THE MESSENGER SPACECRAFT POWER SYSTEM

G. Dakermanji, J. Jenkins, P. Schwartz, L. Kennedy

The Johns Hopkins University Applied Physics Laboratory
Johns Hopkins Road, Laurel, Maryland, 20723-6099, USA
E-mail: george.dakermanji@jhuapl.edu

ABSTRACT

The MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) spacecraft, being designed and built by The Johns Hopkins University Applied Physics Laboratory (APL), will orbit and survey the planet Mercury for one Earth year. Over the mission, the spacecraft distance to the sun will vary between approximately 1 and 0.3 Astronomical Units (AU). This imposes severe requirements on the spacecraft thermal and power systems design. The spacecraft is maintained behind a thermal shade. The two single-axis, gimbaled solar array panels are designed to withstand the expected high temperatures. Triple-junction cells are placed between Optical Solar Reflector (OSR) mirrors, and the panels are tilted to reduce panel-operating temperatures. A peak power tracking system has been selected to allow operation over the widely varying solar array I-V curves. A 23-Ahr NiH_2 battery, consisting of eleven Common Pressure Vessels (CPV) with two cells in each vessel, is used to support the launch and the expected eclipses. In order to reduce cost and schedule risk while increasing the likelihood of mission success, the approach taken in the power system design, including the solar arrays, is the use of conventional design, materials and fabrication techniques.

1. INTRODUCTION

Mercury is one of the least studied of the solar system planets. It holds answers to several critical questions regarding the formation and evolution of the terrestrial planets [1]. Only the Mariner 10 spacecraft has visited Mercury. 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 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, one Earth year in duration, MESSENGER will complete global mapping and the detailed characterization of the exosphere, magnetosphere, surface, and planet interior.

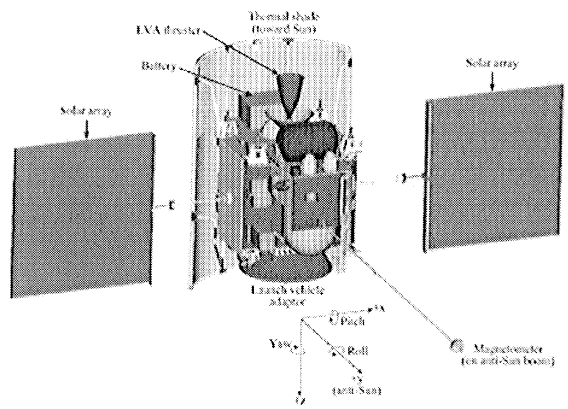

Fig. 1. MESSENGER Spacecraft [2]

The instrument payload includes a dual imaging system with wide and narrow fields-of-view, monochrome and color imaging, and stereo; X-ray and combined gamma-ray and neutron spectrometers for surface chemical mapping; a magnetometer; a laser altimeter; a combined ultraviolet-visible and visible-near-infrared spectrometer to survey both exospheric species and surface mineralogy; and an energetic particle and plasma spectrometer to sample charged species in the magnetosphere. Fig. 1 shows the deployed spacecraft configuration.

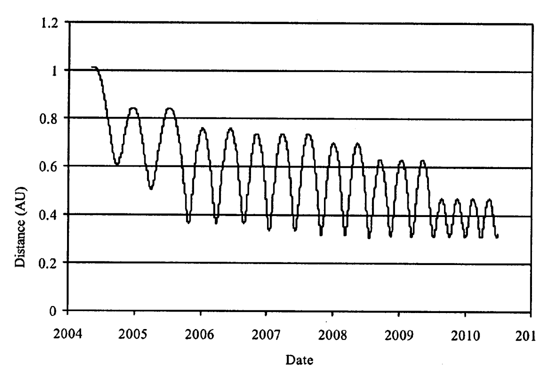

Fig. 2. MESSENGER Spacecraft Trajectory

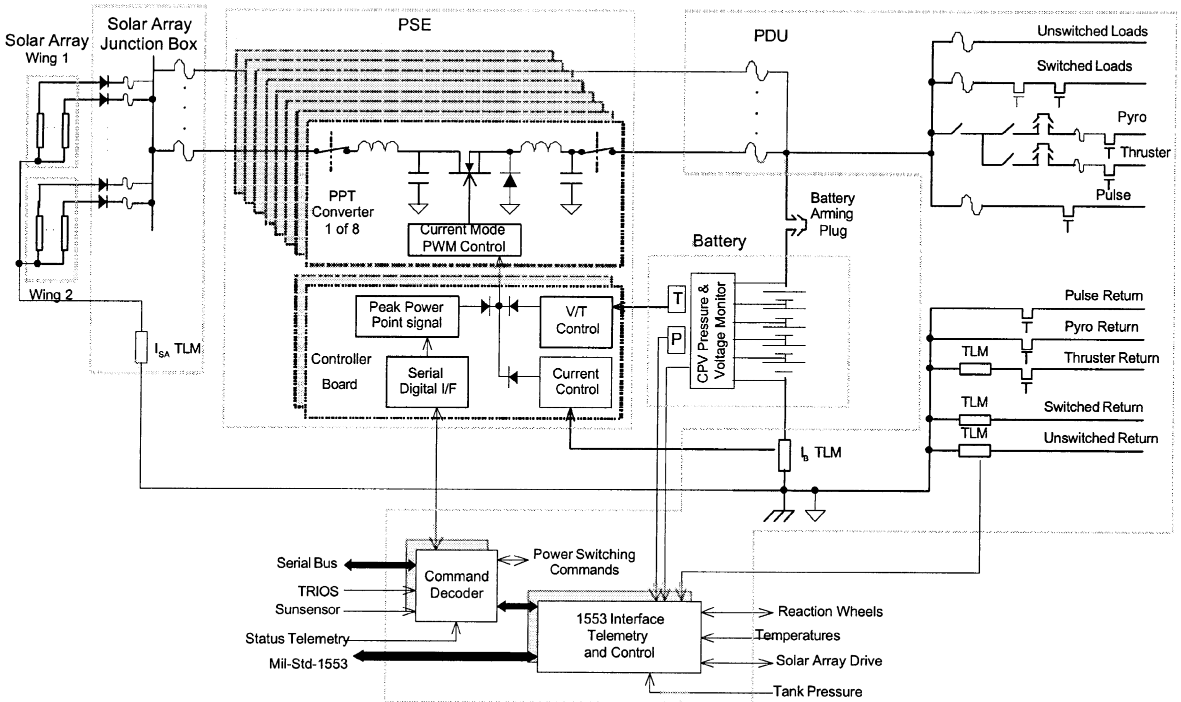

Fig. 3. Simplified Block Diagram of the MESSENGER Power System

Trajectory optimization for a Delta 7925H-9.5 launch vehicle yields a 20-day baseline in March 2004 and a 12-day backup launch period in May 2004 [2]. Fig. 2 shows the sun distance variations for the primary launch date. The spacecraft trajectory requires two gravity assist flybys from Venus and two from Mercury. The spacecraft to sun distance varies between around 1 AU and 0.3 AU. The solar illumination, which varies inversely with the square of the sun distance, has severe impact on the thermal designs of the solar array panels and spacecraft. The primary launch has an aphelion at 1 AU while the back up trajectory has an aphelion of 1.015 AU. The array size is driven by the spacecraft power demand during the initial few weeks after launch in the back up launch trajectory.

A thermal shade protects the spacecraft from the high intensity solar illumination (Fig.1). The spacecraft is kept behind the thermal shade, made of 3M Co. Nextel ceramic cloth material. The spacecraft attitude control maintains the sun shield pointed to the sun at all times when the spacecraft-sun distance is less than 0.85 AU. To reduce the spacecraft and instrument heater-power requirements during the initial 60 days after launch, the spacecraft is turned around such that the thermal shade is pointed away from the sun. The solar panels, which are outside the thermal shade, are designed to survive steady-state normal sun illumination incidence at the 0.3-AU mission perihelion.

2. POWER SYSTEM DESCRIPTION

The large solar distance variations impose severe requirements on the solar array design. The normal operational solar array maximum power point voltage is expected to vary between 45 and 95 volts, but this doesn't include the higher transient voltages expected on the cold solar arrays at the exit from eclipses. A peak power tracker topology with strong heritage to the APL-designed TIMED spacecraft power system has been selected [3,4]. This architecture isolates the battery and the power bus from the variations of the solar array voltage and current characteristics, and optimizes the solar array power output over the highly varied solar array operating conditions of the mission. The 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 are used on the solar array. The solar cell strings are 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 270°C. All material and processes used in the solar panels are designed to survive the worst case predicted temperatures. The array strings are isolated with de-coupling diodes 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 limit (maximum 150°C). The panels are rotated by the SADA to limit the temperature to the preset value. The panels are rotated toward normal incidence as the panel temperature drops below the limit value. 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 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 peak power tracker converter modules is used. The loads are connected directly to the 22-cell, 23-Ahr NiH_2 battery used to support the spacecraft loads during launch and eclipse periods. The nominal bus voltage is 28 volts and can vary between 22 and 35 volts depending on the state 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 volts. To minimize the overall spacecraft weight, the battery size selected will not support the full spacecraft load during the predicted eclipses, and therefore 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 is ampere-hour integration Charge-to-Discharge (C/D) ratio control performed by the spacecraft Integrated Electronics Module (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 85%. 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 a C/100 trickle charge rate. The MP 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 with temperature compensated voltage (V/T) limits that are implemented in hardware. Whenever the battery voltage reaches the V/T limit, the V/T loop will force the charge current to taper. The battery charge control technique used reduces the battery overcharge and its associated heat dissipation and extends the battery life.

The eight Peak Power Tracker (PPT) modules have fuses at their input and output lines. A four-pole-double-throw latching power relay is 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. To minimize the inrush currents into a PPT module, the input and output capacitors are pre-charged when the module isolation relay is open.

The peak power tracking, ampere-hour integration, pressure control and C/D ratio control are performed by the spacecraft MP, with the telemetry and commanding of the charge current done using the MIL-STD-1553 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) performs the fault detection and correction.

Radiation damage on the solar cells is caused predominantly by solar flare protons. The estimated total dosage on the solar cells with 0.15 mm microsheet cover glass is $4x10^{14}$ equivalent 1-Mev/cm^2 electrons. The power system is designed to support about 390 W load power near Earth and 640 W with illuminated solar panels during Mercury orbit.

2.1 Solar Array

The MESSENGER solar array consists of two deployed single panel wings; each panel is 1.54 m wide and 1.72 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 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 and provide 0.76-m 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 0.76-m strut is made of titanium to limit thermal loads on the spacecraft. Once the panels are deployed, a small pin-pusher is 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 aluminized Kapton. 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 absorbance. 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 26%. Each cell is covered by a 0.15-mm-thick cerium-doped microsheet cover glass with magnesium fluoride anti reflective coating. Extensive analysis indicated that conductive coatings for electrostatic discharge protection are not required on either cell coverglass or the panel backside. The cover glass is bonded to the cells with standard DC93-500 transparent adhesive. The Cell-Interconnect-Cover glass (CIC) strings will be bonded to the Kapton insulated side of the panel using standard controlled-volatility RTV adhesives, and the cell interconnect will utilize either silver plated Invar or Kovar material, depending on the final cell lay-down manufacturer selected.

The standard one-sun cell top metalization grid design is used for MESSENGER cells. The resistive losses increase with the increased sun illumination intensity to five-sun expected around Mercury with off pointing. However, modifying the metal grid design to optimize the cell power for five-sun intensity will cause a reduction in cell efficiency at one sun, which is the minimum power margin case for MESSENGER. The reduction in cell efficiency caused by the metal grid resistance is offset by the expected solar cell efficiency increase with increased solar intensity.

Triple junction cells require individual bypass diodes to protect the solar cells from reverse bias. Shadowing on the MESSENGER panels is expected during an attitude anomaly. Diodes or diode/cell assemblies were obtained from the three US solar cell manufacturers, and their survival was verified after subjecting the diodes to the equivalent current of 11-sun illumination at the maximum expected solar panel temperature during a spacecraft attitude anomaly at 0.3 AU.

All electrical interconnections including cell repairs are welded or high-temperature brazed. High-temperature wire is used. The staking of the wires is done with controlled volatility RTV. The wires are routed behind the thermal shade to connectors at the SADAs. The solar panel temperatures are sensed using platinum wire sensors placed underside of the solar cell side face-sheet in small bored and potted cavities.

Each string has 36 cells. The strings run parallel to the S/C X axis 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, APL successfully completed a series of high-temperature tests including IR heater and high-intensity illuminated high-temperature tests in vacuum at the Tank 6 facility at NASA Glenn Research Center (GRC) [5].

Extensive cell characterization and panel testing continues as part of the design verification and solar panel qualification:

Cells from the three US manufacturers where exposed to accelerated life testing at high temperatures. The duration and temperatures of the tests where calculated by the cell manufacturers to correspond to their respective cell activation energies. All cells were found to be capable of supporting the MESSENGER mission. CICs using DC-93500 with CMG and CMX cover-glass from three 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%.

Several additional test panels with solar cells and OSRs were fabricated by four cell lay-down manufacturers. The panels were tested in vacuum over a temperature range from -130°C to 270 °C. they were tested at NASA-GRC at 11-sun intensity illuminations. 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 set to 80°C/minute to simulate the thermal shocks expected at Mercury eclipse exit.

The temperature coefficients and I-V characteristics of the cells were measured to around 275°C. The cell OSR layout is shown on the test solar panel in Fig. 4.

The mass of the two solar array panels is estimated to be 32.15 kg.

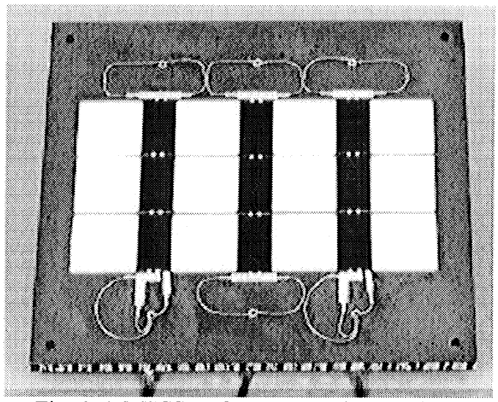

Fig. 4 A MESSENGER Test Solar Panel

2.2 Battery

The MESSENGER spacecraft has a single battery consisting of eleven 23-Ahr Nickel Hydrogen (NiH_2) CPVs with two cells in each pressure vessel, in rabbit ear configuration from Eagle Picher, Space Energy Products Division. The cells are capable of supporting the required eight-year mission life. Bypass circuits are placed across each CPV to eliminate the potential mission single-point failure caused by an open circuit of a pressure vessel. In case 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, during either charge or discharge. The battery chassis is electrically insulated from the spacecraft structure and connected to S/C ground through 10 kΩ resistors. The battery inter-cell wiring minimizes the induced magnetic fields. Approximately 300 eclipses are expected during the one-year Mercury orbit phase. The maximum eclipse duration is 60 minutes. Power management is done during Mercury eclipses to allow adequate reserve battery energy to recover from any attitude anomaly. The Mercury orbital operation maximum Depth-of-Discharge (DoD) expected is approximately 55%. The worst case predicted time from launch to deployment and sun acquisition is 65 minutes. The predicted battery DoD from launch to sun acquisition is about 55%.

The preliminary battery package design is shown in Figure 5. The thermal design of the battery maintains operation between -5 and +10°C during most of the mission. However, the temperature may reach 20°C 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 gage are processed in bridge circuits in the battery. The voltages of the CPVs are monitored. The pressure and voltage signals are multiplexed in the battery and read by the PDU over redundant serial data interface busses.

The battery weight is approximately 23.5 kg and the size is 46.36 cm x 34.3 cm x 22.86 cm.

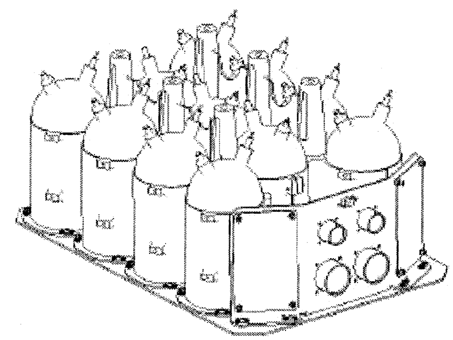

Fig.5 Preliminary Battery Configuration

2.3 Power System Electronics (PSE)

The power system electronics (PSE) are based on the TIMED spacecraft power system electronics [3,4]. It contains eight buck-type peak power tracking converter modules which are controlled by redundant controller boards. Only one controller is active at a time. The outputs of the battery current and V/T controllers are ORed with the IEM peak power tracking control signal. All control loops are active simultaneously. The dominant signal drives the peak power tracking converters. When the load and battery recharge power demand 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 the 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 solar array open circuit voltage (Voc). 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 is stopped by setting the reference to a fixed low level. This prevents the peak power tracking from disturbing the current or V/T control loops. The peak power tracking is continued when the current and V/T control signals go low.

The peak power tracking is performed by adjusting the reference voltage to the buck regulators. The output of the buck regulators 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 is performed by the MP. The signals are sent to the PDU on the Mil-STD-1553 bus. The PDU decodes the signal and sends them to the PSE on a dedicated 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 stored 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 which 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. Maximum and minimum S/A voltages are preset in software to values corresponding to the maximum predicted solar array Voc voltage and minimum battery 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 PM peak power tracking software algorithm will continue to scan between the two limits.

The peak power converters are current-mode-controlled step down (buck) regulators. The Unitrode UC1846 PWM controller chip is connected to provide zero to almost 100% duty cycle ratio. Operating the converters at approximately 60 kHz optimizes the 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.

The sixteen V/T levels provided, limited to 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 standard V/T limits to support the NiH_2 battery with one shorted CPV specified as:

Slope: -2.33 ± 0.20 mV/°C
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 estimated PSE dimensions are 28.6 cm x 19.55 cm x 14.2 cm, and the weight is 6.5 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 will protect the peak power tracker module input and output fuses in case of shorts of the string isolation diodes 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 panel are pointed normal to the sun.

The estimated box dimensions are 17 cm x 20 cm x 5 cm, and the weight is 1 kg.

2.5 Power Distribution Unit (PDU)

The PDU contains the circuitry for the S/C pyrotechnic firing control, the power distribution switching, load current and voltage monitoring, fuses, external relay switching, reaction wheel relay selects, power system relays, Inertial Measurement Unit (IMU) reconfiguration, 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 PDU block diagram is shown in Fig. 6.

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 Fault

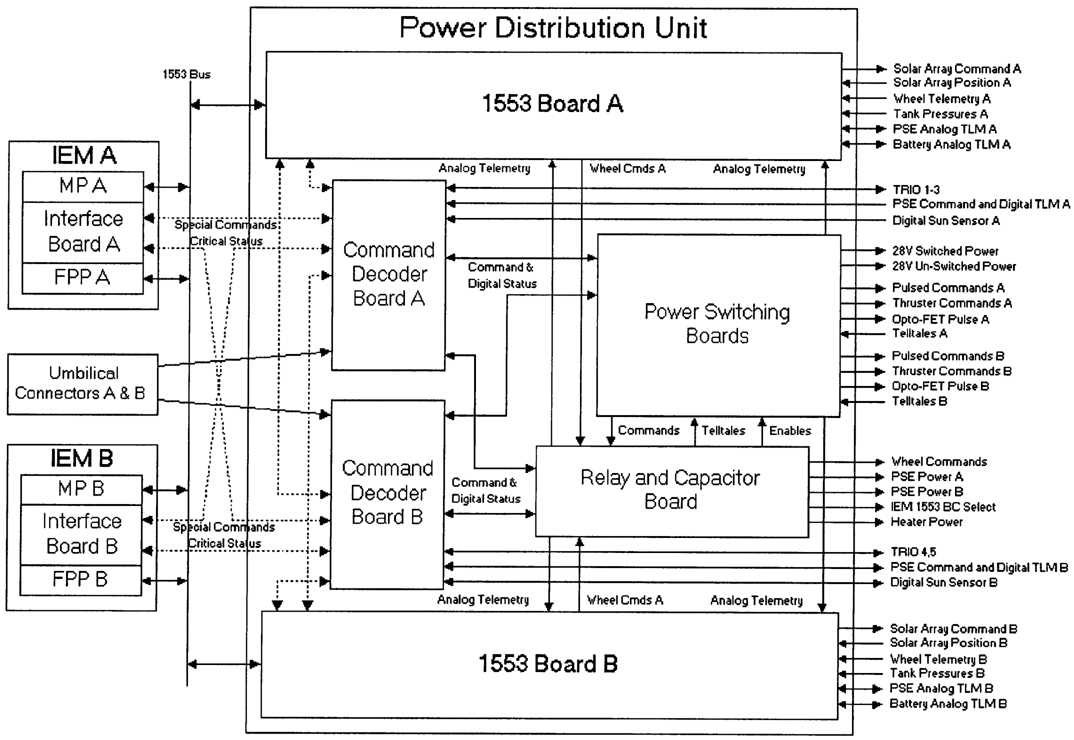

Fig. 6 Power Distribution Unit Block Diagram

Protection Processor (FPP) and the PDU CD cards that enables the PDU to turn off non-critical loads and reconfigure the MP.

Solid-state, radiation-hardened, power Metal Oxide Semiconductor Field Effect Transistors (MOSFET) have been selected for power switching distribution. The load switches consist of two P-channel MOSFETs in series to allow for high-side switching to ensure the capability to turn-off load even with a failure in one of its MOSFETs. Each of the 54 loads has independent A- and B-side current and voltage monitoring, and the spacecraft is protected from load faults by redundant FM-12 type solid-body fuses from the S/C main bus. Each side of the PDU can gate the power MOSFETs through independent, isolated circuitry and share only the power MOSFETs.

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 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, 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 temperature, 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 total command and telemetry capability that includes both sides are:

54 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 estimated PDU dimensions are 23.4 cm x 22.6 cm x 23.9 cm, and the weight is 12.5 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 of –20 to 200 degrees around the X-axis, zero being the panel normal in the –Y (sun) 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 about 6.45 kg.

3. 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. Extensive solar array development and testing are ongoing to characterize and qualify the design. The spacecraft Critical Design Review is scheduled for mid March 2002. The design is progressing for launch in March 2004.

4. ACKNOWLEDGEMENTS

The MESSENGER mission is supported by the NASA Discovery program under contract to the Carnegie Institution of Washington and The Johns Hopkins University Applied Physics Laboratory.

5. REFERENCES

1. Solomon S. C. et al. The MESSENGER Mission to Mercury: Scientific Objectives and Implementation. *Planetary and Space Science*, vol. 49, Issue 14, pp. 1445-1465, December 2000.

2. Santo A. G. et al., The MESSENGER Mission to Mercury: Spacecraft and Mission Design, *Planetary and Space Science*, vol. 49, Issue 14, pp. 1481-1500, December 2000.

3. Dakermanji G. et al., The TIMED Power System Design, *Fifth European Space Power Conference*, SP-416, Vol.1, Tarragona, Spain, 21-25 September 1998, pp. 447-453.

4. Dakermanji G. et al. The Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics Spacecraft Power System, *32nd Intersociety Energy Conversion Engineering Conference (IECEC),* Honolulu, Hawaii, 27 July- 1 August 1997, Vol. 1, pp. 544-549.

5. Ercol J. et al. Prototype Solar Panel Development and Testing for a Mercury Orbiter Spacecraft, *35th Intersociety Energy Conversion Engineering Conference (IECEC),* AIAA-2000-2881, Las Vegas, Nevada, 24-28 July 2000, Vol.1, pp. 449-459.