# THE STARDUST SOLAR ARRAY

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## ABSTRACT

The Stardust satellite, launched in February 1999, is a seven-year mission to gather material from the comet Wild-2 and return the material to earth in January 2006. The satellite will orbit the sun, traversing distances from a little under 1 AU to 2.72 AU. On April 18th 2002, the spacecraft reached its furthest distance and broke the record for being the farthest spacecraft from the sun powered by solar energy. This paper describes the analysis, design, fabrication, and flight data of the Stardust solar panels. Telemetry data from the solar panels at 2.72 AU are in excellent agreement with flight prediction.

## INTRODUCTION

On February 7, 1999 the Stardust spacecraft was launched. The Stardust Program is part of NASA's Discovery Missions, which are low-cost solar system exploration missions with highly focused science goals. The Lockheed Martin Space Systems Company (LMSSC) built the Stardust Spacecraft. The Jet Propulsion Laboratory (JPL) manages the Program. Stardust's seven-year mission is to gather interstellar dust and material from the comet Wild-2 and return the material to earth. In order to intersect the comet in January 2004, the spacecraft (see Figure 1) must make three loops around the sun for an encounter with Wild-2 at a distance of 1.86 Astronomical Units (AU). The spacecraft will then orbit the sun one more time before dropping off a probe with the samples to earth in January 2006. The spacecraft in its seven-year mission will go through three aphelion points. The first aphelion point is at a distance of 2.19 AU. The second aphelion point is at 2.72 AU, and the third aphelion point is at 2.68 AU. On April 18th 2002, the Stardust spacecraft reached its furthest distance (2nd aphelion) from the sun at 2.72 AU setting the record for being the farthest satellite from the sun powered by solar energy

## ELECTRICAL POWER SYSTEM SELECTION

The Electrical Power Subsystem (EPS) and solar array design considerations are very unique for the Stardust Mission. Not only are these subsystems required to operate

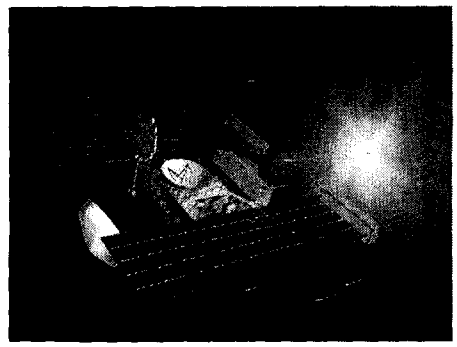

**Fig 1** Artist rendition of Stardust Satellite at Comet Encounter

in the deep space environment, they also must operate near earth, as the spacecraft will be returning to earth. Solar intensities on the array will vary from a little over 1 sun at the end of the mission, 0.135 suns at second aphelion, and 0.289 suns at encounter when the most power is needed. In addition, the solar array will see great swings in temperature from –61°C at 2nd aphelion to +79°C at the end of mission. At near earth, a solar cell will deliver the most current and the least voltage. Conversely, at the aphelions and at comet encounter, a solar cell will put out the least current and the most voltage. Two major issues come up in considering the design of the solar array and EPS. The first is solar cell operation. At the aphelions and at the encounter, the solar cells will be required to operate under low intensity and low temperature (LILT) conditions. Studies in the past have shown that solar cells may show severe performance degradation under LILT conditions [1]. Under LILT, the solar cells can experience parasitic contact/junction and shunt resistance losses. The second issue is EPS selection. A direct energy transfer system (DET) is lightweight, inexpensive, and has very little parasitic power losses.

However, a DET system will result in a very large solar array. Solar cell strings (cells in series) will have to be oversized to accommodate near earth voltage requirements. The number of strings will have to be oversized to meet deep space power requirements. Given the criteria of the Discovery Missions, the preceding conflicting requirements had to also be met with a low cost, low risk solution.

Three solar cells were traded, standard silicon, high efficiency silicon (HES), and single junction gallium arsenide (GaAs)[1]. Standard silicon solar cells under LILT conditions have shown poor performance in the past [1]. Performance under LILT conditions for standard silicon solar cells at 2.7 AU is shown in Figure 2. In addition, the standard silicon solar cells were less efficient than the other two solar cells. Preliminary LILT and radiation testing on Sharp HES solar cells showed good performance. The passivation layers in the HES solar cell protect them from parasitic contact/junction losses and provide for high shunt resistance. Figure 3 shows the performance of HES, and GaAs solar cells at various Stardust mission phases. Surprisingly, the Sharp HES solar cell showed significantly higher efficiency at 2nd aphelion than the GaAs solar cell. The reason is the large temperature coefficient of the HES solar cell. Because of the better performance, lower weight and significantly lower cost than GaAs solar cells, off-the-shelf Sharp HES 10 Ohm-cm solar cells were selected for the mission.

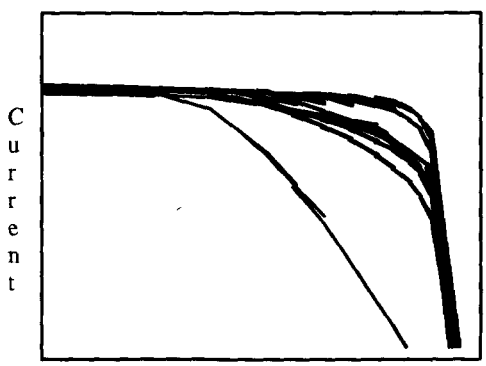


**Fig. 2** Standard silicon solar cells at 2.7 AU

In order to meet the conflicting requirement for the solar array to operate at 1 AM0, moderate temperatures and LILT conditions, an electrical power system (EPS) trade was performed. A DET system was traded with a peak power tracking (PPT) EPS. In order to meet all mission power requirements with a standard DET system, an approximately 11.1 $m^2$ solar array would be needed. The total allotted area and mass for the solar array allowed it to be no more than 6.7 $m^2$. A PPT system offered the advantage of pulling off the maximum power at the various mission phases. However, a PPT system has disadvantages in weight, cost, and parasitic power losses. Also, for this application, the PPT system would require a brand new design, which would entail greater cost and more importantly greater schedule risk. In a rendezvous mission, schedule is of paramount importance. The creative design solution to this problem was to use a Solar Array Switching Unit (SASU[2]) box with the DET system. This solution takes advantage of the fact that the voltage of the solar cells varies by almost a factor of 2 from the highest to lowest voltage. Solar cell strings of approximately the same length are laid out on the solar panels. The string length is long enough to meet the voltage requirements of the deep space phases of the mission. During the other phases, the string lengths are doubled (total number of strings halved) via the SASU. String length was optimized for comet encounter. The longer string lengths meet the Voltage requirements of the higher temperatures. Figure 4 shows a comparison of power delivered over mission time with a SASU in the architecture and without a SASU. As can be seen, the SASU not only results in meeting mission requirements at 2nd and 3rd aphelion, it also results in having margin at comet encounter.

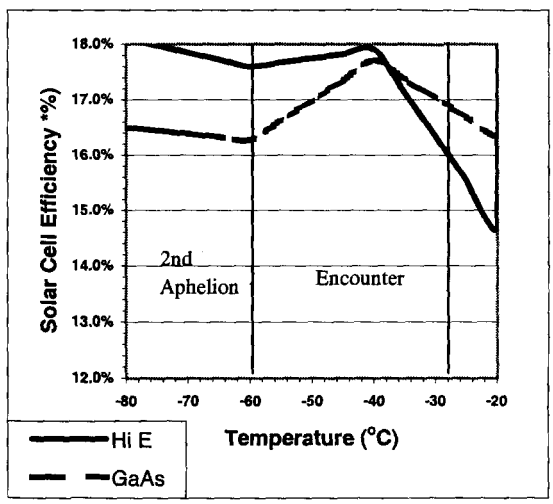


**Fig. 3** HES and GaAs solar cells at various mission phases

## SOLAR CELL CHARACTERIZATION

The Sharp HES 10 Ohm-cm solar cell is nominally 16.3% efficient at AM0, 28°C. When this solar cell was selected it was fully qualified and had been in production for some time. The solar cell is characterized by a textured front surface to increase light absorption, a back surface field reflector, and passivation layers. A 152-micron microsheet coverglass with a magnesium fluoride antireflective coating was attached to the solar cell. Because the solar array is mainly sized for deep space, blue-red reflecting covers that are typically used with this solar cell were not needed.

Once the solar cell was selected, a complete characterization of the solar cell performance under LILT and radiation conditions was performed at JPL. Fourteen solar cells were characterized at 3 radiation fluences (0, 7.2E13 e/cm2, 3.5E14 e/cm2), four sun intensities (1.0, 0.289, 0.208, and 0.139), and 6 temperatures (28°C, 0°C, -28°C, -39°C, -60°C,

[1] This work was performed in 1996. At that time multijunction solar cells were not mature and were therefore not considered.

[2] United States Patent number 6,211,650

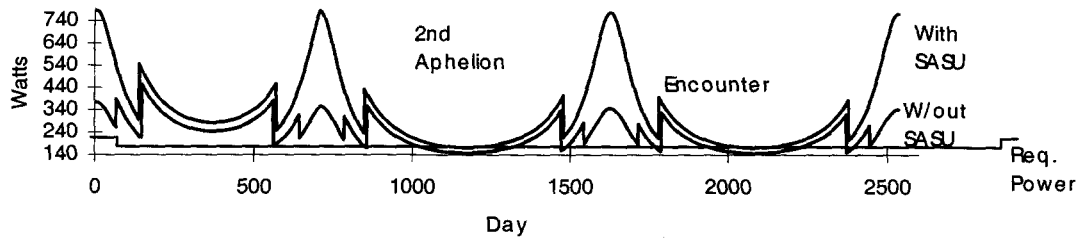


**Fig. 4** Power produced with a SASU and without a SASU

and –80°C). The three radiation levels approximate beginning of life, 1st aphelion, and encounter. Table 1 shows all pertinent data. At LILT conditions the table shows that for all radiation levels, the solar cell performance is well behaved. There is a significant improvement in fill factor and efficiency at the LILT conditions. Figure 5 shows the I-V curves at 0.139 suns at –80°C for the three fluences. As can be seen, the I-V curves are well behaved and show no evidence of Schottky contact problems near the open circuit voltage.

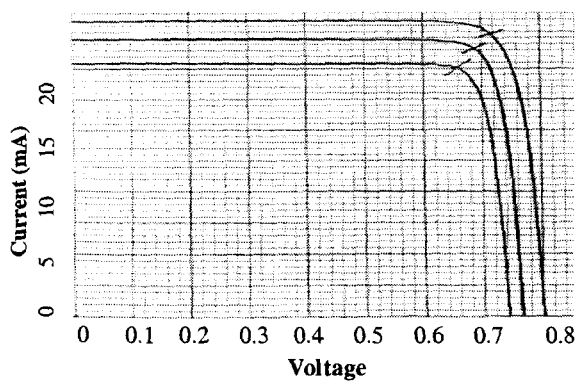


Fig. 5 I-V curves at 0.139 suns, -80°C, 0, 7.2E13, 3.5E14 e/cm$^2$

## SOLAR PANEL DESIGN, FABRICATION & TEST

The Stardust solar array consists of two wings. Each wing has 3 solar panels, a forward panel, a midboard panel, and an aft panel. Two sizes of solar cells were used, 4x6 cm$^2$ and a 2x2 cm$^2$. The 2x2 cm$^2$ solar cells were used to trickle charge the batteries. The design of the solar panels was straightforward using standard industry practices. The solar cells were purchased from Sharp as CICs (cover-integrated-cell), meaning the solar cells arrived with 1 mil silver foil interconnects welded on the front and the coverglass bonded in place. The 1 mil silver mesh interconnects was welded to the top of the solar cell using Sharp's standard flight heritage welding process. The solar cells were interconnected into solar cell strings using standard flight proven LMSSC soldering processes. The solar cell strings were bonded to the Graphite Epoxy substrates using standard aerospace approved RTV silicones and standard vacuum bag bonding. Because of the possibility of shadows from the comet shields used to protect the panels from comet particles during unplanned maneuvers, bypass diodes were installed on the solar cell strings. A qualification panel was built and went through a qualification acoustic and thermal vacuum test. The flight panels went through the same test sequence except at acceptance levels. In addition, the flight panels were thermal vacuum baked to remove all volatile condensables.

## COMPARISON OF PANEL PREDICTS TO FLIGHT DATA

A comparison was made of the predicted power output at 2$^{nd}$ aphelion and the telemetry from the spacecraft. Telemetry from the vehicle consists of a short circuit current sensor, and an open circuit voltage sensor (both having 3 solar cells in series), current from all circuits, solar array usage, battery voltage, solar panel temperature, and solar array off angle. Two types of predict are discussed. The first is the baseline predict. This was the predict used in the design of the solar array. This prediction is not based on ground test solar panel electrical performance data of the array. Conservative losses for radiation, UV darkening (2%), Off-Angle (15°), solar cell mismatch (2%), and temperature prediction was used in the baseline predict. The radiation environment used was the JPL SPE 91 Flare environment [2] with a 99% Confidence Level. This is a very conservative flare model. Power for the Array is predicted at the SASU. The baseline predict used the solar cell equation to model LILT conditions [1]. A modified prediction model was run based on ground test solar panel electrical performance data, temperature telemetry, off angle telemetry, a more benign UV darkening (1%), and exact radiation data. Adjusted Flare Proton data from the NASA GOES meteorological satellite from the time of launch until 2$^{nd}$ aphelion encounter was used to calculate an equivalent 1 MeV electron environment. Adjustments were made to the GOES data to account for the distance of the spacecraft from 1 AU. The GOES data was downloaded off their website. The modified predict used the Brown solar cell model [1] to calculate I-V parameters.

Table 2 shows the results of the comparison. As can be seen from the Table, the actual radiation environment is almost four times less than the design environment. The actual solar panel temperature is colder than predicted. All electrical parameters for the modified prediction are significantly

higher than the baseline predict. The difference for the open circuit voltage sensor is due to the actual radiation environment being more benign than predicted (2.3%) and the panel being colder than predicted (2.7%). The main reasons for the short circuit current modified predict being higher is the lower radiation degradation (~2%), lower off pointing (~2%), lower UV degradation (1%), and temperature difference (0.5%) compared to the baseline prediction. The modified predict for the array is higher than the baseline predict by almost 10%. At this point in the mission, there is around 7 to 10 volts of margin at the max power point so differences in the two predictions are all coming on the current side of the curve. Again, differences between the two are similar to the Isc sensor, that is off pointing (2%), Radiation (2%), Ultraviolet Darkening (1%) and Temperature (0.5%). In addition, the baseline predict had a 2.1% mismatch loss whereas the Modified Predict is based on solar panel electrical performance where mismatch is accounted for in the test data. Accounting for all these additional losses brings the modified predict and baseline predict to within 2%. The 2% difference is due to the difference in the solar cell models used. The error column in Table 2 compares the modified prediction to the telemetry. The modified prediction is in excellent agreement with the telemetry with the exception of the Isc sensor, which is around 3.6% higher than expected. The difference here may be due to either the off pointing being less than 6° (1%), and possible reflections off the vehicle.

## CONCLUSIONS

HES solar cells are a good cost effective choice for deep space missions up to 2.7 AU. The Stardust solar array is operating as expected at 2.7 AU.

## REFERENCES


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## ACKNOWLEDGEMENTS


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**Table 1**. LILT Radiation Test Data on 10 Ohm-cm Sharp High Efficiency Silicon Solar Cells

| Fluence (1 MeV e/cm$^2$) | Relative Intensity | Temp (°C) | Isc (mA) | Voc (mV) | Imp (mA) | Vmp (mV) | Fill Factor | Efficiency (%) |
|---|---|---|---|---|---|---|---|---|
| 0 | 1.00 | 28.0 | 187.0 | 624.1 | 172.0 | 517.7 | 0.763 | 16.3 |
| 7.20E+13 | 1.000 | 28.0 | 184.2 | 569.1 | 171.5 | 467.3 | 0.765 | 14.7 |
| | 0.289 | -28.0 | 50.0 | 650.1 | 47.5 | 566.2 | 0.828 | 17.0 |
| | 0.289 | -39.0 | 49.4 | 678.6 | 47.2 | 595.6 | 0.837 | 17.8 |
| | 0.139 | -60.0 | 23.3 | 711.0 | 22.2 | 634.1 | 0.851 | 18.5 |
| | 0.139 | -80.0 | 23.0 | 760.7 | 22.0 | 685.6 | 0.864 | 19.9 |
| 3.50E+14 | 1.000 | 28.0 | 176.2 | 523.5 | 164.4 | 430.2 | 0.766 | 12.9 |
| | 0.289 | -28.0 | 46.6 | 617.5 | 44.2 | 540.9 | 0.831 | 15.1 |
| | 0.289 | -39.0 | 46.0 | 646.7 | 43.8 | 572.7 | 0.844 | 15.9 |
| | 0.208 | -27.0 | 34.1 | 611.2 | 32.4 | 534.4 | 0.831 | 15.2 |
| | 0.208 | -39.0 | 33.6 | 640.0 | 32.0 | 565.0 | 0.842 | 15.9 |
| | 0.139 | -60.0 | 21.4 | 684.0 | 20.3 | 613.5 | 0.854 | 16.4 |
| | 0.139 | -80.0 | 21.0 | 735.0 | 20.1 | 664.2 | 0.865 | 17.6 |

**Table 2**. Comparison of telemetry to predicts at 2$^{nd}$ aphelion

| Property | Baseline Predict | Modifed Predict | Telemetry | Error |
|---|---|---|---|---|
| **Radiation (1 MeV e/cm$^2$)** | 2.40E+14 | 7.50E+13 | 7.50E+13 | |
| **Temperature (°C)** | -61 | -69 | -69 | |
| **Sun Off Angle** | 15° | 6° | 6° | |
| **Isc Sensor (mA)** | 123.0 | 131.3 | 136.1 | 3.6% |
| **Voc Sensor (mV)** | 2092 | 2201 | 2221 | 0.9% |
| **Array Output (Amps)** | 4.95 | 5.435 | 5.45 | 0.3% |

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