THE MESSENGER SPACECRAFT POWER SYSTEM DESIGN AND EARLY MISSION PERFORMANCE

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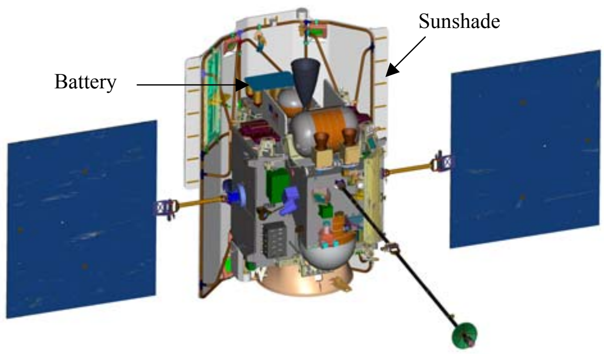
ABSTRACT

The MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) spacecraft was launched on August 3, 2004. The spacecraft will be inserted into Mercury orbit in March 2011 for one year of orbital operation. During the mission, the spacecraft distance to the Sun will vary between approximately 1 and 0.3 Astronomical Units (AU), imposing severe requirements on the spacecraft thermal and power systems design. The spacecraft is maintained behind a sunshade. The two single-axis, gimbaled solar array panels are designed to withstand the expected high temperatures. A peak power tracking system has been selected to allow operation over the widely varying solar array I-V curves. In order to reduce cost and risk while increasing the likelihood of mission success, the approach taken in the power system design, including the solar arrays, was to use conventional design, materials, and fabrication techniques.

1. 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 of orbital operation. Most of what is known about Mercury comes from the Mariner 10 spacecraft. Using three flybys, Mariner 10 was able to map about 45% of the planet surface during a one-year period between 1974 and 1975. During the three MESSENGER flybys of Mercury, regions unexplored by Mariner 10 will be seen for the first time, and new data will be gathered on Mercury's exosphere, magnetosphere, and surface composition. 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 wide-angle

a. Launch Configuration



b. Flight Configuration

Fig. 1. The MESSENGER Spacecraft.

multispectral imager that maps landforms and surface spectral variations. The Mercury Laser Altimeter produces measurements of surface topography.

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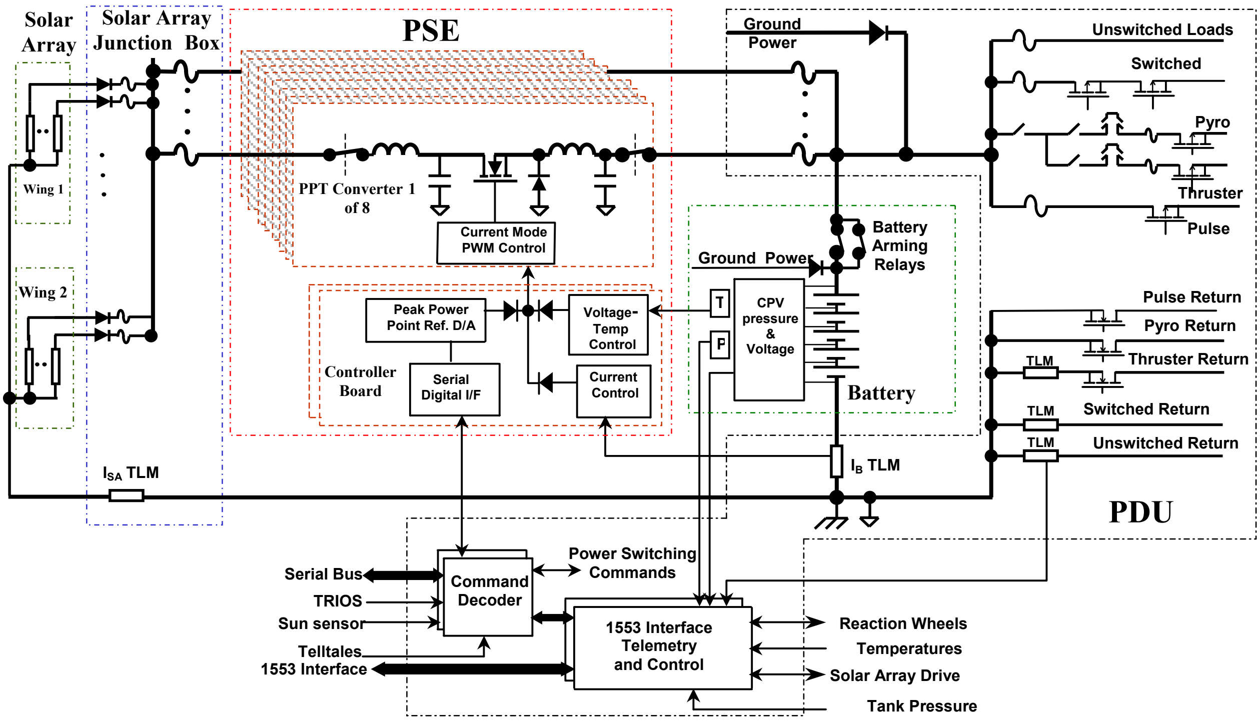

Fig. 3. Simplified Block Diagram of the MESSENGER Spacecraft Power System

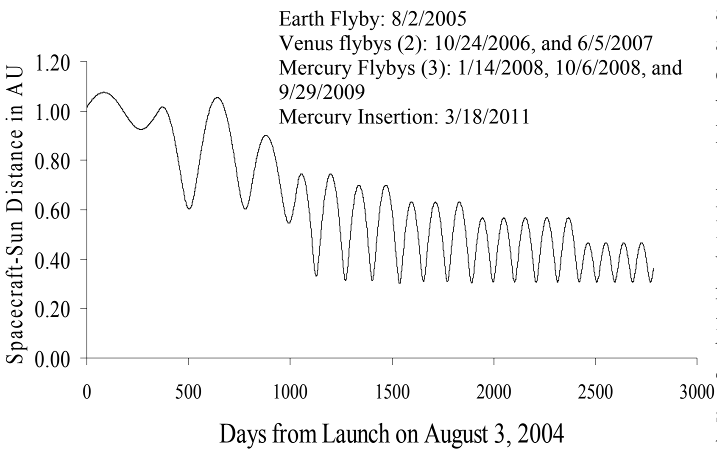

Fig. 2. MESSENGER Trajectory

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 Surfaces Composition Spectrometer (MASCS) includes an ultraviolet-visible spectrometer that measures abundances of atmospheric gases and a visible-infrared spectrometer 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. Fig. 2 shows the solar distance variations during the mission. The spacecraft trajectory requires two gravity-assist flybys from Venus and three from Mercury. The spacecraft-to-sun distance varies between 1.075 Astronomical Units (AU) and 0.3 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.

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 maintains the sunshade pointed to 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 early 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 steady-state solar illumination at normal incidence at the 0.3-AU mission perihelion.

2. POWER SYSTEM OVERVIEW

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. This range does not include the transient voltages expected on the solar array at the exit from eclipses or during abnormal spacecraft attitude control condition at 0.3 AU or orbital sub-solar crossings. A Peak Power Tracker (PPT) topology with strong heritage to the APL-designed 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, the battery, the two solar array (S/A) panels, and the Solar Array Drive Assembly (SADA). A simplified block diagram of the power system is shown in Fig. 3. Triple junction solar cells 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.

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 de-coupling diodes that are placed 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. During cruise and the full Sun orbits, the panel rotation can be performed either by periodic ground commands or by on-board software algorithms in the Integrated Electronics Module (IEM) processors. During eclipsing orbits, the panel rotation will be performed by ground commands. 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. The loads are connected directly to the single 22-cell, 23-Ah Nickel Hydrogen (NiH_2), two cells in one Common Pressure Vessel (CPV) battery. The nominal bus voltage is 28 V and can vary between 22 and 35 V depending on the state of charge of the battery. Each battery pressure vessel contains two battery cells and can be bypassed by a contactor that is automatically activated to short the vessel in case of an open circuit failure of that CPV cell. In case of a bypass switch activation, the corresponding bus voltage range becomes 20 to 32 V. To minimize the overall spacecraft weight, the battery size selected does not support the full spacecraft load during the predicted eclipses, so load management is planned. Two independent series power line switches are used to insure that loads can be removed from the power bus.

The primary battery charge control method is ampere-hour integration Charge-to-Discharge (C/D) ratio control performed by the spacecraft IEM Main Processor (MP). The battery is charged at high rate, limited to C/2, where C is the battery capacity, with available S/A power until the battery State-of-Charge (SoC) reaches 90%. Commands from the MP then lower the battery charge current to C/10. When the selected C/D ratio is reached, the MP commands the charge current to trickle charge rate. The MP also monitors the battery pressure. If the battery pressure reaches a predetermined level indicating full state of charge, the battery charge is commanded by the MP to trickle rate.

The battery voltage is controlled to preset safe levels via temperature-compensated voltage (V/T) limits that are implemented in hardware. Whenever the battery voltage reaches the V/T limit, the V/T control loop will force the charge current to taper. The use of these hardware/software battery charge control techniques reduces the battery overcharge and its associated heat dissipation and extends the life of the battery.

If the spacecraft loads and the battery charge requirements exceed the solar array power available, the power system operates in the peak power tracking mode. During this mode of operation, the solar array voltage is varied to maintain peak power from the solar array. The solar array voltage control signal is determined by the PPT algorithm that is executed by the MP.

The eight PPT converter modules have fuses at their input and output lines. A four-pole-double-throw latching power relay was used on each module card to isolate a failed PPT module when the solar array current is not adequate to clear the fuse. Current and temperature of each PPT module are monitored.

Autonomy algorithms in the IEM processors isolate a failed module if a fault is detected.

The peak power tracking, ampere-hour integration, pressure control, and C/D ratio control are all performed by the spacecraft MP, with the telemetry and commanding done through the MIL-STD-l553 interface bus. The power subsystem design, which also includes the spacecraft load power switching and distribution function, is single-point-failure tolerant. The spacecraft IEM MP and Fault Protection Processor (FPP) perform the fault detection and correction.

The average spacecraft load power during flight, with the sunshade pointed in the anti-Sun direction, is 260 W. The predicted load with the sunshade pointed to the Sun is 380 W. The orbital load during Sun-time is 595 W.

2.1 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 wings are attached to the spacecraft structure in five locations: the hinge root, the center release, and three ball and socket attachments. Each wing stows against the spacecraft bus using two saloon-door hinges to fit inside the Delta II fairing. A 0.76-m-long strut provides separation from the spacecraft when deployed. The hinges allow over-travel. A deforming metal damper slows and then stops the motion. The system has a hard stop to prevent the wings from contacting the spacecraft. The strut is made of titanium to limit thermal loads on the spacecraft. After panel deployment, a small pin-pusher was actuated at each hinge to lock the hinges. 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 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 panel 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 minimum efficiency of 28%, from EMCORE Corp. The cells use a standard one-Sun cell top metallization grid design. The cover glass on each cell is 0.15-mm-thick cerium-doped microsheet, CMG type from Thales Space Technology, with magnesium fluoride anti-reflective coating. Analysis indicated that conductive coatings for electrostatic discharge protection were not required on either the cell cover glass or the panel backside. The cover glass is bonded to the cells with standard DC93-500 transparent adhesive. The Cell-Interconnect-Cover (CIC) glass strings are bonded to the Kapton-insulated side of the panel using RTV adhesives. The cell interconnects utilize silver plated Kovar material. The cell lay-down was carried out by Northrop Grumman Space Technology (NGST).

Shadowing on the MESSENGER panels is expected during an attitude anomaly. The survival of the discrete silicon diode from Sharp Corp. used with EMCORE triple junction cells was verified after subjecting the diodes to the equivalent current of 11-Sun illumination at the maximum solar panel temperature expected during a spacecraft attitude anomaly at 0.3 AU.

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.

Each string has 36 cells. The strings run parallel to the +X axis (Fig. 1) with one string per row. To minimize the magnetic field induced by the currents in the strings, adjacent strings are placed with alternating current polarity, and the strings are back wired such that each string return runs under its cells.

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).

Extensive cell characterization and panel testing was completed as part of the design verification and solar panel qualification, including high-temperature, high - intensity UV exposure in vacuum. CICs, using DC-93-500 with Thales CMG and CMX cover glasses from three U.S. cell manufacturers were exposed to five-Sun ultra-violet (UV) radiation at 150°C, the maximum expected operational temperature, for 4200 hours. The degradation with exposure variations was asymptotic and less than 4%.

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 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. The flight panels were cycled in vacuum 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.

Radiation damage on the solar cells is caused predominantly by solar flare protons. The estimated total dosage with 0.15-mm micro sheet cover glass is $4 \times 10^{+14}$ equivalent 1-MeV/cm2 electrons.

The mass of the two solar array panels is 34.12 kg.

2.2 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 cells are capable of supporting the required 8-year mission life. Bypass switches from G&H Technology, Inc., were placed across each CPV to eliminate the potential of a spacecraft single-point failure caused by an open circuit of a pressure vessel. In the event of a cell open-circuit failure, small diodes provide a current path to blow a fusing element in the activating coil of the bypass switch. This circuit provides protection against an open-cell fault for both charge and discharge operation. The bypass switch performance in case of a fault activation with fully charged cells was verified by activating the switch across a fully charged test cell. A current over 450 A was observed at switch closure. A low resistance current path was maintained through the activated switch.

The battery chassis is electrically insulated from the spacecraft structure and connected to the S/C ground through two parallel 20-kΩ resistors. The battery cells were demagnetized, and the inter-cell wiring is designed to minimize the induced magnetic fields. Approximately 300 eclipses are expected during the one-year Mercury orbit phase. The maximum eclipse duration is 60 minutes. Power is managed during Mercury eclipses to allow adequate reserve battery energy to recover from any attitude anomaly. The maximum Depth-of-Discharge (DoD) expected during Mercury orbital operations is approximately 55%. The battery DoD from launch to Sun acquisition was 18%.

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. The battery package includes vessels with calibrated strain gage pressure transducers. The signals from the strain gauge are processed in bridge circuits in the battery. The voltage of each CPV is monitored. The pressure and voltage signals are multiplexed in the battery and read by the PDU over redundant serial data interface cards.

Fig. 4. The MESSENGER Battery

The battery is shown in Figure 4. Its weight is 24.5 kg (without the extra thermal plate mass needed to support the August 2004 launch), and the size is 36.6cm x 22.83cm x 49.17cm.

2.3 Power System Electronics (PSE)

The PSE contains eight buck-type peak-power-tracking converter modules, each designed to process around 130 W of output power. They are controlled by redundant controller boards with one controller active at a time. The outputs of the battery current and V/T controllers are diode OR-ed with the IEM peak-power-tracking control signal. All control loops are active simultaneously. The dominant signal determines the peak power tracking converters' output current. When the load and battery recharge power exceeds the S/A power capabilities, the IEM-generated signal will control the peak-power-tracking converter operation. During this time, the control signals generated by the current and V/T control loops will be saturated with low output voltages. When the V/T, maximum battery current, or trickle charge current limits are reached, the control signals from the output of the error amplifier will increase and drive the buck converters Pulse Width Modulated (PWM) duty cycle signal such that the S/A operating point moves toward the solar array open circuit voltage (V_{oc}). The IEM MP monitors the outputs of the current and V/T controllers, and if the control signals indicate V/T or current limit control, the peak power tracking continues; however, the allowed change in the solar array control signal is made very small in order to minimize the effect of the peak power tracker controller from perturbing the active control signal. The PPT control signal remains close to the value of the active controlling signal to minimize the response time of the PPT loop when the peak power tracking is resumed.

The peak power tracking is performed by adjusting the solar array reference voltage to the buck converters. The output of the buck converters is clamped to the battery voltage. 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 IEM MP. The peak power calculations and the setting of the converter module reference voltage are performed by the MP. The signals are sent to the PDU on the Mil-STD-1553 bus. The PDU decodes the signals and sends them to the PSE on a dedicated and redundant PSE-PDU serial telemetry and command bus. The solar array current of each wing is monitored by the PDU data acquisition system. The MP uses the previously commanded solar array voltage data to calculate the instantaneous power of each solar array wing. These new power values are compared to previously stored power values. If the new values are larger than the previous values by a preset amount, then the MP control signals that set the reference to the solar array voltages are moved in the same direction as the previous change. Otherwise they are moved in the opposite direction with a smaller step voltage value. Two step sizes are used to speed up the conversion. A large step is used initially; if a direction change is made, then the smaller step size is used. The step size reverts to the larger step size if more than a predetermined number of steps are taken without a change in direction. The power calculations and the steps are performed at 5-Hz speed. Maximum and minimum S/A control voltages are preset in software to values corresponding to the maximum predicted solar array V_{oc} voltage and the minimum battery voltage plus PPT converter minimum input to output differential voltage, respectively. If these limit values are reached then the respective direction of solar array voltage change is reversed. During eclipse, when the measured wing currents are zero, the IEM MP peak-power-tracking software algorithm will set the reference voltage to that corresponding to the battery voltage plus the minimum PPT converter input to output differential voltage. This will force the eclipse-exit cold solar array operating voltage to be low, reducing the exposure of the PPT converters to the high voltage and high current from the cold solar array.

The peak power converters are current-mode controlled step down (buck) regulators. The Unitrode UC 1846 PWM controller chip is connected to provide zero to almost 100% duty cycle ratio. The converters are operated at a frequency of approximately 50 kHz. This was selected as a trade-off between the power efficiency and the weight of the converter modules. The current mode controller topology allows sharing of the current among the eight converters. The frequencies of the converters are not synchronized.

Prior to the solar panel deployment after launch and fairing ejection, the third stage is spun up to 60 rpm to stabilize the spacecraft trajectory during the third stage motor burn. The exposed solar cells are illuminated during the high spin mode at a 1 Hz rate. Although the PPT function runs at a 5 Hz rate, the step change to the solar array voltage is 0.5V per algorithm call and execution. The sinusoidal shape solar array power during the high spin rate will have a maximum-power-point voltage of around 80V. When the wing currents are zero, the PPT algorithm outputs the solar array voltage control word equivalent to the battery voltage plus the minimum PPT input to output differential voltage. It will take over 16 seconds for the PPT algorithm to step the solar array voltage up to the peak power point. Tests were conducted on the power system using the spacecraft Solar Array Simulator to test the PSE operation during the spacecraft high spin rate. The test verified that a power level close to the power level available from the solar array at a solar array voltage of approximately 38 V was delivered to the spacecraft during this mode. Telemetry playback of the launch data also verified that significant solar array power was delivered to the spacecraft during the 1-Hz spin mode.

The current of each PPT converter is monitored using small magnetic amplifier placed on the positive (high) side of the solar array input power lines to the PPT converters.

The sixteen V/T levels, limited to a 35-V maximum battery voltage, are selected by binary switching of semiconductor switches of different resistors in a voltage divider circuit, and comparing these with a reference in the error amplifier. The cell V/T limits are modified NASA/Goddard Space Flight Center standard V/T limits to support the NiH_2 battery:

Slope: -4.5 ± 0.20 mV/ ºC
Separation between levels: 0.020±0.002 V
Lowest level at 0 °C: 1.3 ± 0.015 V

The battery current is sensed across a resistive shunt in the battery negative line with differential amplifiers. The signal is then compared in an error amplifier to the selected reference. If the trickle charge or the C/10 rate semiconductor switches are not set then the pre-set reference will limit the maximum battery charge current to C/2.

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

2.4 Solar Array Junction Box

The Solar Array Junction Box contains the isolation diodes in series with each string of the two solar array panels, the current shunt resistor of each wing, the peak power tracker module solar array side fuses, and solar array voltage telemetry buffer resistors.

The fuses placed in series with each string inside the solar array junction box protect the peak power tracker modules' input and output fuses in case of a short in one of the string isolation diodes during the fault condition when the battery voltage is higher than the solar array voltage. This condition can occur during a spacecraft attitude anomaly when the spacecraft is closest to the Sun and the solar panels are pointed normal to the sun.

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

2.5 Power Distribution Unit (PDU)

The PDU contains the circuitry for the S/C 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, 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. Each side of the PDU can command all of the PDU's circuitry. The commands to control the PDU under normal circumstances are uplinked through a Main Processor (MP) located in the IEM and sent over the Mil-STD-1553 bus to one of the PDU 1553 cards. The commands are then sent internally through the PDU Motherboard to either one or both PDU Command Decoder (CD) cards. In the case of a fault on an MP, an asynchronous serial bus exists between the IEM FPP and the PDU CD cards that can reconfigure the MP and the PDUs.

Solid-state, radiation-hardened, power Metal Oxide Semiconductor Field Effect Transistors (MOSFETs) have been selected for power switching distribution. The load switches consist of two P-channel MOSFETs in series to allow for high-side switching and to ensure the capability to turn-off a load even with a failure in one of its MOSFETs. Each of the 58 loads has independent A-and B-side current and voltage monitoring, and the spacecraft main power bus is protected from load faults by redundant FM-12 type solid-body fuses. Each side of the PDU can control the power MOSFETs through independent, isolated circuitry that share only the power MOSFET's gate.

Propulsion and pyrotechnic firing schemes use a combination of high- and low-side switching MOSFETs and enable relays for galvanic isolation of power grounds. This arrangement requires separate enable, arm, and fire commands from the IEM. High current relays are configured to enable and disable the propulsion and pyrotechnic busses separately and provide the S/C with three modes of operation: disabled, side A, and side B. Pyrotechnic circuits have voltage monitoring to provide safety checks before arming, and propulsion circuits have current telemetry for monitoring.

Telemetry that is collected by the PDU consists of external relay statuses, load currents, load voltages, battery and solar array currents, solar array and battery temperatures, solar array voltages, reaction wheel speeds, propulsion tank and battery pressures, solar array position, S/C temperature, S/C load current, and Digital Sun Sensor (DSS) information. All telemetry interfaces with A and B sides of the PDU. Analog telemetry is digitized in 12-bit A/D converters, and all telemetry can be sent to either IEM from either side of the PDU through MIL-STD-1553 busses. The PDU command and telemetry capability, including both sides, is as follows:

58 Load commands	53 Used
30 Pyro commands	28 Used
72 Propulsion commands	70 Used
58 External Relay commands	56 Used
40 Internal PDU commands	38 Used
554 Analog telemetry	494 Used
120 TRIO telemetry	120 Used
100 Discrete telemetry	70 Used

The PDU dimensions are 23.4 cm x 22.6 cm x 35.2 cm (with the thermal vias), and the weight is 12.81 kg.

2.6 Solar Array Drive Assembly (SADA)

Each solar panel is independently rotated by a Moog Type 2 stepper motor actuator with harmonic drive gearing. A cable wrap that allows rotation from -20 to 200^o around the X-axis, with zero being the panel normal in the -Y (sunshade) direction, is used to transfer the solar array power and panel temperature signals to the power system electronics.

The weight of the two SADA units is 6.08 kg.

3. LAUNCH SITE OPERATIONS

The flight battery was mounted on the spacecraft at Astrotech Space Operations facility near Cape Canaveral. The final battery reconditioning was performed about two weeks before launch after spacecraft spin balance tests. The battery was left discharged and open circuit during the spacecraft fueling operations, transportation to launch site, and integration on top of the launch vehicle. The battery charging was done after establishing the battery air-conditioned cooling. A final top-off charge was done after the installation of the fairing. Trickle charge and battery cooling were maintained on the battery until T-6 minutes. At launch the battery temperature was 18˚C.

4. LAUNCH AND EARLY MISSION OPERATIONS

After launch, following the fairing ejection, the Sun angle on the stowed solar panels was very favorable and substantial solar energy was supplied to spacecraft.

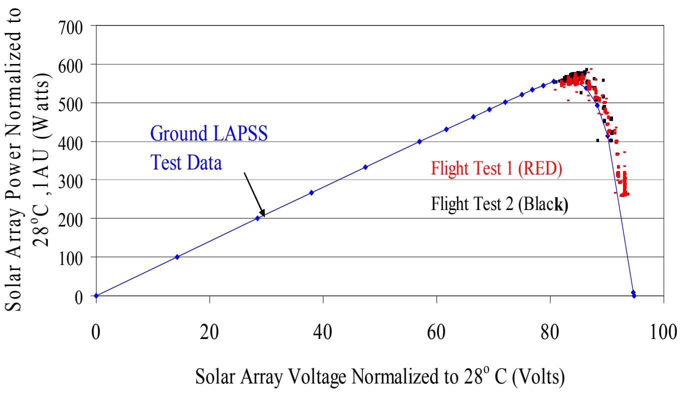

Fig. 5. Measured Flight and Ground LAPSS Data

Spacecraft and instrument operation was verified, and the spacecraft is performing as designed. The battery is fully charged and placed in a C/150 trickle charge for the long cruise phase. The battery heaters, under thermostatic switch operation, are maintaining the battery between -6˚C and -3˚C. Special tests were conducted to check the health of the solar panels and evaluate the peak available power. The loads were increased until the battery discharged, causing the peak power tracking to be initiated. The solar array panel temperatures, voltage, and currents were monitored. The results of two tests conducted are shown in Fig. 5. The results indicate that the panels did not suffer any damage at launch and that the measured panel power agrees well with the LAPPS measured power at NGST.

5. SUMMARY

A power system for a spacecraft mission to orbit Mercury is a challenging task. The power system design presented meets the MESSENGER mission requirements. The spacecraft and the power system were extensively tested, and the spacecraft has 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. The peak power tracker system topology is applicable to missions where the solar array temperature is highly variable.

6. ACKNOWLEDGEMENTS

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