THE MESSENGER SPACECRAFT SOLAR ARRAY DESIGN AND EARLY MISSION PERFORMANCE

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ABSTRACT

The MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) spacecraft was launched in 2004 and will be inserted into Mercury orbit in March 2011 for one year of orbital operation. The large solar distance variations from 1 AU near Earth to 0.3 AU near Mercury impose severe requirements on the solar array design. 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. To minimize the development cost and risk to the program schedule, the approach taken was to avoid the development of new materials and processes but rather to extend the use of existing proven materials and processes. Extensive cell characterization and panel testing were performed successfully to demonstrate meeting the worst-case predicted environments during the mission.

OVERVIEW

MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging), is a NASA Discovery Program spacecraft designed and built by The Johns Hopkins University Applied Physics Laboratory (APL). It was launched on August 3, 2004. The spacecraft will be inserted into Mercury orbit in March 2011 for one 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. During the orbital phase of the mission, MESSENGER will complete global mapping and the detailed characterization of the exosphere, magnetosphere, surface, and interior [1].

MESSENGER was launched on a Delta 7925H-9.5 launch vehicle. Fig. 1 shows the spacecraft flight configuration. The spacecraft trajectory requires two gravity-assist flybys from Venus and three from Mercury. The spacecraft-to-Sun distance, shown in Fig. 2, varies between 1.075 Astronomical Units (AU) and 0.3 AU during the mission.

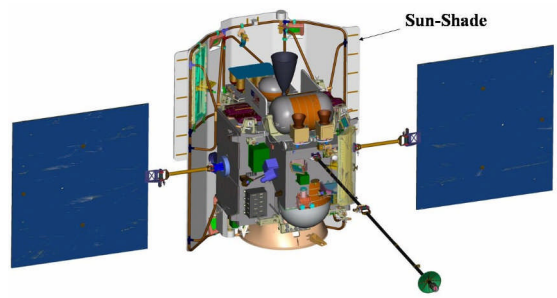

Fig. 1. The MESSENGER Spacecraft Flight Configuration

The large solar distance variations impose severe requirements on the spacecraft power and thermal systems 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 conditions at 0.3 AU or orbital sub-solar crossings.

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 available solar array power output over the highly varied solar array operating conditions of the mission.

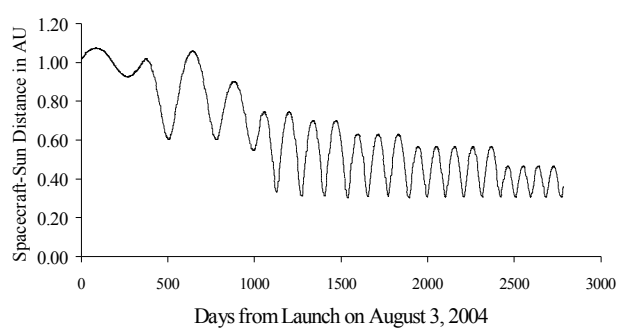

Fig. 2. MESSENGER Spacecraft Trajectory

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SOLAR ARRAY DESIGN

The high temperatures expected at close distances to the Sun drive the solar array design. Triple junction solar cells were used on the solar array. To reduce the thermal panel absorbance, the solar cell strings were placed between Optical Solar Reflector (OSR) mirrors with a cell to OSR ratio of 1:2. In case of an attitude control anomaly near Mercury, the solar array temperature may reach 275°C. All material and processes used on the solar panels were designed and tested to survive the worst-case predicted temperatures.

Thermal control is performed by tilting the panels away from normal incidence with increased solar intensity. 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 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 solar array wings are rotated to the same incident Sun-angles so that they will operate at the same temperature.

The array strings are isolated with de-coupling diodes that are placed inside the spacecraft to protect them from the expected high panel temperatures. In order to reduce the load power required by the spacecraft heaters and thus minimize the solar panels size, the spacecraft is flipped such that the sunshade is pointed away from the Sun allowing the spacecraft to be heated by the Sun. To prevent the overheating of the spacecraft boxes, the spacecraft is flipped back with the sunshade pointed to the Sun when the spacecraft-Sun distance is less than around 0.95 AU.

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 570 W. The predicted orbital load during Sun-time is 595 W.

The MESSENGER solar array consists of two deployed single-panel wings. Each panel is 1.54 m wide and 1.75 m long. A photograph of one of the MESSENGER solar panels is shown in Fig. 3.

The panel substrates are 18-mm-thick aluminum honeycomb with composite facesheets. The facesheets are 0.6-mm thick with local 0.5-mm doublers. The graphite-cyanate-ester materials on the panel facesheets 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 panel front cell side is insulated with 0.05-mm Kapton, co-cured with the graphite fiber facesheet. The back facesheet 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.

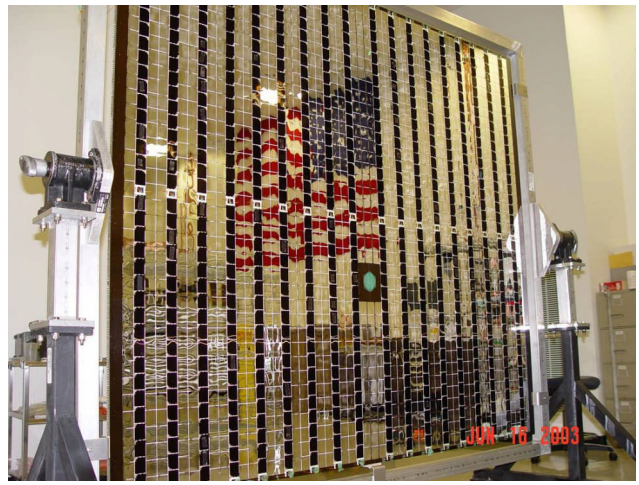

Fig. 3. MESSENGER Solar Panel

The solar cells are 0.14-mm thick, 3 cm by 4 cm Advanced Triple Junction (ATJ) cells with a 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 DC-93500 transparent adhesive. The Cell-Interconnect-Coverglass (CIC) 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 done by Northrop Grumman Space Technology (NGST).

All electrical interconnections, including cell repairs, are welded. High-temperature wire is used. The wires are routed along the titanium boom to connectors at the solar array drive assembly. Both the wires and the boom are wrapped with multi-layer insulation. The solar panel temperatures are sensed using platinum wire sensors, placed beneath the solar-cell-side facesheet in small bored cavities.

Each string has 36 cells. The strings run parallel to the +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.

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Radiation damage on the solar cells is caused predominantly by solar flare protons. The estimated total dosage with 0.15-mm microsheet coverglass is $4x10^{+14}$ equivalent 1-MeV/cm^2 electrons.

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

Test panels, approximately 18”x18” each, 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.

CELL TESTS

Extensive cell characterization testing was performed as part of the design verification. Cell characterization for solar cell parameters variations with illumination intensity and temperature were conducted at APL using the APL Large Area Pulsed Solar Simulator (LAPSS) and a Spectrolab X-25 solar simulator.

Tests to characterize the ultraviolet (UV) degradation of the CIC samples at high-intensity UV exposure in vacuum were conducted at NASA Marshall Space Flight Center (MSFC). CICs using DC-93500 with Thales CMG and CMX cover glasses obtained from three U.S. cell manufacturers were exposed to five-Sun UV radiation at 150°C, the maximum expected operational temperature, for 4200 hours. The cell I-V curves were plotted regularly while maintaining vacuum by pivoting the cell coupon to a window and illuminating the cells with the Spectrolab X-25. The UV source and the X-25 were kept stationary. No vacuum breaks were made to avoid atomic oxygen bleaching issues. The UV degradation at 150°C was asymptotic and less than 4% over the 4200 hours test duration.

Metallization stability and diffusion concerns were satisfied by the extensive tests conducted at EMCORE [5]. Metallization stability tests were also done at APL with welded and un-welded cells during high-temperature, long-duration vacuum exposure. Before and afterwards, test measurements of the cells and CICs were done at the cell manufacturer.

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 successfully after subjecting 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.

Thermo-Gravimetric Analysis (TGA) was performed on all materials used on the solar panels that are exposed to high temperature to derive their activation energies. Lifetime equivalence calculations and tests were performed successfully on the qualification panels using the activation energy data. Mass loss and condensable from the material tests data were used in the spacecraft contamination analysis to verify that the solar array materials were acceptable to meet the mission contamination requirements.

FLIGHT PANEL TESTS

The flight panels were tested in thermal vacuum at Northrop-Grumman over levels typical of standard LEO-GEO spacecraft solar panel temperature ranges. The panels were subjected to six thermal vacuum cycles. Extended temperature tests were performed at APL. The flight panels were cycled in vacuum over the -140°C to 240°C maximum operational temperature range. The 240°C simulated the short-duration panel temperature excursions when the spacecraft is over the Mercury sub-solar region. The high-temperature excursions are expected in a limited number of orbits around Mercury. The flight panels were not exposed to temperatures over the substrate material glassing temperatures, above 240°C, due to concerns over possible weakening of the substrate mechanical strength required for launch. Panel-level vibration and acoustic tests were not performed. Each assembled wing was vibration tested at APL. The panels were mounted on the spacecraft during spacecraft-level vibration and acoustic tests.

SOLAR PANELS FLIGHT PERFORMANCE

Following the successful launch of the MESSENGER spacecraft, two 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 indicated that the panels did not suffer any damage at launch and that the measured panel power agrees well with the LAPSS measured power at NGST. Fig. 4 shows the ground LAPSS data taken during final panel acceptance testing at Northrop-Grumman and the measured data of the two wings during the flight peak power tests.

Figure 5 shows the predicted and measured solar array panel temperatures since launch. The correlation is very good. The panel-Sun angle has been maintained nearly normal to date. The maximum temperature reached on the panel so far in the mission has been around 90 °C. Figure 6 shows the variations of the predicted peak power of the

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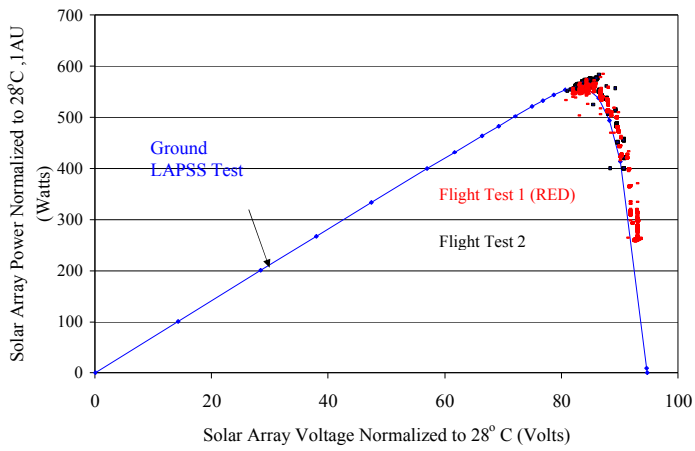

Fig. 4. Solar Panel Flight and Ground (LAPSS) Data

solar array and the actual load power since launch. The jumps in load power correspond to the spacecraft flips to point the sunshade toward the Sun or anti-Sun direction.

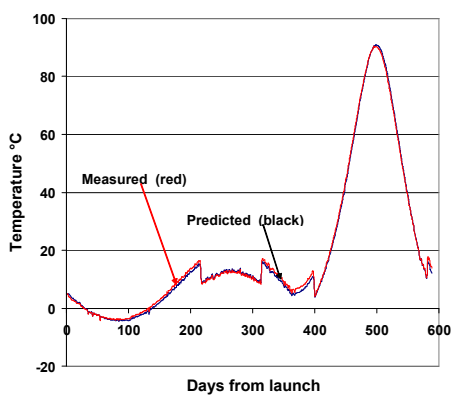

Fig. 5. Predicted and Measured Solar Panel Temperatures since Launch

SUMMARY

A solar array for a spacecraft mission to orbit Mercury is a challenging task. Extensive solar array development and testing were conducted to characterize and qualify the cell and panel design. The solar array design approach should be directly applicable to spacecraft missions that require high-temperature operation. The panels are operating as designed. The flight data to date indicate that the solar panels are generating the expected power and that they did not suffer any damage during launch. Flight data also indicate that the environmental degradation so far has been minimal.

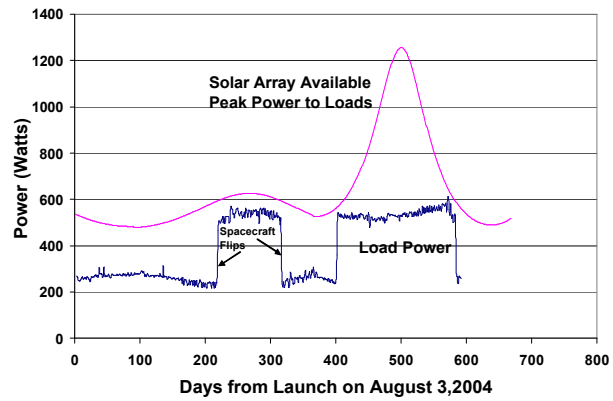

Fig. 6. Spacecraft Load Power and Available Peak Solar Array Power

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