PROTOTYPE SOLAR PANEL DEVELOPMENT AND TESTING FOR A MERCURY ORBITER SPACECRAFT

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ABSTRACT

A Mercury orbiting spacecraft imposes many design challenges in the area of spacecraft thermal control and electrical power generation. The Discovery Mission MESSENGER (MErcury Surface, Space, ENvironment, GEochemistry, and Ranging), being designed and built by The Johns Hopkins University Applied Physics Laboratory (APL), will orbit and survey the planet Mercury for one year. In order to reduce cost and schedule risk while increasing the probability for mission success, the MESSENGER solar arrays will be constructed from conventional "off the shelf" materials and technologies.

INTRODUCTION

This paper describes the high temperature and high intensity illumination testing which were used to thermally evaluate a series of prototype engineering solar panel concepts. Since all of the materials, construction techniques and technologies chosen for the solar panel construction are typically used for fairly benign thermal environments, qualification or proof of concept testing was necessary to assess the risks associated with such a thermally demanding mission. The solar array concept for the MESSENGER Mission consists of two double sided rotatable wings, allowing for tailored temperature control during power generation as solar distance decreases. One side of each array is fully packed with 5.5 mil single junction gallium arsenide (GaAs/Ge) cells. The opposite side is packed with a 70%/30% mixture of Optical Solar Reflectors (OSRs) and the same type of GaAs/Ge cells. Each side of each solar panel is designed to maximize power generation and minimize operating temperature for a given solar distance range. From launch until the spacecraft reaches about 0.60 Astronomical Units (AU), where the solar constant is about 2.8 greater than at Earth, the fully packed side of each array is responsible for power generation. Once inside of 0.60 AU, the array is flipped to the 30% cell side which takes advantage of the higher solar constant while maintaining the solar panel operating temperature below 150°C.

The solar panel structure uses sandwich construction comprised of high thermal-conductivity graphite-epoxy (Gr/Ep) face-skins and an aluminum-honeycomb core. The primary thermal environment that drives the MESSENGER prototype solar panel design and the dual-sided configuration is the high solar intensity condition experienced when at planet perihelion, where the solar constant is eleven times that at Earth. The mirrored side of the array safeguards against solar panel temperatures exceeding 250°C in the event of a direct Sun pointing anomaly when at Mercury perihelion. And, the dual sided design minimizes the solar panel overall size and mass by using the fully packed face at solar distances associated with the beginning of the mission when temperatures are very benign and the solar constant is low.

A large portion of the proof of concept thermal vacuum testing was accomplished using a custom designed high temperature infrared oven to test solar panel specimens between +300°C and –105°C. The oven has allowed for accurate and repeatable component testing while proving to be very reliable and cost effective. The more expensive and complicated high intensity solar illumination tests were done only after thorough infrared temperature cycle testing and high temperature soaks verified solar panel materials and construction. The high-illumination testing was done at NASA's John H. Glenn Research Center at Lewis Field in the Tank 6 high intensity vacuum chamber. These tests were used to verify the prototype solar panel thermal design and to demonstrate ability to predict the expected test panel temperature at a high intensity solar simulated environment. The paper describes the details and results of the prototype tests conducted along with plans for the flight solar panel development and test.

BACKGROUND

Mercury has never been explored by a remote orbiting spacecraft. The only man-made space probe to visit Mercury was Mariner 10. Built and launched in the United States, Mariner 10, a 3-axis stabilized solar powered spacecraft, has provided the only images and scientific exploration of Mercury. Using three flybys, Mariner 10 was able to map only about 45% of the planet surface during a one-year period between 1974–1975. The dark side flybys were all at planet aphelion and never near the sub-solar point. Mariner 10 was designed and tested to withstand a solar-only 5 Sun (one Sun is equal to the solar flux at 1 AU or 1365 W/m^2) environment, ignoring the intense omni-directional heat radiated from Mercury's surface on the Sun-lit side.

Due to severe mass restrictions and extremely harsh thermal environments, a Mercury orbiting spacecraft poses many engineering and operational challenges. MESSENGER is a 3-axis stabilized solar powered spacecraft using a high-performance all chemical propulsion system fully integrated into an all graphite-epoxy structure. The power system will utilize two low mass dual-sided solar array wings that can be rotated and flipped as necessary to control solar panel temperatures as the spacecraft approaches Mercury perihelion. The mission design uses a ballistic trajectory with multiple Venus and Mercury gravity assists. The MESSENGER spacecraft will eventually orbit Mercury for one Earth year (or four Mercurian years) and return a wealth of scientific data and complete planet coverage, something not accomplished by Mariner 10.

The MESSENGER mission was recently selected by NASA to be the eighth program in the highly successful series of the Discovery missions. Mission cost and launch vehicle choices are very constrained under NASA Discovery guidelines. The largest acceptable launch vehicle for a NASA Discovery mission is a Boeing Delta II 7925H-9.5 (Maximum Launch Mass=1066 kg). Driven by the 2700 meter per second mission Δ-velocity (ΔV) requirements, over one half of the launch mass is allocated to propellant. It quickly becomes apparent that due to the high ΔV nature of this mission, the spacecraft mass allocated to useful payload is limited. Spacecraft mass must be used with great discretion since the main purpose of the mission is to get maximum science return. Figure 1 illustrates the MESSENGER spacecraft. As shown, the dual-sided solar panel wings extend beyond the protective umbra created by the thermal shade, making the solar arrays the only critical component exposed to direct high intensity solar illumination. Designing the spacecraft for minimum mass will require the attention of all spacecraft sub-systems, and the seemingly vulnerable solar array is no exception. A mass saving benefit from the dual-sided approach is that the overall solar panel area is small because each face is optimized for power at the worst case solar distance and temperature. Each face of the dual sided solar array is designed to produce power at a cell operating temperature of 150°C or lower. As illustrated by Figure 2, the array is flipped from the fully packed side to the OSR side and back as the solar distance varies between approximately 0.45 and 0.60 AU. Figure 3 illustrates the variation in respective solar array face rotation angle as a function of solar distance as the array temperature is held at or below 150°C. The rotation angle is zero when the Sun line is perpendicular to the active face of the solar array.

Why Use a Dual-Sided Solar Array Concept?

The MESSENGER mission first baselined a single sided fully packed solar array using single junction GaAs/Ge cells bonded to graphite composite substrates. As illustrated by Figure 4, the fully packed solar array has to be rotated in excess of 75° to maintain the steady state solar array temperatures below the desired operating point of 150°C while at Mercury perihelion (11-Sun illumination condition). A major deficiency with this design is that a direct Sun pointing anomaly at the perihelion solar distance could easily cause the array to approach 400°C. Solar cell and Gr/Ep adhesives are only rated to maximum temperatures between 200 and 260°C.

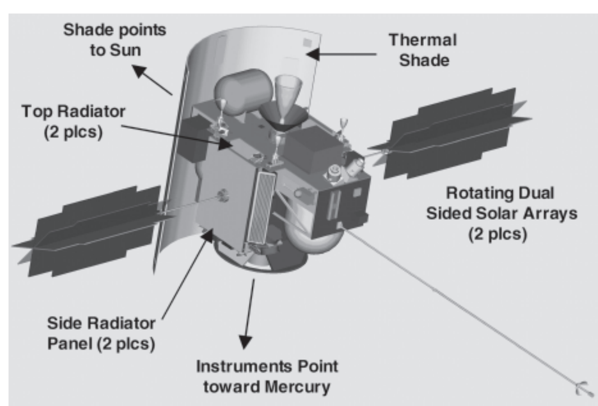

Figure 1. MESSENGER spacecraft with body mounted multi-layer insulation (MLI) removed.

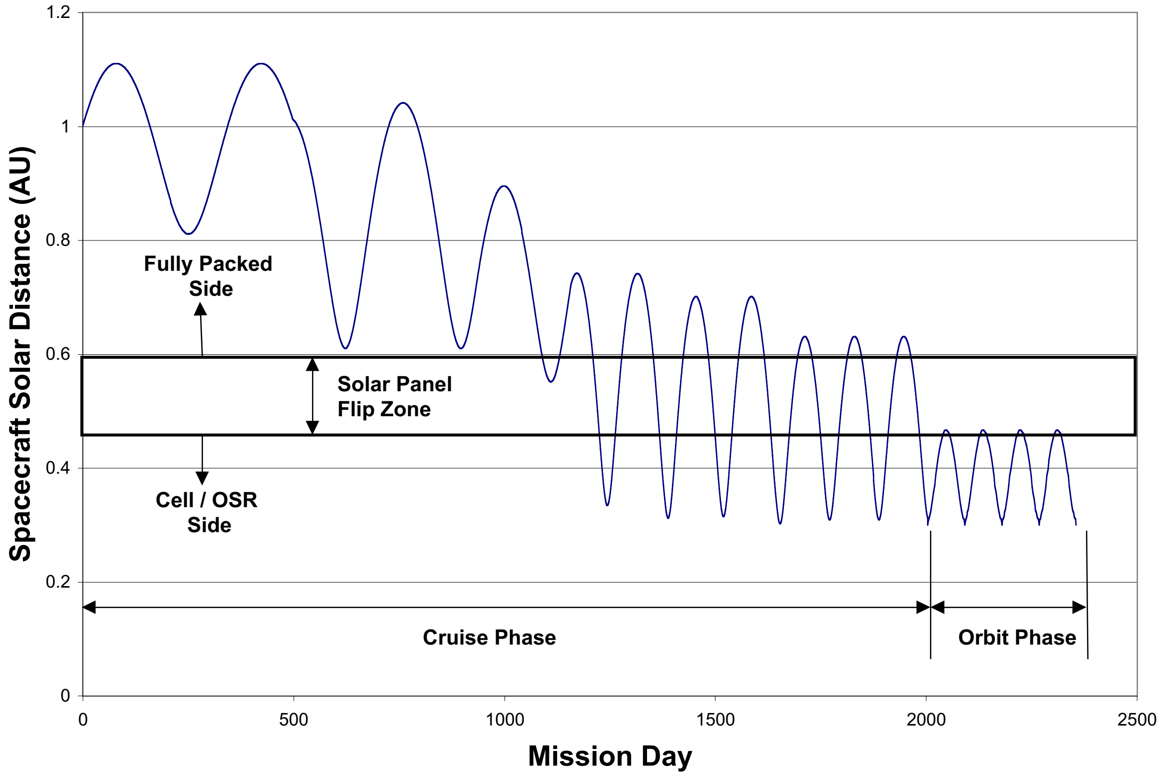

Figure 2. The MESSENGER mission profile. Cruise phase is five years and orbit phase is one year.

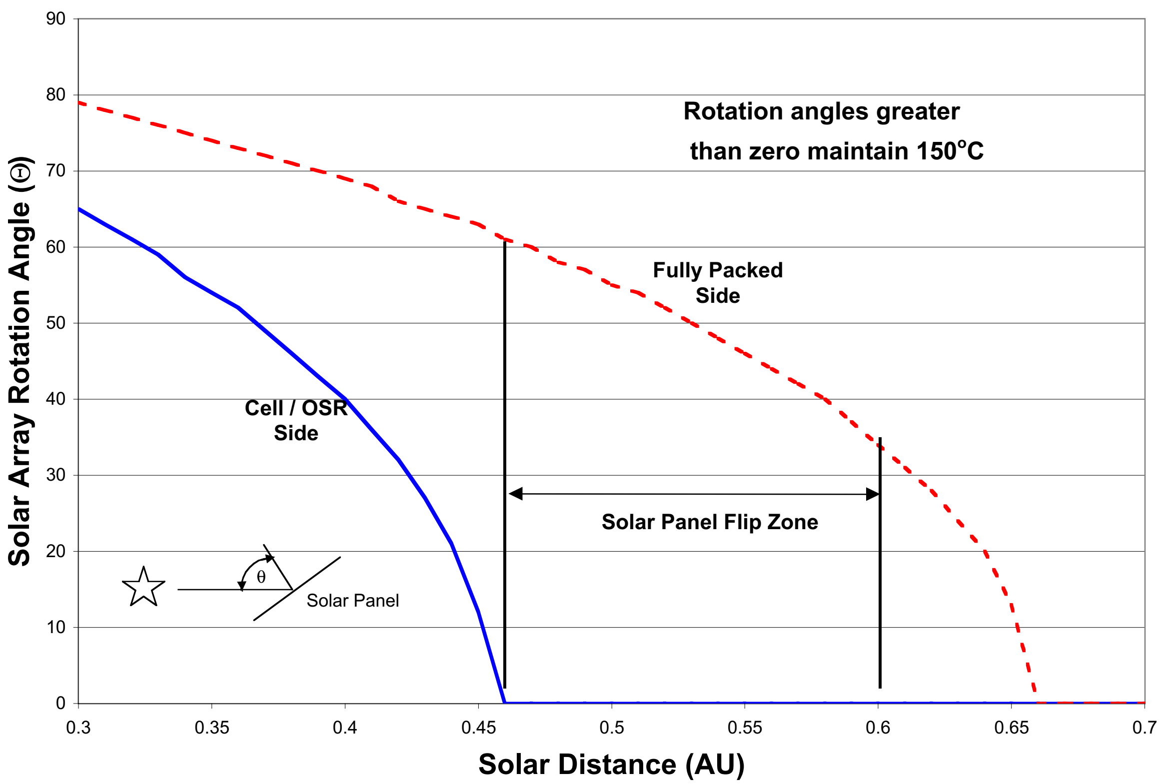

Figure 3. The solar panels are rotated to maintain the maximum temperatures at or below 150°C. The dual sided solar arrays have a wide flip zone to reduce operational constraints and minimize risks.

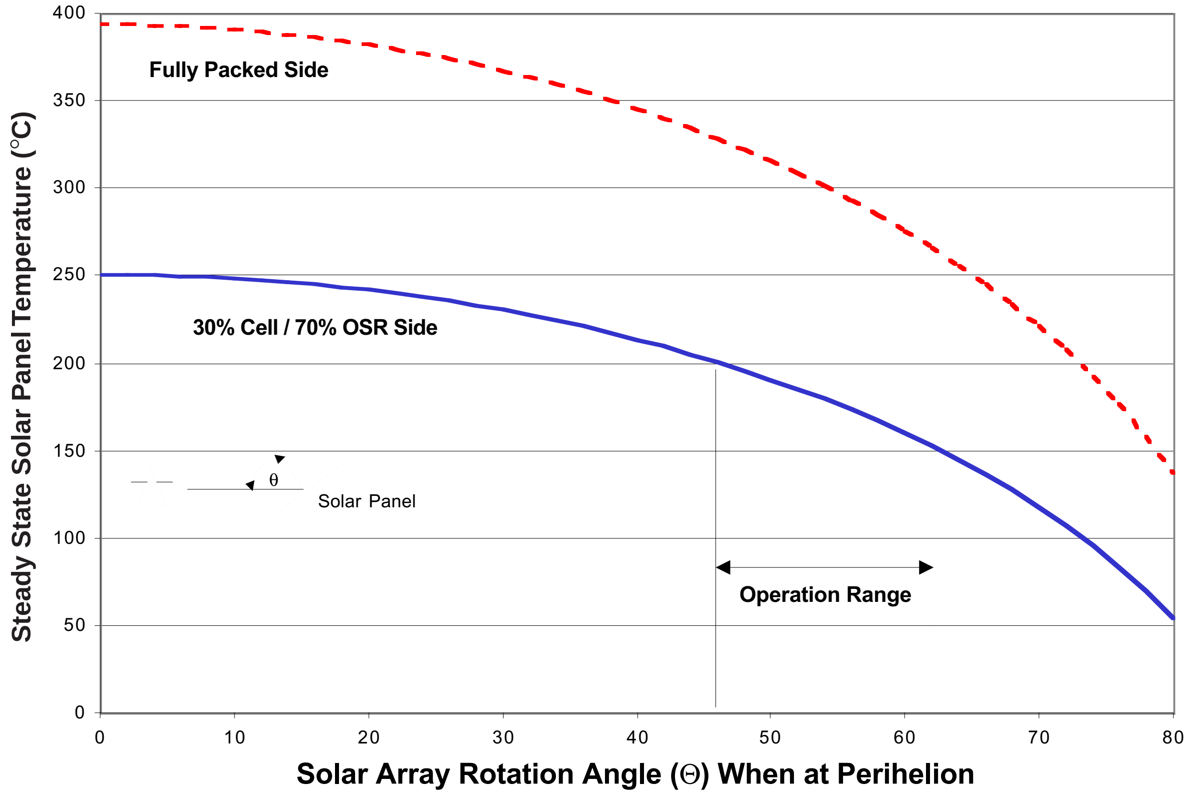

Figure 4. The OSR array side will maintain the steady state solar panel temperature below 260°C if Sun pointed when at perihelion. Note that the fully packed side could reach 400°C under the same condition.

Trying to qualify this technology to temperatures in excess of 400°C was not a feasible option. Another critical deficiency with the single sided concept is that it is thermally unstable. The analysis represented by Figure 4 illustrates that controlling the solar array temperature to 150°C while at perihelion could be very difficult because of the sensitivity between tilt angle and temperature. Also, at solar panel aspect angles greater than 65°, solar cell reflectivity begins to dominate power calculations and introduce large uncertainty.

Once the single sided concept was abandoned, a trade study was done to determine the effects of adding mirrors with the cells in order to reduce the overall solar panel temperature. Power system analysis showed that the total surface area necessary to do the complete mission using only a single solar array face that combined OSRs with solar cells would be almost a factor of four larger than the original fully packed design. The OSR design also has to incorporate thicker face skins to thermally connect the solar cells to the OSRs. Although the single solar array face that utilized OSRs with solar cells was not feasible from a spacecraft mass perspective, it was from a thermal perspective. The next trade study combined a fully packed solar array face with an OSR and solar cell face. This design would allow the MESSENGER power system to select the appropriate face depending on solar distance and solar panel temperature. Since each of the solar array faces will be designed to produce power over a defined solar distance range, the dual-sided approach allows for the panel area optimization because the worst case power condition will always dictate this parameter and set the maximum face size. This in conjunction with the deficiencies described for the single sided fully packed solar array made the dual-sided concept extremely attractive. More detailed power and thermal analysis were performed, and the dual-sided solar array became the baseline for the winning Discovery proposal, MESSENGER.

Prototype Solar Panel Construction

The prototype solar arrays were constructed of conventional materials that gave the highest probability of success for steady state temperature excursions between 200 and 260°C. The flight solar array concept calls for solar cells and OSR mirrors to be bonded to high conductivity graphite epoxy face sheets comprised of K13C2U cloth and RS3 and RS4 resin systems using conventional, NuSil Technology CV-2568, silicone adhesive, illustrated by Figure 5. The solar cells and cover

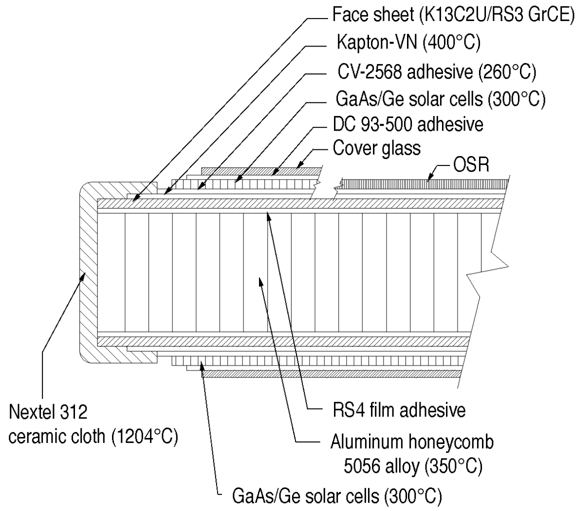

Figure 5. Solar panel cross-section showing maximum vendor recommended steady state temperatures. Solar panel materials and construction techniques were tested to temperatures in excess of 250°C.

glasses, Kapton™-VN, and aluminum honeycomb core as components are rated to temperatures at or above 300°C. The adhesives used to bond these different solar array components together creating the solar array assembly are the materials that have the lowest manufacturer's rated temperature. For example, Dow Corning rates DC-93500, which is used to bond cover glasses to solar cells, to only 200°C. Manufacturers of the resin systems RS3 and RS4, used to bond the individual face sheet plys together and to the honeycomb core, are only rated to 232°C. So it was decided to purchase and test multiple configurations of populated and unpopulated Gr/Ep sandwich panels that all used the same cyanide-ester resin systems and solar cell adhesives. Six different test panels were purchased and are listed in Table 1. Figures 6 and 7 depict the mirrored and non-mirrored sides of the prototype dual-sided solar array panel that was tested and is identified as Panel 4 in Table 1.

TECSTAR, Inc. did the solar cell and OSR bonding and the electrical wiring for all the test panels. The solar cells were TECSTAR 5.5 mil thick GaAs/Ge "standard" production cells cut to the required size. The solar cell

Table 1. Six prototype solar panels were purchased for testing under MESSENGER thermal environments. The prototype panels were fabricated to prove that conventional solar panel construction techniques and materials could be used in the high-temperature and vacuum environment an orbiting spacecraft would experience at Mercury.

Generic Panel Identifier	Configuration Description	Construction Details
Panel 0	Unpopulated Test Panel #1	A Kapton-insulated substrate with inserts. Face sheet composition is M55J/RS3 2.5mil UDPP with RS4 sandwich film adhesive and 3.1 lb/ft^3 hexcell core.
Panel 0A	Unpopulated Test Panel #2	A Kapton-insulated substrate without inserts (UDPP K13 C2U/ RS-3, 2.5 mil, with 4.5 lb/ft^3 hexcell core.) Panel 0A is populated on one side with individually wired heater elements which are arranged as the cells and OSRs of panels 3 & 4 (below). The other side of the panels is populated by Silver Teflon to emulate OSRs.
Panel 1	Single Sided Populated Panel #1	A substrate identical to Panel 0 populated with 12 older technology GaAs/Ge "reject" cells. Cells are mechanically sound, though electrically poor. Cell size is larger than intended for MESSENGER flight.
Panel 2	Single Sided Populated Panel #2	A substrate identical to Panel 0 populated with 36 high-temperature GaAs/Ge cells and representative OSRs. The cells are larger than what will be defined for the flight panels. OSRs were placed on the panel to qualify materials and methods, and as such are not positioned to be of thermal use.
Panel 3	Dual Sided Configuration Panel #1	A substrate identical to Panel 0A is populated on a single side with 9 high-temperature GaAs/Ge cells from the Panel 2 production lot which are cut down to 4×2-cm. OSRs are cut and arranged with the cells to be representative of the flight thermo-optical design. Each string and individual corner cells of strings A and C are wired for electrical monitoring. Thermocouples are bonded between two cells each of circuits A and C and between two OSRs each of the exterior OSR strings.
Panel 4	Dual Sided Configuration Panel #2	A substrate identical to Panel 0A populated on the front side identically to Panel 3. The backside is populated with 2 by 4 cm cells cut from the Panel 2 production lot. The backside panel is also instrumented. With two flight-candidate platinum thermistors.

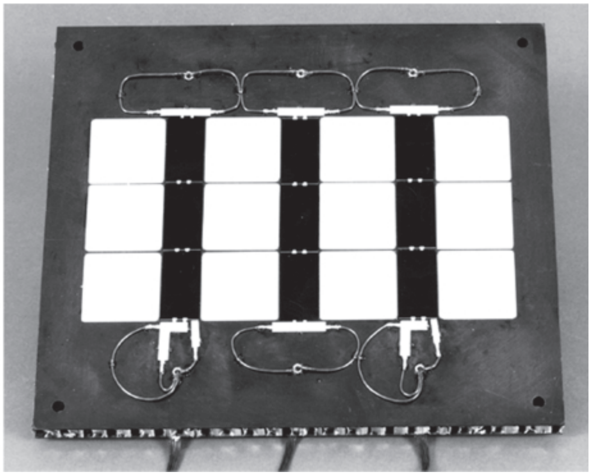

Figure 6. The prototype 30% solar cell and 70% OSR face of the dual sided array.

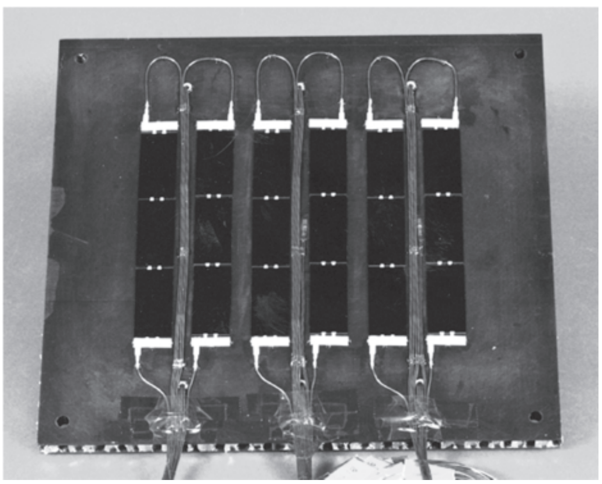

Figure 7. The prototype fully populated face of the dual sided array.

interconnects end termination were all parallel gap welded and high temperature Kapton was co-cured with the GrEp to form the electrical barrier between the solar cells and the graphite panel. Ceria doped cover glass, three tenth of a millimeter thick, coated with dual anti-reflective coating was bonded to the cell using DC 93500 adhesive. NuSil 2568 adhesive was used to bond the cells and OSR to the kapton insulation on the panels.

High-Temperature Test Program Development

Development of a test program and testing philosophy was essential to verify that conventional solar array materials and manufacturing processes could be used for high temperature applications. Thermal vacuum testing of the samples would have to represent the minimum and maximum temperature excursions experienced during the expected 65-minute eclipse as well as the nominal solar panel operating temperatures when illuminated. Also, because the solar arrays represent a single point failure, the worst case steady state survival temperature would also be tested to show design robustness in the event of a Sun pointing anomaly when at Mercury perihelion.

Each prototype test panel would first be baked out for 24 hours at 80°C. This bake-out helps to remove trapped air pockets in the solar cell adhesives that could rupture at the high temperatures and damage the cells. The test program combines solar panel acceptance testing that is typical for an Earth orbiting spacecraft with the higher temperature cycle and soak testing required for a Mercury orbiting spacecraft. Electrical performance tests would be done after each set of cycle tests and after each of the high temperature soaks to verify electrical functionality and document any temperature incurred damage.

Six conventional workmanship cycles would be performed between +120°C and –105°C, establishing a test baseline. High temperature cycling would follow the workmanship cycles and would have maximum plateaus at +150°C, +180°C and +200°C and the same lower temperature of –105°C. Also, high temperature soaks representing steady state survivability in the event of a Sun pointing anomaly at perihelion had to be demonstrated. Each of the soak temperatures, 230°C, 240°C and 250°C, would be held for a minimum of one hour to verify material and construction robustness, removing any solar array recovery time constraint if an anomaly were to occur. At the completion of each level of cycle testing and after each of the high temperature soaks the test panel would be removed from the test chamber and visually inspected for damage such as substrate delamination or cracked cells or OSRs. After the visual inspection, the test panel would be taken to a laboratory for electrical performance testing if necessary. Once the inspections and electrical test is complete the prototype solar panel would be re-installed for further temperature cycle testing.

The first high temperature tests were done using a large (9 m^3) liquid nitrogen (LN_2) cooled vacuum chamber and infrared (IR) heat lamps. A dummy black aluminum test panel, which represented a typical prototype solar panel, was located inside the chamber and heated at high intensity with the IR lamps. Thermocouple measurements made on the aluminum panel showed temperature gradients in excess of 70°C when the maximum dummy panel temperature was 310°C with the chamber walls at –180°C. A few failed attempts were made to increase the flux uniformity by repositioning the IR lamps and dummy test panel. It quickly became evident that this test setup method would impose a very high risk to the sample being tested. Also, because the test samples had to be removed for inspection after each set of thermal cycles, the test setup proved to be ineffective from both cost and schedule standpoints and therefore was abandoned.

A smaller and simpler test apparatus had to be designed. The new test apparatus had to deliver uniform sample temperatures between –180 and 300°C, it had to be time and cost effective, it had to be reliable, and it had to be simple to operate. The new apparatus would be designed to operate inside the large 9-m^3 chamber so that vacuum, liquid nitrogen, thermocouple and electrical interface resources could be easily utilized. The large chamber would not have to be temperature controlled, eliminating the long transitions from vacuum to ambient conditions. The small (0.125 m^3) test apparatus, known as the "E-Box", is shown in Figure 8. The E-Box is thermally isolated from the large chamber, the east chamber, illustrated by Figure 9, using stainless steel chains and high-temperature multi-layer insulation allowing for independent temperature control. Thirteen 1000-watt resistive OMEGALUX™ strip heaters combined with an LN_2 cooling loop plumbed from an internal east chamber LN_2 feed will provide the necessary temperature control capability for the E-Box. The E-Box has two 2600 cm^2 (400 in^2) compartments that are identically heated and cooled, allowing for multiple sample testing.

Figure 9. The E-box in test configuration installed in the East chamber. Vacuum and high temperature kapton MLI insulate the E-Box from the room temperature walls of the East chamber.

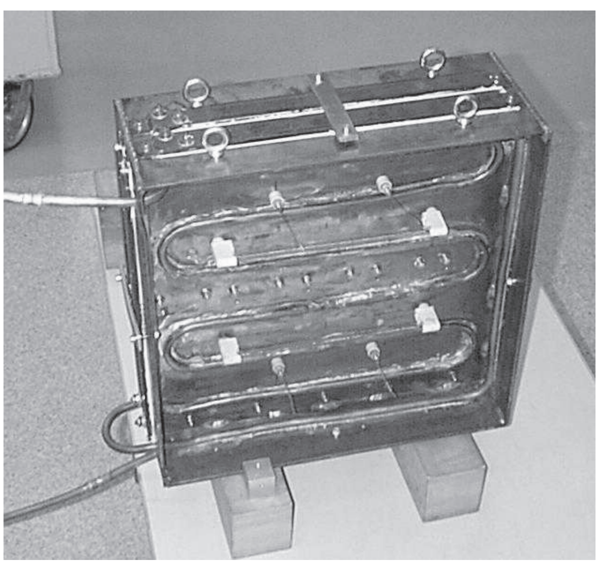

Figure 8. The E-box prior to large chamber installation. The E-box is liquid nitrogen cooled, and heated using 13 one kilowatt strip heaters.

Infrared Testing using the E-Box

Pathfinder infrared thermal vacuum testing in the E-Box started on April 23, 1998 using two unpopulated GrEp test panels. One test panel, identified as Panel0, was constructed from M55J fibers formed into two 0.25mm substrates using RS3 and co-cured with aluminum honeycomb as a sandwich using RS4. The second test panel, identified as Panel0A, was constructed from K13C2U fibers formed into two 1.0 mm face sheets using RS3 and co-cured with aluminum honeycomb as a sandwich using RS4. The pathfinder testing was used to verify the survivability of the RS3 and RS4 resin systems at extended high temperature before the expensive populated panels were subject to the same sequence of tests. The test procedure, which includes chamber breaks, inspections and E-Box power levels, was also refined to minimize the overall test time.

The low conductivity test panel, Panel0, shown in Figure 10, was extensively thermocoupled to measure the temperature gradients induced by heating and cooling. Since the Panel0 face sheets have very poor thermal conduction characteristics, temperature gradients measured between any two thermocouples illustrated quantitatively the heating uniformity. With an average temperature of 250°C for each test panel, the maximum differential measured between the warmest and coolest thermocouple on Panel0 was less than 5°C. The pathfinder thermal vacuum testing with the unpopulated test panels verified that the cyanide-ester resin systems could be taken to temperatures above the manufacturer's specification for extended time without damage or delamination of the sandwich construction. The pathfinder testing also verified that the heating and cooling of the test samples was very uniform and low risk while the turnaround time between cycle testing, hardware inspections and reinstallation of the sample in the E-Box was small.

Upon the satisfactory completion of the pathfinder testing, solar panel prototypes populated with mechanically-sound, electrically-rejected, but functional, GaAs/Ge cells and OSRs were then tested using the same thermal profile and inspection procedures established during the pathfinder tests. Table 2 lists a summary of the results of all the panel level infrared testing performed at APL. Figure 11 depicts the as-run thermal vacuum test data for Panel3 and Panel4, representing the prototype dual sided solar panel configuration.

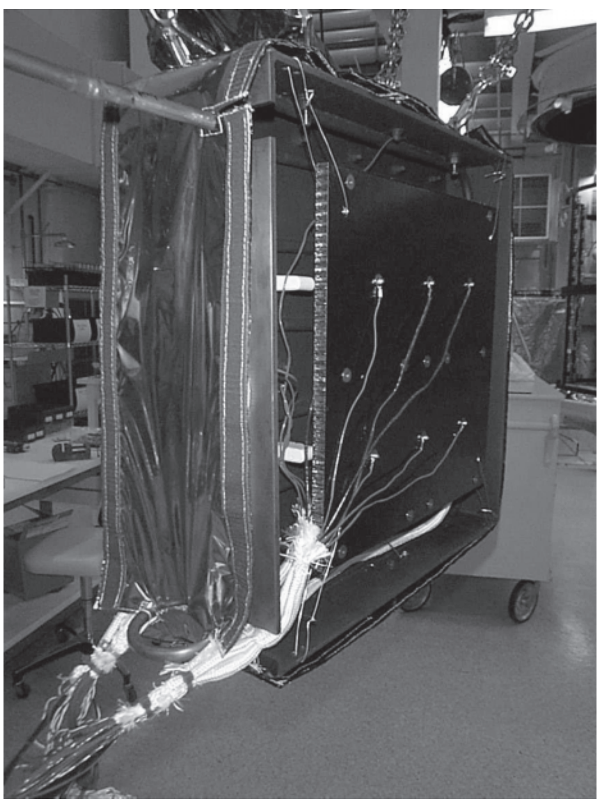

Figure 10. The E-Box being used to test Panel0 and Panel0A. Panel0 is visible in the right chamber with Panel0A (not seen) installed in the left chamber.

Simulated High-Intensity Solar Illumination Testing

Once the dual sided prototype engineering panels were successfully temperature cycled and soaked using the E-Box, Panel3 was taken to the John H. Glenn Research Center (GRC) at Lewis Field Tank 6 thermal vacuum test facility. The Tank 6 facility is a 18.5-meter long by 7.7 meter diameter horizontal vacuum chamber with a liquid nitrogen cryogenic-wall and a solar simulator. The solar simulator uses nine 30 kW Xenon arc lamps and was designed to provide no less than 1 Sun on a 4.6 meter diameter target, 17.4 meters from the source with a subtended angle of less than 1°. In order to achieve the high intensity solar flux levels required to simulate Mercury perihelion, the targets were located at a distance of approximately 5.8 meters from the source. The vacuum system can provide ambient pressures as low as 10^{-6} torr. Tests and analysis done at APL prior to GRC solar testing indicated that heat transfer concerns are mitigated at pressures below about 150 millitorr and that 10 Suns with a room-temperature sink is equivalent to 11 Suns with a liquid nitrogen or deep space sink. Therefore all testing was conducted without high-vacuum pumps and liquid nitrogen cooling, resulting in a nominal tank pressure of 25 millitorr and an average ambient tank temperature of 22°C.

The solar array panel was installed, aligned and electrically checked on Monday, February 22, 1999. The vacuum system was started at about 10:00 am on February 23. The tank pressure was 28 millitorr at 12:49 pm when the first solar simulator lamp was started. Video and infrared cameras were mounted at window ports near the simulator window to observe the test article. A total of eight lamps were started sequentially in the following order: 5, 4, 6, 7, 2, 8, 9, 1. The lamp startup order was determined by change coupled device (CCD) imaging on a Teflon sheet prior to test panel insertion and was selected to minimize the initial flux variation on the test panel. Lamp number 3, which was planned to be used after lamp 6, failed to start and required the addition of lamp 1 as an alternative. After each lamp was started and adjusted to its operating level, solar cell I-V curves were acquired. Panel temperatures were allowed to stabilize with each lamp operation.

Table 2. MESSENGER thermal/vacuum testing at APL.

Test Report	Date	Test Item	Test Type	Test case Summary
97-032	12/18-22, 1997	Solar Panel Test Plate (Aluminum)	Engineering Development	1 Cycle (-120°C <> +310°C)
98-013	4/23–5/14, 1998	Panel0 and Panel0A Panel1 and Panel2	Thermal/Vacuum Performance	Bakeout 85°C 6 Cycles (-105°C <> +115°C) 8 Cycles (-105°C <> +150°C) 4 Cycles (-105°C <> +180°C) 4 Cycles (-105°C <> +200°C) 4 Cycles (+23°C <> 220,230, 240, 250°C)
98-023	5/22–26, 1998	Panel1	Electrical Performance T/V (Illumination)	7 Cases (+50°C <> 200°C)
98-033	8/5–24, 1998	Panel2	Long Duration T/V Test	1 Case (+23°C <> +180°C)
99-004	1/27–2/5, 1999	Panel3 and Panel4	T/V Performance Test	Bakeout 85°C 6 Cycles (-105°C <>115°C) 4 Cycles (-105°C <>150°C) 4 Cycles (-105°C <> 180°C) 4 Cycles (-105°C <> 200°C) 2 Cycles (+23°C <>230, 250°C) 6 Cycles (-105°C <>115°C)
99-008	2/8–9, 1999	Panel2	T/V Performance	1 Cycle (-100°C <> 260°C)
99-010	2/10-12, 1999	Panel3 and Panel4	Electrical Performance T/V (Electrical Continuity at High Temperature)	Bakeout +100°C Three Hour Soak @ 250°C
99-017	3/1-4, 1999	Panel2	T/V Performance	Bakeout 100°C 1 Cycle (+100°C <> 280°C) 2 Cycles (+23°C <> 300,*320°C)

*Test Panel Failed

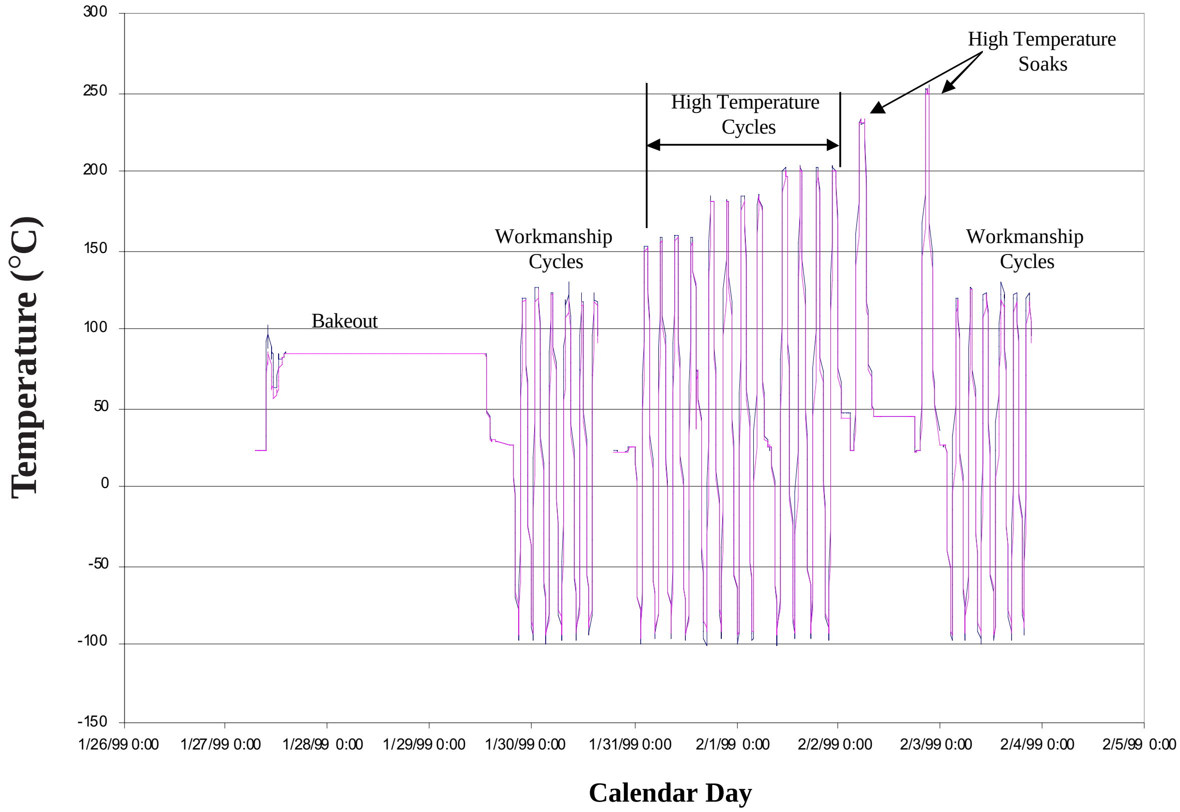

Figure 11. The actual test profile as run for Panel3 and Panel4, the dual sided prototype solar panels.

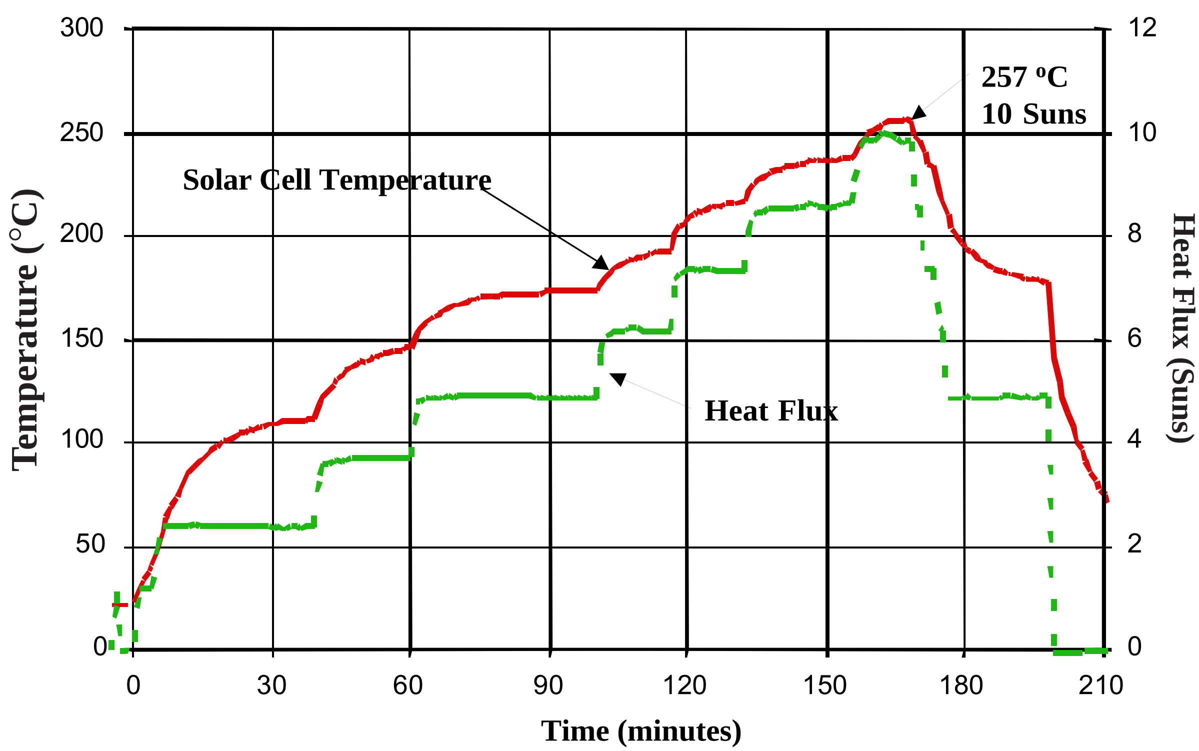

Figure 12. The results of the high-intensity illumination testing at GRC. The data represents cell temperature and simulated solar flux.

Figure 12 shows the maximum measured panel temperature and estimated local solar flux at that thermocouple location over the 200-minute illumination period. With four lamps operating, the maximum-recorded temperature was 174°C with a localized flux of 4.9 Suns. The eight-lamp configuration resulted in a maximum panel temperature of 257°C with a localized flux of 10.0 Suns. Both test points provided excellent agreement with pre-test analyses. No visible or electrical abnormalities were observed during the test. Post test removal and inspection of the panel showed no visible effects of the testing. Darkening of some non-optical adhesives occurred but without effect to adhesion. Pre-test and post test I-V characterizations performed at GRC's Solar Cell Measurements Lab with a Spectrolab X-25 were essentially identical.

CONCLUSIONS

The APL prototype test program met the goals of thermally evaluating the materials and processes used for conventional solar array construction. The MESSENGER prototype dual sided solar array has proven to be thermally robust and temperature tolerant. Results of the thermal testing presented verifies that the conventional adhesives typically used for the construction of cyanide-ester based GrEp solar panel substrates and solar cell bonding processes are not adversely effected by temperatures below 300°C. As illustrated, multiple solar panel samples using two different pitch based fibers passed all required cycle and dwell testing, with one sample, Panel2, being tested to high temperature failure (>310°C). The thermal modeling which was used to establish the maximum temperature limits during E-Box testing (250°C) was confirmed during the solar illumination testing at GRC (257°C). And the comprehensive prototype test program has verified that the actual maximum panel temperature is within the maximum survivable limits of the chosen materials and construction techniques by over 40°C. The infrared thermal vacuum testing with the E-Box provided inexpensive and conclusive test data regarding the integrity of the materials and construction techniques chosen for the engineering panels, avoiding the more complicated and expensive solar simulation testing until design concepts were well defined.

The prototype tests to date were intended to demonstrate that the basic material and processes used for a "standard" near Earth type solar array can also be used for a solar powered spacecraft in Mercury orbit. Tests to verify the effects of eclipse induced thermal shock, solar cell reverse biasing when shadowed during high intensity illumination, and adhesive darkening during high-temperature/high-intensity UV illumination were not conducted, but are presently being evaluated. Efforts are currently in progress to evaluate the outgassing properties of the candidate adhesives at high temperature. Also under consideration is the use of IR and UV reflective coatings, applied to solar cell cover glasses and tested at elevated temperature and with high intensity UV illumination. Although the prototype solar panel testing to date has been weighed heavily toward thermal design and panel survivability, future testing will focus on refining the flight panel electrical design, assessing long term exposure effects on solar array materials and coatings and determining solar array performance under the MESSENGER mission environments.

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ACRONYM LIST

APL	The Johns Hopkins University Applied Physics Laboratory
CCD	Charge coupled device
GRC	Glenn Research Center
IR	Infrared radiation
LN_2	Liquid nitrogen
MESSENGER	MErcury Surface, Space, ENvironment, GEochemistry, and Ranging
MLI	Multi-layer insulation
OSR	Optical Solar Reflector
UV	Ultra-Violet

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