 AU) duty cycle while maintaining the battery between -5 and 0°C, indicating good heater power and temperature margins for the battery.

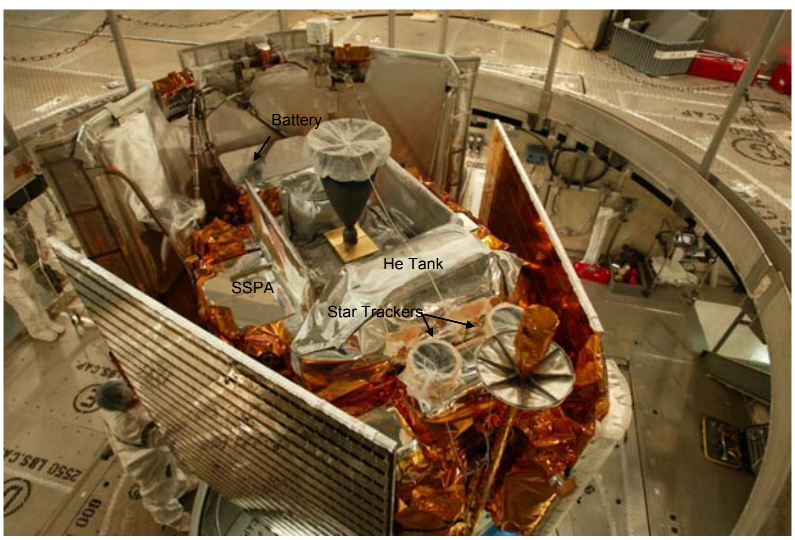

Figure 8. MESSENGER Battery and Top Deck Flight Configuration

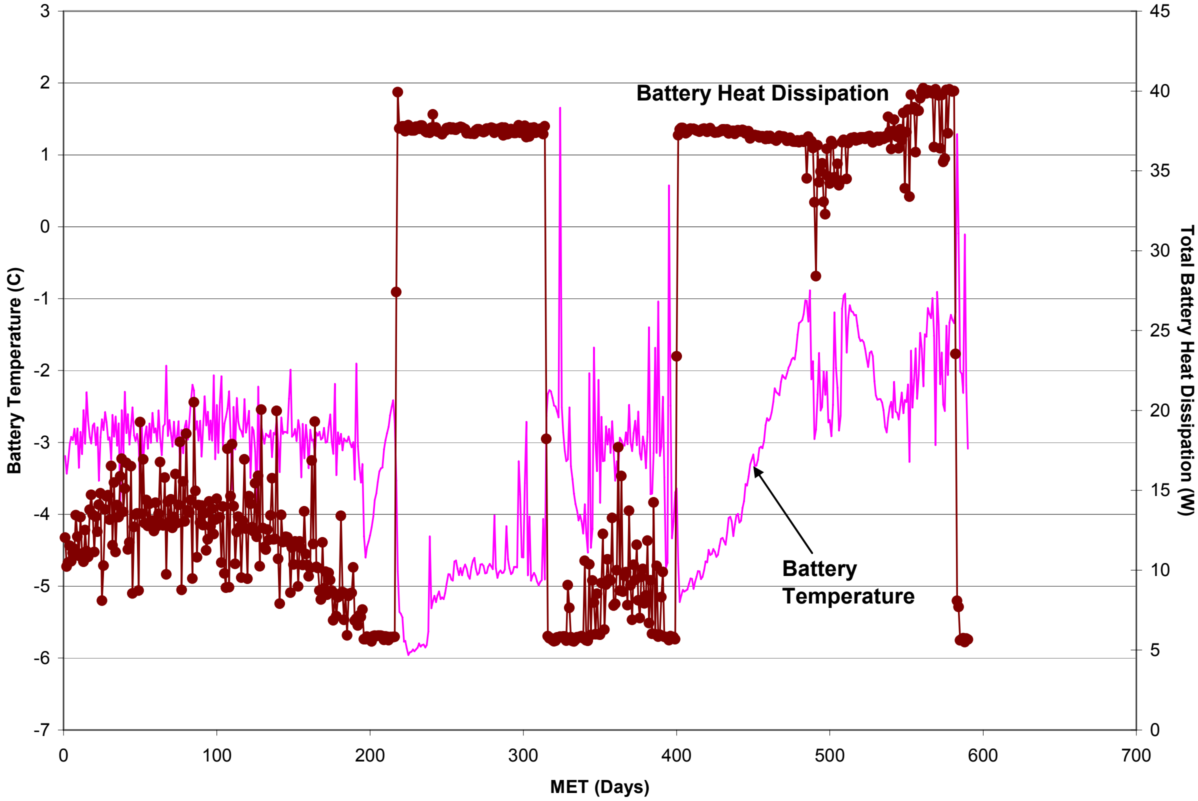

Figure 9. In-flight Battery Heat Dissipation and Temperature

The battery assembly includes three primary and three redundant temperature sensors on the battery cells as well as primary and redundant radiator plate sensors. The battery is also fitted with thermostatically controlled primary and back-up dual trace heater circuits that are mounted to the radiator plate and covered with silver–Teflon. The exploded view showing the mechanical design detail and a flight battery photo are shown in Figure 10. The battery mass is 24.5 kg (without the extra thermal plate mass needed to support the August 2004 launch), and the size is 36.6 cm x 22.83 cm x 49.17 cm.

Figure 10. MESSENGER Flight Battery with an Exploded-view Drawing

VI. Power System Electronics

The PSE contains eight buck-type peak-power-tracking converter modules, each designed to process around 100 W of output power. Each card can dissipate about 8 W. The control loop of the buck converters varies the duty cycle to maintain the input voltage from the solar array wings to the reference value set by the spacecraft command and data handling (C&DH) processor. The peak power tracking converters process all the power from the solar array. The maximum power the PPT trackers can process is around 800 W. The primary and backup controller circuits of the PPT converters and the housekeeping DC/DC converter are on two printed circuit (PC) cards. The maximum PSE power dissipation is 65 W.

The PSE is packaged in slices with one card per slice. There are eight PPT converter slices and two controller board slices. The slices are machined aluminum 6061-T6, in a tongue-and-groove design with venting through joints. The PC boards are mounted to the slice frames at eight places and staked to the frame with Scotchweld 2216 B/A Gray for structural and thermal support. Six titanium bolts passing through the entire unit hold the ten slices together. An exploded view of the PSE is shown in Figure 11.

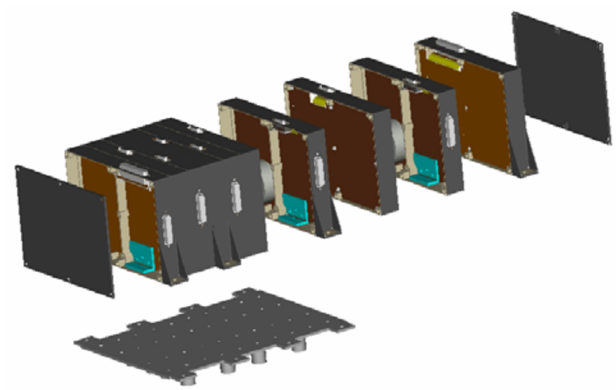

Figure 11 . Exploded View of the PSE

A heat spreader with thermal vias distributes heat dissipations among the slices and provides thermal interface from chassis to heat pipes that are located underneath the spacecraft deck. It is made of machined aluminum 6061-T6, 0.32 cm thick. There are sixteen thermal vias that are 1.9 cm in diameter and 2.86 cm long. A 0.25-mm CHO-SEAL 1224 gasket provides a thermal and electrical interface to the individual slices.

Due to the coefficient of thermal expansion (CTE) mismatch between the aluminum chassis and the composite spacecraft deck, a slip joint with hard chrome washers is employed at the mounting feet to mitigate the thermal-induced stresses on the deck inserts. Figure 12 provides details of the slip joint design. Tests were conducted to determine proper torque values for nickel-plated washers. The same slip joint design is used on the SAJB and the PDU.

The PSE dimensions are 29 cm x 20 cm x 17.2 cm (with the thermal vias), and the mass is 8.55 kg.

Thermal analysis and modeling were performed using the COSMOS Version 2.6 software package. The model used 6604 shell elements and 70 truss elements. Analysis results indicated that all components are well within their allowable operating temperatures for a base plate temperature at 70°C that is well within the predicted values.

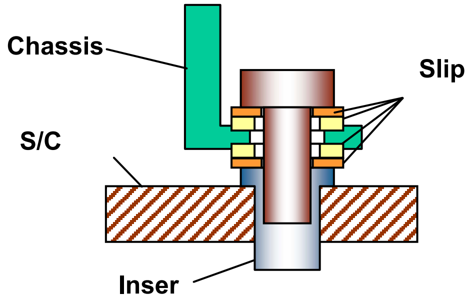

Figure 12. Typical Slip Joint Used on the PSE, SAJB, and PDU

VII. Solar Array Junction Box

The SAJB (Figure 13) has one PC board assembly that contains isolation diodes configured in series with each string of the two solar array panels, the current shunt resistor of each wing, the PPT module solar array side fuses, and solar array voltage telemetry buffer resistors. The maximum dissipation in the solar array diodes under worst-case conditions is 26 W. The PC board design includes eight layers with components populated on both sides. The board is mounted to a machined 0.125-cm aluminum (6061-T6) housing in twelve places. High-thermal-dissipation components are bonded to an aluminum heat sink that is bonded to the bottom side of the board to couple the heat load directly to the base plate.

Thermal analysis and modeling were carried out similarly to the PSE design. All components meet the NASA derating guidelines under the worst-case condition for a base plate temperature at 75°C.

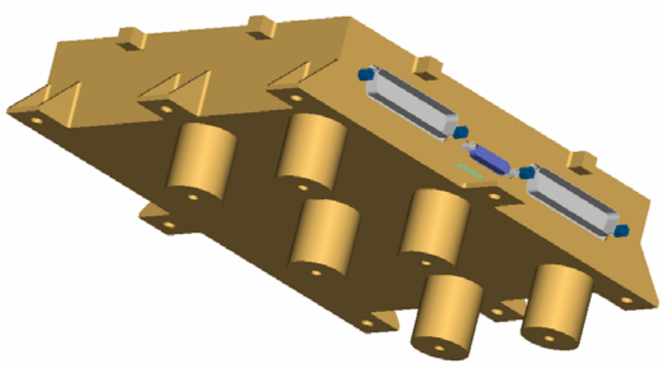

Figure 13 . SAJB Mechanical Design

The SAJB dimensions are 16.9 cm x 25.1 cm x 6.38 cm (with the thermal vias), and the mass is 1.48 kg.

VIII. Power Distribution Unit

The Power Distribution Unit (Figure 14) contains the circuitry for the spacecraft pyrotechnic firing control, power distribution switching, load current and voltage monitoring, fuses, external relay switching, reaction wheel relay selects, power system relays, Inertial Measurement Unit (IMU) reconfiguration relays, Integrated Electronics Module (IEM) select relays, solar array drives, propulsion thruster firing control, and propulsion latch valve control. There are two sides to the PDU: A and B. Telemetry is collected by the PDU.

Because of mass limitations for the mission a substantial effort was spent in reducing the size and mass of the PDU [5]. The PDU is packaged in slices with a PC board mounted in each slice. There are 10 slices in the PDU design. In addition to the slices, there are two compartments located on top of the PDU for the fuse cards that plug into a motherboard. The slices are machined magnesium alloy ZK60-A, in a tongue-and-groove design with venting through joints. The PC boards are mounted to the slice frames at eight places and staked along the edges with Scotchweld 2216 B/A Gray for structural and thermal support.

The PDU thermal control is a passive thermal conduction design where each electronic slice has an independent thermal path to the heat sink. The PDU is designed to operate with a base-plate temperature that varies from -34°C to +65°C. It dissipates 26 W in the worst-case condition with both primary and redundant sides ON. Except for the command decoder/DC-DC converter board, all PDU circuit boards rely on internal copper layers to transfer heat generated from the electronic components to the heat sink. The command decoder/DC-DC converter board requires an additional 1-mm-thick magnesium heat sink integrated into the chassis design. To enhance the thermal transfer between the slices and the heat spreader, a layer of Choseal 1285, 0.5-mm-thick gasket, is added between the electronic modules and the heat spreader. The heat spreader also has thermal vias that connect the chassis to the spacecraft heat pipes located beneath the structural decks. The PDU dimensions are 23.4 cm x 22.6 cm x 35.2 cm (with the thermal vias), and the mass is 12.81 kg.

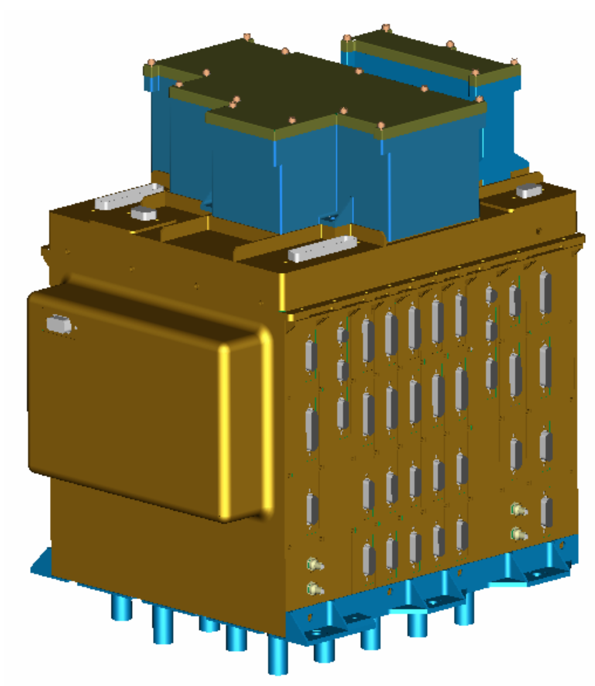

Figure 14 . PDU Mechanical Design

IX. Fight Performance

The spacecraft is performing as designed. Figure 15 shows the predicted solar panel power available to the spacecraft loads and the measured flight power since launch. The step changes indicate the points where the spacecraft was flipped to point the sunshade toward or away from the Sun. The plots show the effectiveness of spacecraft flipping in the power management during the mission so far. Figure 16 compares predicted solar array temperature with the temperatures measured in flight. To date the solar panels have been maintained almost normal to the Sun. The maximum temperature of the panels has been less than 90˚C. The battery heaters, under thermostatic switch operation, are maintaining the battery between -6˚C and -1˚C. Figure 17 shows the measured PSE, PDU, and SAJB temperatures since launch along with box base plate and heat pipe radiator temperatures. The PSE, Junction Box, and PDU temperatures are well within their predicted values.

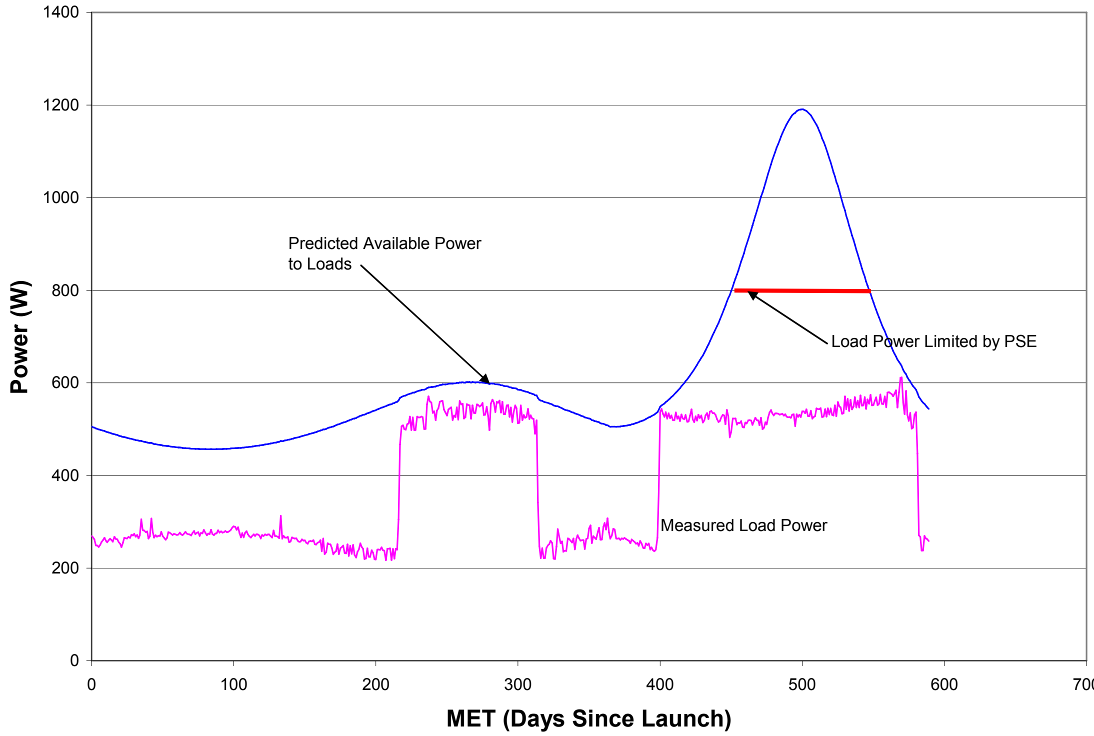

Figure 15. Predicted and Measured In-flight Solar Panel Power

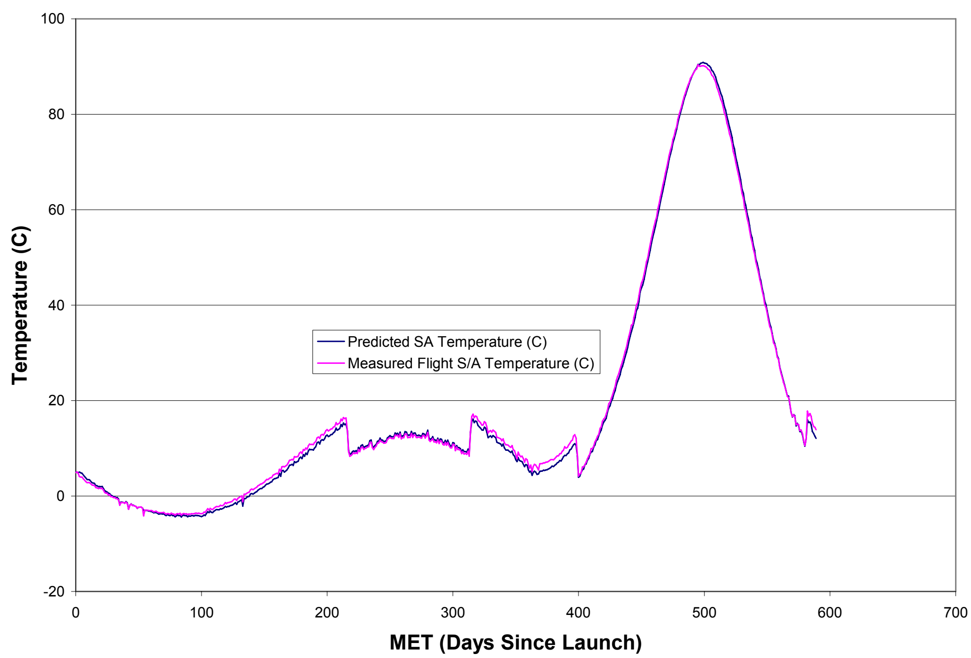

Figure 16. Predicted and Measured In-flight Solar Array Temperature

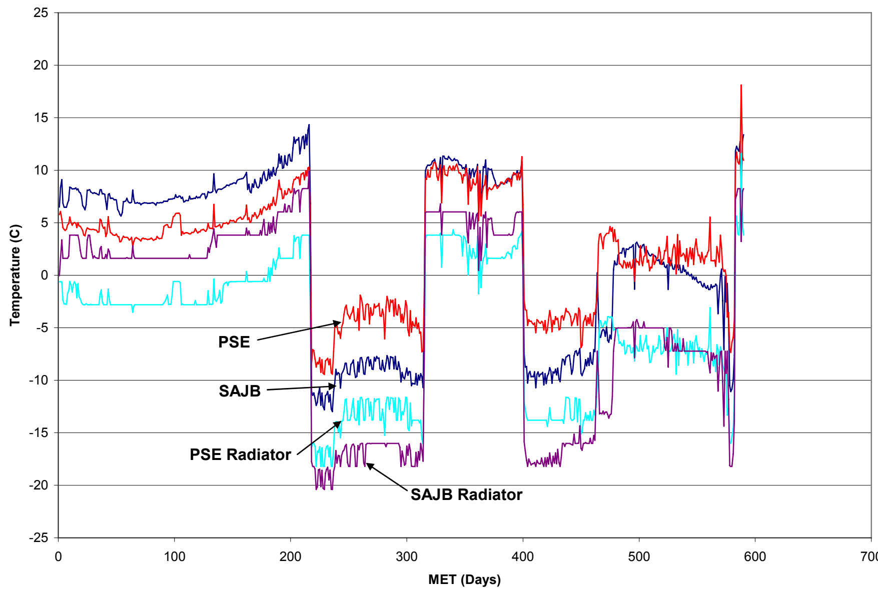

Figure 17. Measured PSE and SAJB Temperatures since Launch, including Box Base Plate and Heat Pipe Radiator Temperatures

SUMMARY

The power and thermal systems for a spacecraft mission to orbit Mercury are challenging. The designs presented meet the MESSENGER mission requirements. The spacecraft and the power system have been operating as designed since launch on August 3, 2004. Extensive solar array development and testing were conducted to characterize and qualify the design. The solar array design approach should be directly applicable to spacecraft missions that require high-temperature operation. Packaging and thermal design of the electronics were effective for system boxes with high power dissipation. The spacecraft composite structure required to minimize the spacecraft mass generated complexity for the high dissipater boxes and for the EMI control. The issues were resolved successfully. The spacecraft continues its trajectory for a Mercury orbit insertion in 2011.

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REFERENCES

[1]Solomon, S. C., et al. "The MESSENGER Mission to Mercury: Scientific Objectives and Implementation," *Planetary and Space Science*, vol. 49, No. 14, 2001, pp. 1445-1465.
[2]Dakermanji, G., et al. "The TIMED Power System Design," *5th European Space Power Conference*, SP-416, Vol. 1, European Space Agency, Noordwijk, The Netherlands, 1998, pp. 447-453.
[3]Ercol, C. J., et al. "Prototype Solar Panel Development and Testing for a Mercury Orbiter Spacecraft," *35th Intersociety Energy Conversion Engineering Conference (IECEC)*, AIAA-2000-2881, Vol. 1, 2000, pp. 449-459.
[4]Dakermanji, G., et al. "The MESSENGER Spacecraft Power System Design and Early Mission Performance," *7th European Space Power Conference*, SP-589, European Space Agency, Noordwijk, The Netherlands, 2005.
[5]Le, B., et al. "The MESSENGER Power Distribution Unit Packaging Design," *21st Digital Avionics Systems Conference*, Irvine, California, paper 9B3, 8 pp., October 2002.
[6] http://messenger.jhuapl.edu/