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POWOW – AN ALTERNATIVE POWER SOURCE FOR MARS EXPLORATION*

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ABSTRACT

Electric propulsion has emerged as a cost-effective solution to a wide range of satellite applications. Robust development of the near-Earth infrastructure that will enable Mars exploration can also rely upon solar-electric powered vehicles. Some of these options will be discussed as well. The POWOW concept is a solar-electric propelled spacecraft capable of significant cargo and short trip times for traveling to Mars. When used for Mars missions, it would enter areosynchronous orbit (Mars GEO equivalent) and beam power to surface installations via lasers. The concept has been developed with industrial partner expertise in high efficiency solar cells, advanced concentrator modules, innovative arrays, and high power electric propulsion systems.

The previous spacecraft design providing 898 kW using technologies expected to be available in 2003 produced areal power densities approaching 350 W/m^2 at 80 ^{o}C operating temperatures and wing level specific powers of over 350 W/kg were projected. Because of the high power in this satellite, high voltage operation (up to 1000 V) new work on the effects of hypervelocity impact on test modules operated at these voltages is included here. Electric propulsion options have been refined and focus only on high power Hall thrusters of new as well as conventional designs.

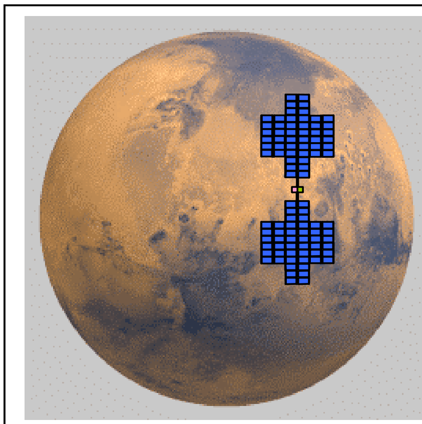

Figure 1: POWOW Spacecraft in Mars Orbit (not to scale)

INTRODUCTION

Space transportation is the key that unlocks the universe for us. It has been convenient in the past to focus attention primarily on the close-to-Earth environment (e.g. LEO, GEO and recently MEO). The Apollo missions to the Moon were initial steps beyond those destinations. However, with the advent of the International Space Station (ISS), a wider range of destinations has emerged. With the ISS as a staging point, a rich assortment of missions has emerged, some for the second and third times. These mission opportunities form the basis for a robust transportation infrastructure.

Specifically, in addition to the LEO, GEO and MEO destinations, the Earth-Moon and Earth-Sun libration points (L1 and L2 respectively) offer new destinations. For example, a transportation fuel station/depot at L1 can provide for lunar excursions for observatories, and for human/robotic exploration at the south pole. Libration point L2 offers the opportunity for observatories that can be serviced as well. Either of these locations can be used for energy-effective launching points for deep space exploration or, relevant to this opportunity, human or robotic exploration of Mars. In the past, nuclear-powered options for missions to Mars have been examined, with little attention being given to the solar-electric option. Solar-electric options were not carefully addressed, as it was generally believed that the decrease in solar irradiance at Mars would require an excessively large, costly spacecraft. It is the purpose of this paper to explore a spacecraft with a solar-electric propulsion system capable of making timely journeys to Mars. In the concept, shown in figure 1 (not to scale), the spacecraft would be placed in an areosynchronous orbit about Mars (17,000 km), from which it could beam power to the surface. During this journey, the solar intensity would decrease by about a factor of two to three (0.52 to 0.36) depending on the ellipticity of the Martian orbit and the time of launch. Because the areosynchronous orbit remains stationary above one point on the Martian surface, beam steering could be used to transmit power to several different locations, possibly reducing the surface infrastructure. With this in mind, utilization of the L1 point for power transmission to the lunar surface becomes feasible.

The ability to beam power to several locations simultaneously is a major advantage for exploration applications. For example, an ISRU plant could be powered in one location, a camp in another and conceivably power could be transmitted even to a roving vehicle. In light of this scenario, the POWOW spacecraft outlined here has a broad applicability to the transportation infrastructure supporting a range of missions along the "Highways in Space".

SPACECRAFT DESIGN

The notional design of the spacecraft encompasses the following technologies expected to be available in the next 5 years: solar cells and concentrating solar cell modules, advanced concentrator arrays, electric propulsion and beamed power transmission. Each of these will be discussed in detail.

Solar Cell Technology

Over the past 5 years, advances in III-V triple-junction solar cell technologies for space have seen the industry reach efficiencies in the 25-26% AM0 performance range. For this study, the baseline was chosen to be a 26% efficient cell operating at 80 °C with 78% of the power remaining (p/p_o) after a dose of 1 x 10^{15} electrons/cm^2. In general, the cell designs are based on the use of germanium substrates, and layers of GaInAsN, GaAs, and GaInP or similar materials. Key challenge is to develop layers that grow epitaxially on Ge and that have appropriate band gaps to capture the maximum amount of the solar spectrum. Additionally, layer thickness must be carefully controlled to maximize both efficiency and radiation resistance. These activities are underway at all space solar cell manufacturers. The 2003 projection indicates a 32% efficient, four-junction cell operating at 80 °C under concentration with a p/p_o of 83%. This, of course, is a significant stretch in technology but several aggressive programs are underway aiming at such a target. These cells would be operating at a solar concentration of 8.5x in the selected module design.

Solar Array Module Technology

The Deep Space 1 mission demonstrated the feasibility of long duration electric propulsion and validated the use of this innovative concentrator solar array technology. That array continues to operate successfully and, as noted above, has opened the trade space of missions (commercial as well as public) to new, cost-effective options. The baseline for the POWOW spacecraft is the Stretched Lens Array (SLA) pioneered by ENTECH, Inc. Significant improvements have been made in this module over the technologies used in the Deep Space 1 array.

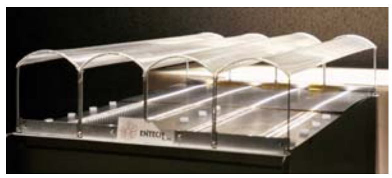

Figure 2: Stretched Lens Array Demonstration Module

Figure 2 shows a photograph of a demonstration module of the second-generation SLA fabricated by ENTECH for NASA. It consists of linear Fresnel lenses made from silicone rubber, a linear array of solar cells mounted on a thin composite radiator. The silicone lenses are flexible and are deployed by spring action. Present lens thickness is about 180 µm. The measured lens efficiency is 92% and the concentration ratio is 8.5x. The concentration level can be increased to 15x that will reduce the sun pointing tolerance. Sun pointing tolerance at 8.5x concentration is 2° normal to the length of the SLA, and at 15x would reduce that tolerance to 1°. Sun pointing tolerance along the length is better than 20°. The graphite composite radiator is about 190 µm thick for the 8.5cm lens aperture. This value would be substantially reduced were the lens aperture to be reduced in half. Radiator thickness is adjusted so as to maintain cell temperature at 80 °C under all levels of concentration. Design of the radiator is a key aspect of the success of the SLA design. Like the silicone lens, the radiator sheet is now supported as a stretched membrane between edge elements,

Unlike reflective concentrators, these refractive Fresnel lens concentrators can be configured to minimize the effects of shape errors, enabling straightforward manufacture, assembly, and operation on orbit. By using a unique arch shape, these Fresnel lenses provide more than 100X larger slope error tolerance than either reflective concentrators or conventional flat Fresnel lens concentrators.[1] 3M forms the continuous web of thin lensfilm material from space-qualified DC93-500 silicone.

The prototype SLA module shown in figure 2, Inc. achieved significant milestones. The prototype module used space quality solar cells from two vendors with efficiencies as high as 28%. The design parameters of the module were as above. The module was tested both at the NASA Glenn Research Center and at Able Engineering, Inc. with similar results. Figure 3 shows the test results under a Large Area Pulsed Solar Simulator (LAPSS) at the NASA Glenn Research Center. The lens/receiver efficiency measured 27.4% at room temperature. Furthermore, most significantly, this module achieved a specific power of 378 W/kg and an areal power of 375 W/m^2 at room temperature. These performance values have not been simultaneously achieved in any other module and met a goal established by NASA nearly two decades ago. In addition, the silicone stretched lens material was exposed to 3 space UV suns equivalent irradiance in hard vacuum at the Marshall Space Flight Center. There was only very slight degradation in the material after nearly 7000 ESH exposure.

To verify the thermal durability of the new stretched membrane lens, multiple samples were exposed to GEO thermal cycling by Able Engineering, and all passed the equivalent of more than 20 years in GEO (over 1,830 thermal cycles from –180C to +90C).

Auburn University conducted micrometeoroid impact tests of both lens and cell samples in their Hypervelocity Impact Facility. Micrometeoroid particles between 60 – 120 μm in diameter were shot at various samples. The lens impact tests showed clean penetrations with no peripheral damage such as tearing. Secondly, ENTECH mounted the single-cell photovoltaic receivers to composite radiator sheet, and fully encapsulated the receivers to enable high-voltage operation. During the impact tests, these samples, shown in figure 4, were in simulated LEO plasma, with the cells biased up to 1000 volts relative to the plasma. Peak velocities were in excess of 10 km/sec and caused only minor damage to the cover glass over each cell. One arc was observed, but it came from an impact (shown by the circle) on a

Figure 4: ENTECH module post-shot photograph

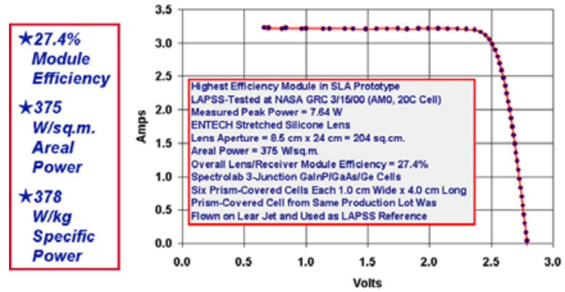

Figure 3: Performance of SLA at AM0

poorly insulated foil lead coming from the test cell assembly. Various impact sites can be seen on the module as well. This observation is not unique to the SLA but applies to all high voltage array designs and indicates the importance of care in all aspects of module and array design. This high-voltage capability, with very little mass penalty, is one of the key advantages of the SLA approach over conventional planar arrays. The small cells can be insulated against arcing without adding much mass to the array, due to the small size of the solar cells. For high-power arrays (e.g., 20 kW and larger), this high-voltage capability provide significant savings in wiring mass and cost compared to conventional lower-voltage planar arrays. The added encapsulation can also be designed to provide excellent radiation tolerance for high-radiation missions.

Additional improvements can be made in this module that should increase the performance parameters noted above by at least twofold. Thus it is possible that a long-term goal of achieving at least 500 W/kg may not be unreasonable – an outstanding accomplishment for a module that is not based on thin film cells.

Array Design

Under the NASA Cross-Enterprise Technology Development program, the SLA team is developing a new ultralight rigid panel wing as the platform of choice for SLA[1]. Figures 5 through 7 show the basic rigid panel SLA wing approach in schematic form. The flexible lenses fold down flat against the rigid panels for compact stowing during launch (Fig. 5). On orbit, as the panels begin to unfold, spring driven end arches deploy and tension the individual stretched lenses

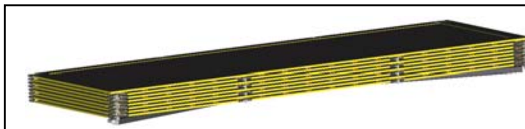

Figure 5: Stowed Rigid Panel SLA Wing

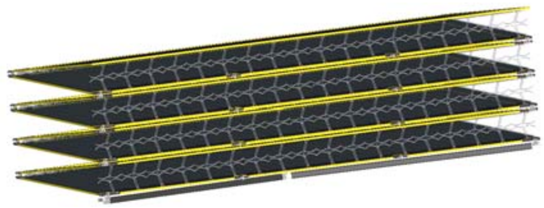

Figure 6: Rigid Panel SLA Lens Deployment

across the panel's length (Fig. 6). The wing continues to deploy (Fig. 7) until the panels are all co-planar in their final locked wing position.

Figure 8: Wing Deployment

This unfolding rigid panel solar array approach has been widely used for many years for NASA, DOD, and commercial spacecraft. One unique feature in the new SLA rigid panel array relates to the panels themselves, which use only a single face sheet and no sheet forms the photovoltaic receiver mounting surface and the waste heat radiator, and is stretched in drum-like fashion over the peripheral honeycomb picture-frame structure. The individual pop-up lenses use the same basic deployment and support approach that has been used successfully

Figure 9: Stowage of Lensfilm in Module

on numerous SLA prototypes (Fig. 9). The low mass of the supported cells and lenses enables the use of very lightweight "picture-frame" panels, and by snubbing during launch to the inboard and outboard panels which are reinforced honeycomb panels.

A detailed mass and performance analysis has been done for the rigid panel SLA wing point design. This wing has a total mass of 37 kg and provides a total lens aperture area of 24 m^2. At beginning of life (BOL), this wing provides over 7 kW of output power using cell efficiencies of 21.7% at the 80 °C operating temperature. Specific power exceeds 190 W/kg. These values are based on the use of a 190 μm thick composite sheet radiator; with a thinner 130 μm thick composite sheet radiator, the BOL specific power exceeds 200 W/kg using today's solar cell technologies. Key features of the rigid-panel SLA wing using today's technologies are summarized in Table 1. In addition to excellent performance, mass, and stiffness characteristics, the rigid panel SLA wing approach also enables the outermost panel to be populated with planar cells for pre-deployment power generation (e.g., during LEO to GEO orbit transfer), if a specific mission needs this capability.

Feature	Value or Characteristic
Point Design Basis	7,135 Watts (BOL)
SLA Implementation	Pop-up lenses
Base Platform Design Maturity	Most components flight proven on DS1
Specific Power (130 micron facesheets)	203 W/kg
Stowed Volume	0.093 m^3/kW
Stowed Stiffness	40 Hz
Deployed Stiffness	0.12 Hz
Stowed Power	Easily implemented on outer panel
Ease of Adding Planar Panel	Easily implemented on outer panel
Flatness & Warping	Well understood flat stable platform
Deployment Testing	Can use existing off-loaders
Power Testing	Pop-up lenses allow each panel to be tested as a complete assembly before wing integration
Commercial Appeal	Easier to integrate on commercial spacecraft. Readily accepted configuration.
Self Shadowing	No self shadowing

Table 1: Key Features of the Rigid Panel SLA Wing

Wing Design

Initial design of the wing uses the rigid panel building block. The present configuration uses 26% efficient multijunction cells and a 78% radiation degradation factor in a module whose dimensions are nominally 8 m by 4 m. Six of these units are combined with their deployment mechanisms resulting in a 48 kW "building

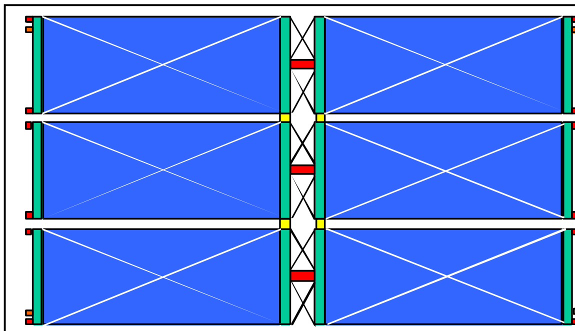

Figure 10: 48 kW Building Block

block" shown in figure 10. The 48 kW elements package neatly into a 2m x 2m x 4m envelope. Four such units could be packaged in the Space Shuttle cargo bay.

ELECTRIC PROPULSION TECHNOLOGIES

In the past year, additional progress has been made on the electric propulsion subsystem. General Dynamics – Space Propulsion Systems (formerly Primex Aerospace) is developing a new design for a 50 kW Hall thruster. GD–SPS included plans to design both a conventional circular Hall thruster with enhanced control of the ionization and acceleration zones, as well as a novel linear Hall thruster. Conceptually, the linear design effectively cuts the circular thruster at a point and unfolds it. The linear Hall thruster, if successful, could provide real packaging advantages for multi-hundred kW systems like POWOW. This provides a near-term pathway to electric propulsion subsystems that can form the building blocks of 500 kW (10 thrusters) to 1 MW (20 thrusters) total power systems.

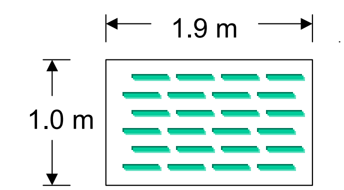

Figure 11: 50 kW Linear Hall Thruster Design

Figure 11 shows the design using the proposed 50 kW Linear Xe Hall thrusters. Isp is expected to be above 3000 s, efficiency above 65% and thrust at least 1 N. This new design can be contrasted to that using conventional 25 kW Hall thrusters. In the previous design, 48 Hall thrusters were used to provide the necessary thrust. They covered an area of 8.75 m^2. The unit shown in figure 11 has reduced the number of thrusters to 24, with 18 being active and 6 spares and covers an area of 1.9 m^2. This represents an 80% reduction in area and we expect to achieve the same reduction in mass and volume. Hall thrusters will operate at a voltage of 300 - 800 V hence are consistent with the high voltage module tests described above.

2003 TECHNOLOGY

Taking all of the design considerations noted above into account, figure 12 shows the configuration of the rigid array POWOW system using technology expected to be available in 2003. Each wing is built from eight 3x2 modules using cell efficiencies as high as 32% BOL. With increased tolerance to radiation, the combination leads to a 7 kW EOL panel. Thus each wing produces 336 kW for a total of 672 kW EOL. BOL performance is 820 kW. Array dimensions remain nominally 53 m wide by 50 m in length. Mass of each wing has not been estimated yet, however it is expected that mass savings up to 500 kg may result due to the rigid design compared to the previous design[2]. Thus the mass of each wing may drop to 1988 kg, but detailed calculations have not been performed.

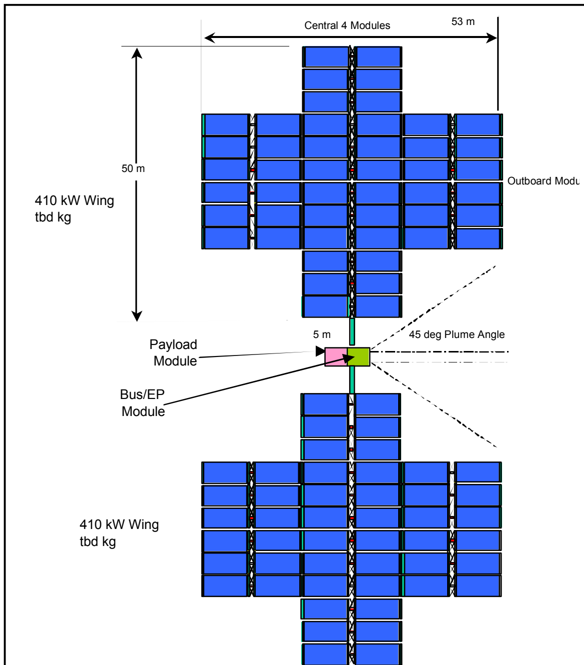

Figure 12: 820 kW POWOW Configuration

POWER BEAMING

In the previous work, two power beaming options were considered: microwave and laser. Because of on-orbit aperture limitations, the microwave option was dismissed due to the 17,000 km areosynchronous orbit. Simple calculations of the apertures required on the surface and in space were much too large to be reasonable. Furthermore, very low excitation of the receiving antenna was produced such that simple use of solar energy would produce more surface power for that area. Of the laser options, 10.6 and 1.06 µm wavelengths were considered. The surface conversion system for a 10.6 µm laser was either a thermal engine or a thermophotovoltaic converter. Sizes of concentrators on the surface and their aperture areas were determined. Using a 1 m laser aperture on orbit, the 10.6 µm case required a surface receiving aperture area of 360 m^2. Although large, it is well within current technology limits. Chain efficiencies were determined and surface power delivered was determined. While the 10.6 µm option was feasible, it was dismissed because of concern over exciting modes in the 7mb Martian CO_2 atmosphere.

For the 1.06μm laser option, the surface receiver aperture became only 36 m^2, or a diameter of only 6.8m. An additional benefit of using a wavelength in the near IR is that a single junction, direct band gap solar cell would be an excellent, high efficiency converter. With appropriate choice of materials and cell band gap, efficiencies of at least 50% are reasonable.

Various research efforts are being conducted into laser options in this wavelength range. Lawrence Livermore National Laboratory reported a 192 bar laser diode array that produced 23 kW of power at 0.900 μm wavelength and 43% efficiency.[3] However, cooling requirements for this array are substantial because of the close alignment of the diode bars. At the nominal Mars-Sun distance, the POWOW spacecraft array would produce about 360 kW that could be available for beaming to the surface at Mars. With a 43% laser efficiency, a beam of 150 kW could be sent to the surface using only 7 of the laser diode arrays. With a nominal solar cell conversion efficiency of 50% on the surface, 75 kW would be available for surface operations. Other laser options may also become available. One interesting option may be to develop a dispersed array of diode lasers operating at lower voltages but dispersed across the arrays. This would significantly ease cooling requirements and potentially obviate the need for high voltage on the arrays. Aperture issues and beam coherence issues would have to be resolved to establish feasibility of this approach. Further study is required for these options.

OTHER MISSION OPTIONS

As the development of the Earth-Moon infrastructure continues, significant opportunities for the POWOW class of spacecraft emerge. For example, for the 820 kW POWOW noted here, if the power system specific mass is 6 kg/kW and a propulsion specific mass reaches 5 kg/kW then escape trip times are only 72 days. These trip times are quite short compared to lower power transfer vehicles. The payload for this case is 5.4 MT. Other possible destinations within this sphere include L1 and L2 as well as well as operations in the weak stability boundary. Asteroid sample return missions as well as other deep space destinations are included.

Because of the semi-rigid approach used in the current design, a broad range of spacecraft applications near earth also emerge. Missions to GEO can benefit from the option of populating the outermost panel of the semi-rigid array with planar cells to provide power during the transfer orbit. Furthermore, many mission planners and spacecraft program managers prefer the low-risk, rigid panel deployment approach over the flexible blanket approach outlined previously.[2]

SUMMARY

A revised design of a 820 kW solar electric propulsion spacecraft for transit to Mars was presented. Design features were based on solar cell, module and array technologies expected to be available in the year 2003. Present values of electric propulsion thruster performance were used and estimates were made based on a novel new concept. This concept is the linear Hall thruster design that simply takes the circular Hall thruster and unwraps it into a linear format. Laser beamed power options used current technology. These elements combined to yield the following observations. Using lasers to beam power to the surface of Mars was the preferred option. Calculations for a 1 m laser aperture on orbit and a laser operating a 1.06 μm leads to a receiving aperture of only 6.8 m diameter. A 900 nm wavelength laser operating at 43% efficiency was chosen for design purposes. With a 820 kW spacecraft at an average Mars-Sun distance, a total power of 75 kW could be delivered to the Martian surface.

ACKNOWLEDGEMENTS

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