POwer WithOut Wires (POWOW)

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1. ABSTRACT

Electric propulsion has emerged as a cost-effective solution to a wide range of satellite applications. Deep Space 1 successfully demonstrated electric propulsion as the primary propulsion source for a satellite. The POWOW concept is a solar-electric propelled spacecraft capable of significant cargo and short trip times for traveling to Mars. There 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 present baseline spacecraft design providing 898 kW using technologies expected to be available in 2003 will be described. Areal power densities approaching 350 W/m^2 at 80 $^{\circ}$C operating temperatures and wing level specific powers of over 350 W/kg are projected. Details of trip times and payloads to Mars are presented. Electric propulsion options include Hall, MPD and ion thrusters of various power levels and trade studies have been conducted to define the most advantageous options. Because the design is modular, learning curve methodology has been applied to determine expected cost reductions and is included.

2. INTRODUCTION

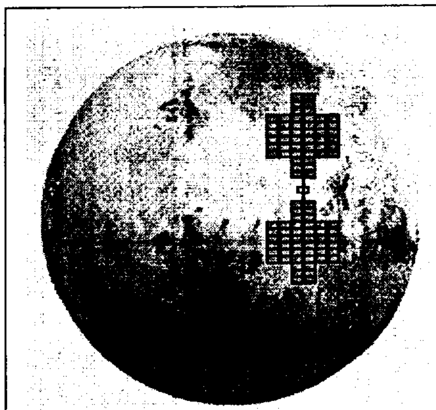

Figure 1: 898 kW POWOW Spacecraft in Mars Orbit

With recent announcements that evidence of possible life forms had been discovered in Antarctic meteorites believed to have come from Mars, the interest in sending probes and ultimately people to Mars is taking new prominence. In the past, nuclear-powered options have been examined, with little attention being given to the solar option. Solar 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, from which it could beam power to the surface. Because this 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. A team consisting of TECSTAR, Entech, Inc., Able Engineering and the Primex Aerospace Co. was assembled with the Space Power Institute to conduct an in-depth study of the concept. The team members and their responsibilities are:

- Space Power Institute, Auburn University
  - System design, integration, laser beamed power
- Able Engineering Co., Inc.
  - Advanced lightweight concentrator array, learning curve
- Entech, Inc.
  - Ultralightweight SLA concentrator modules
- Primex Aerospace Corporation
  - Electric propulsion system design definition
- Tecstar - Applied Solar Division
  - Concentrator multijunction solar cells

3. MISSION CONCEPT

As a brief summary, the solar-electric mission would leave from a high earth orbit, near escape velocity. The vehicle uses a modular solar array composed of an 8 kW array. This array is designed to be applicable to a wide range of missions in addition to the Mars trip. Six of these 8 kW array elements are combined into a 48 kW building block. Finally, 16 of these subarrays are combined into a two-wing 898 kW power supply for the satellite. Electric propulsion subsystem consists of 48 25 kW Hall thrusters (36 active, 12 spares). The trip times given in a later section include the transfer to Mars and insertion into orbit with a 4 MT payload. 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. This decrease in solar intensity is also accounted for in the trip time calculations. The trip times depend strongly upon spacecraft power and mass as expected.

Once at Mars, the spacecraft would be placed in an areosynchronous orbit at an altitude of 17,000 km. At this altitude above the equator, the period of rotation of the spacecraft matches that of the planet and it will remain above that point. From this location it would be possible to beam power to areas at least ±10° above and below the equator with beam steering. Laser beamed power is shown to be the system of choice for this application. 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.

4. 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. Table 1 shows the relevant projections of solar cell performance over the next five years.

Table 1: Solar Cell Performance Projections

Characteristic	Present	2001 Technology	2003 Technology
Type	GaInP/GaAs/Active Ge	GaInP/GaAs/GaInAsN /inactive Ge	GaInP/GaAs/GaInAsN /active Ge
P/P_0 at $1xe^{15}$ e/cm^2	78%	78%	83%
Efficiency (80 °C, 8.5x – lot min avg)	23.6%	28.5%	32.2%

For this study, the baseline was chosen to be a 26% efficient cell operating at 80 °C and 8.5x concentration with 78% of the power remaining (p/p_o) after a dose of 1×10^{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. This represents near term technology expected to be available in the 2001 time frame. 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

With success of the Deep Space 1 mission, the use of innovative concentrator solar array technology was space validated. This array continues to operate successfully and has opened the trade space of space 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.

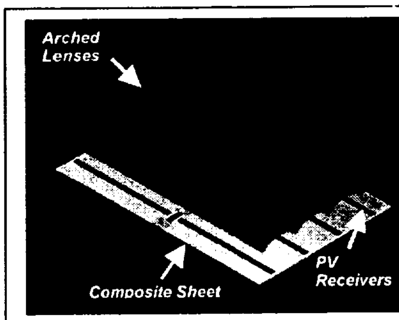

Figure 2: Schematic of Stretched Lens Array

Figure 2 shows a sketch of the present SLA module. It consists of linear Fresnel lenses made from DC-93500silicone rubber, a linear array of solar cells mounted on a thin graphite composite radiator. The silicone lenses are flexible and are deployed by spring action. Their present thickness is about 180 µm but can be substantially reduced in the future. Confirmed lens efficiency is 92% and the concentration ratio is 8.5x. It will be possible to increase the concentration to 15x in the future. Sun pointing tolerance 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 150 μm thick for the 8.5 cm 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.

Recently, a prototype SLA, fabricated by Entech, Inc. under another NASA Space Solar Power Exploratory Research and Technology Program 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 as shown in Figure 4. It is interesting to note that the coated sample exhibited slight degradation while the uncoated sample did not.

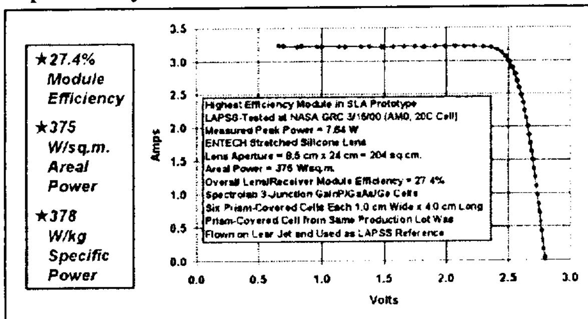

Figure 3: Performance of SLA at AM0

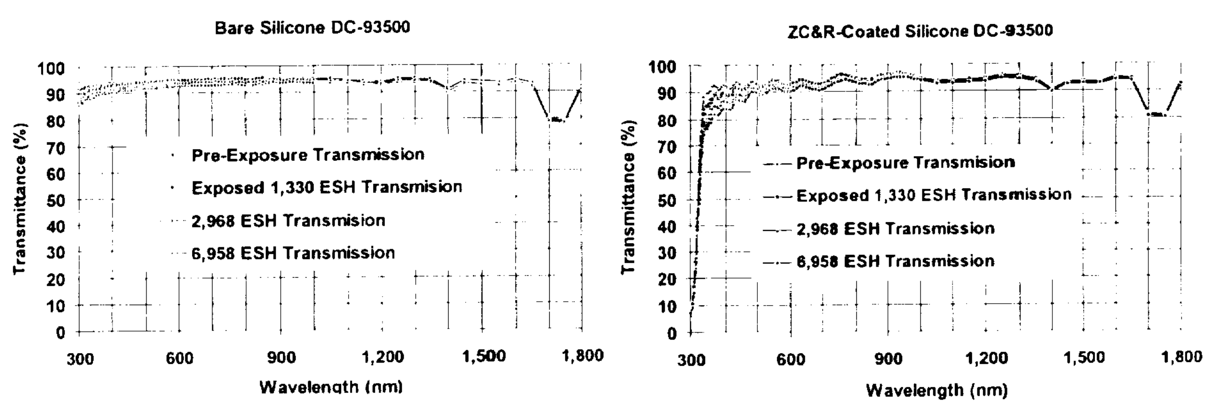

Figure 4: UV Test Results of Lens Material

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 1000 W/kg may not be unreasonable – an outstanding accomplishment for a module that is not based on thin film cells.

One of the other factors that is important in solar arrays that will be used with electric propulsion is their ability to reach high voltages. Having an array that operates at a voltage that fits the propulsion unit saves mass by eliminating a power processor to convert the spacecraft voltage to a higher operating voltage. For the most part, spacecraft bus voltages are 100 V or lower whereas electric propulsion units require maximum voltages between 300 and 1200 V or more. A first order design of the cell/coverglass assembly that should permit high voltage operation is shown in figure 5. The principle behind this design is to use an appropriate amount of cover glass overhang that is determined analytically for the desired environment and operating voltage and then to ensure that additional cover glass adhesive completely encapsulates all four edges of the cells. This should permit operation at any relevant voltage of interest to electric propulsion. Another benefit of operating at high voltages is to reduce the amount of current being processed and thereby reducing the mass of copper wiring substantially.

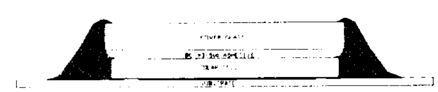

Figure 5: Schematic of Solar Cell/Cover Glass Assembly for High Voltage Operation

Array Design

Able Engineering, Inc.[1] has taken the results from the Entech effort and created a modular array design based on the SLA technology. Some of the ground rules for that design are to use a building block of a nominal EOL power of 8 kW. This power level was selected to be suitable for the emerging GEO communications satellites with power levels from 15 to 30 kW. Similarly, the same array segment can meet LEO mission needs in the 3-4 kW range with no significant changes.

Initial design of the wing uses the nominal 8 kW building block shown in figure 6. The baseline configuration uses 26% multijunction cells and a 78% radiation degradation factor in a module whose dimensions are 8 m by 4 m. The detailed design of the array, its' lens film/lens support structure and the entire packaging for launch have also been designed. The cells are placed on the thin film radiator substrate made as a carbon fiber composite structure. The lens film is supported by a lens positioner that automatically raises the lenses into operational position as the cell panes and lenses are tensioned. The lens positioner has the correct curvature to maintain lens shape necessary for efficient operation. The individual panels are attached to one another and unfold in an accordion fashion.

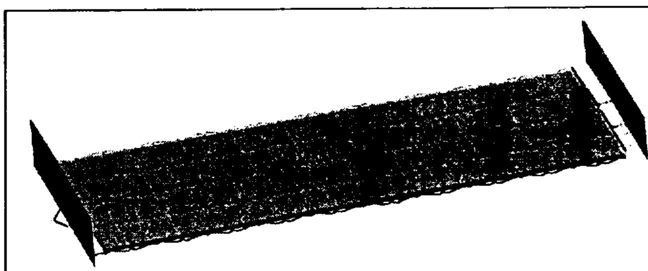

Figure 6: 8 kW Solar Array Wing Design

A range of deployment mechanisms is available for this application. These extendable beams may be scissors mechanisms, coilable masts or inflatable beams etc. Several of these options are shown in figure 7. After preliminary study, the inflatable design is too massive when the weight of the deployment gases and their canisters is considered. Similarly, the coilable mast structure like that used on the International Space Station is more massive than the simple, well tested scissor deployment approach. Thus that system was used as the baseline design.

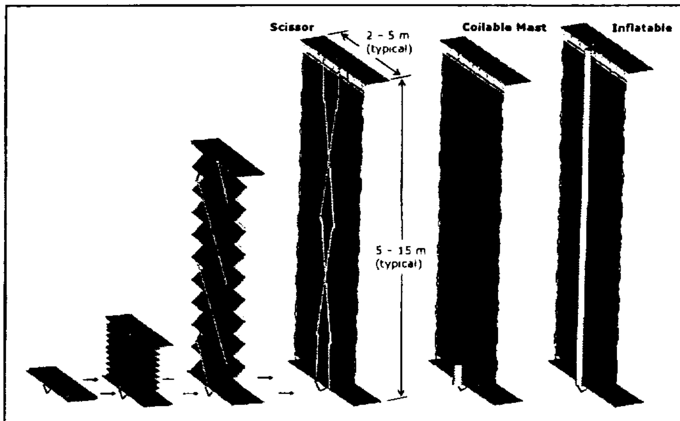

Figure 7: Deployment Options for 8 kW Array

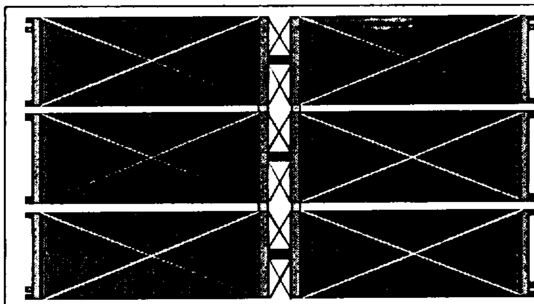

Figure 8: Notional 48 kW Building Block

Six of the 8 kW units are combined with their deployment mechanisms resulting in a 48 kW "building block" shown in figure 8. The 48 kW elements package neatly into a 2m x 2m x 4m envelope. In this packaged configuration, four of these elements can be packaged inside the 5.6m diameter fairing of the Proton Plus launch vehicle or the Space Shuttle shown in Figure 9. With the advanced cell and SLA technologies, the total power in such a launch will approach 250 kW.

In addition to these notional designs, the mass and dynamic response of the nominal 8 kW solar array building block has been determined using 2003 technology levels. The ANSYS code was used for the finite element modeling. Specific assumptions included an infinitely stiff root interface and the blanket tension was applied with initial strain. Figure 10 shows the dynamic response obtained for the major blanket vibration modes. When these single wing data are applied to the full 898 kW spacecraft (449 kW wings) using the classical Dunkerly superposition mode analysis, the approximate bending mode was

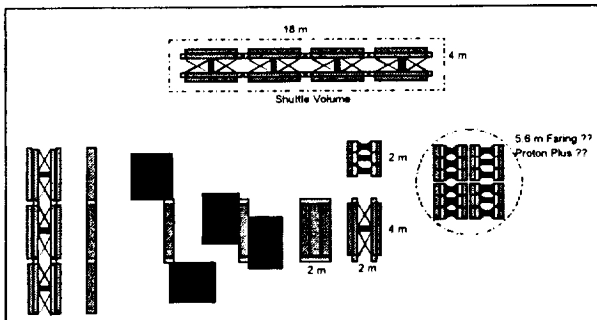

Figure 9: Schematic of Array Stowage and Deployment

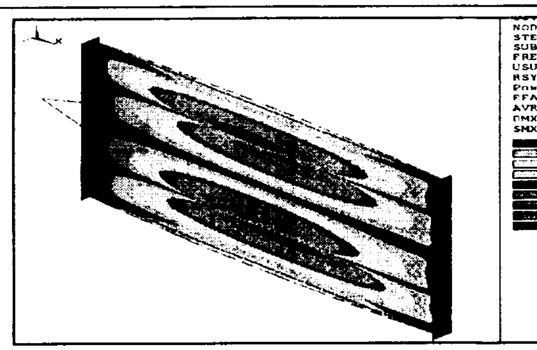

Figure 10: Local Blanket Mode Vibrations

estimated to be 0.08 Hz. These data suggest that some method of virtual stiffness enhancement such as active frequency control of the structural network may be necessary.

5. ELECTRIC PROPULSION TECHNOLOGIES

With power levels up to 898 kW, the design of the propulsion module/array is driven by plume considerations to a first degree and by the secondary issue of array pointing versus thrust vector. The configuration of the system also drives the thruster choices to higher power levels. In this study, four suitable electric propulsion options were identified from a wide range of candidates determined by the Primex Aerospace Corp. These options fell into two general categories – 25 kW class thrusters and very large thrusters with power levels above 200 kW per thruster. The choices were based primarily on relatively near term technologies. A range of other electric propulsion options were not included because they were at much earlier state of development and appropriate performance values could not be reliably obtained. The selected choices are shown below and their characteristics are shown in Table 2:

- 25 kW Xe Ion
- 25 kW Xe Hall
- 256 kW Li Applied Field MPD
- 768 kW Li Self Field MPD

Table 2: Electric Propulsion Thruster Options

Thruster	Isp (sec)	Eff. (%)	Thrust N	Number
Xe ion	6000	65.0	0.514	60
Xe Hall	3242	62.1	0.976	30
AFMPD	5382	54.0	5.12	3
SFMPD	3405	43.6	19.45	1

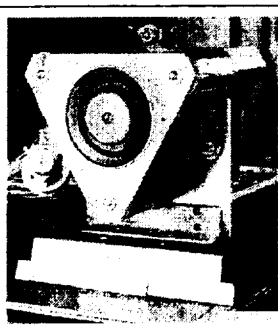

Figure 11: Hall Effect Thruster with Anode Layer, TAL-55D

A picture of the TAL 55D Hall effect thruster is shown in figure 11. Its larger companion, the Russian TM-50 is the thruster used for the values shown in Table 1. The benefit of a Hall thruster is that it can operate at a voltage of 300 - 800 V at the 25 kW level noted above. Trip times and specific mass of the thruster system will depend on the thruster performance values noted in above. The number of thrusters required for the mission (including spares) and specific mass of the thruster system was made using the data for the TM-50 thruster. The specific mass of the thruster system includes the thrusters (including spares), the power processing unit and its radiator, cabling and gimbals. The specific mass is taken at 10 kg/kW in determining trip times. Future mass reductions, especially in Hall thruster design are not included in this study but have been projected to be around 50% in both mass and volume.

6. 898 kW SYSTEM – 2003 TECHNOLOGY

Taking all of the design considerations noted above into account, figure 12 shows the configuration of the 898 kW POWOW system that uses technology that should be available in 2003. Each wing is built from eight 3x2 "Aurora" modules identical in design to those shown in figure 4, but using the advanced cell technologies described above that boost the total power to just over 56 kW EOL. Thus each wing produces 449 kW and has dimensions of 53 m wide by 50 m in length. Each wing has a mass of approximately 2488 kg. The electric propulsion module is centrally located and has a 45-degree plume clearance from the closest array element. There are 48 Xe Hall thrusters arranged in a 6x8 configuration. Thirty-two of the Hall thrusters are active with the remaining sixteen in reserve. The propulsion module weighs about 10,700 kg exclusive of propellant.

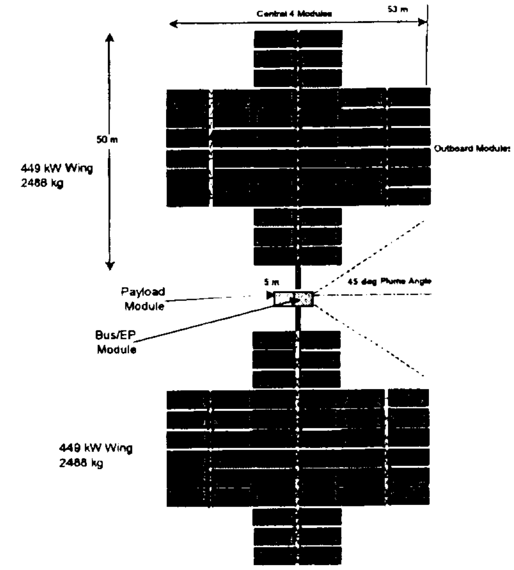

Figure 12: 898 kW POWOW Configuration

7. TRIP TIME CALCULATIONS

The NASA Glenn Research Center has developed a simple program for performing quick estimates of trip times for electrically propelled spacecraft traveling to Mars called All SEP Mars1.0[2]. Input parameters are the power and specific mass of the power system, the specific mass, efficiency, and Isp of the propulsion system and an estimate of the initial mass in heliocentric orbit. Output data include the mass delivered to Mars, the amount of propellant and tankage, the payload and trip time summaries for escape, heliocentric travel and Mars capture.

For this study, the trip times were determined assuming that the spacecraft had been boosted to near escape velocity so travel through the radiation belts was unnecessary. The results of the modeling using the Xe Hall thruster yield a payload of 4.0 MT, IMHELIO of 44.4 MT and a trip time to Mars of 221 days with another 9 days required for capture. The capture orbit in this case is a lower orbit than areosynchronous so the total time is conservative. Total propellant mass is 24 MT. Clearly, numerous propulsion options are available, given the partial list of options noted in Table 1. Thus, figure 13 shows the All SEP Mars 1.0 calculations for a constant payload of 4.0 MT. This value was chosen arbitrarily to demonstrate the options available to the mission designer. Two primary options surface from this simplified analysis. Thrusters with high Isp yield large reductions IMHELIO but at the penalty of significantly increased trip times. A lower Isp reduces trip times but at the penalty of

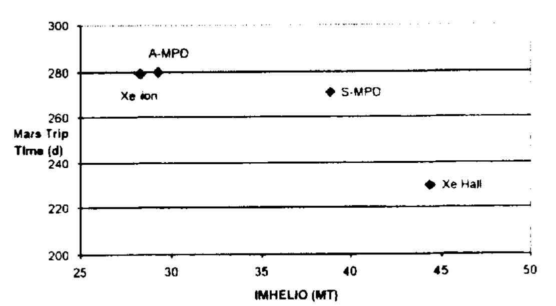

Figure13: Thruster Options for 4.0 MT Payload

a larger IMHELIO. The trip times noted here are reasonable for payload transits to Mars and are similar to chemical propulsion results.

8. POWER BEAMING

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 7 mb Martian CO_2 atmosphere.

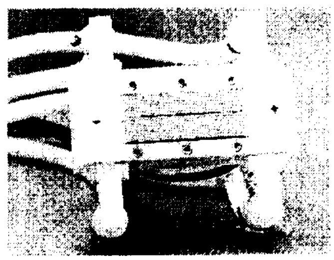

Figure 14: Laser diode array

For the 1.06 μm 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.

Extensive research is being conducted into various laser options in this wavelength range.[3] Figure 14 shows the 192 bar laser diode array that yielded 23 kW of peak power at 0.900 μm wavelength with an efficiency of 43% reported by the Lawrence Livermore National Laboratory[4]. However, cooling requirements for this array are substantial and are not covered in this paper. If about 400 kW of the POWOW spacecraft power is available for beaming to the Mars surface (using the mean Mars-Sun distance), then a beam of nearly 175 kW could be sent to the surface using only seven or eight of the laser diode arrays. With a nominal solar cell conversion efficiency of 50% on the surface, about 80+ kW would be available for surface operations. This is equivalent to the power available from the former SP-100 space nuclear reactor program without requiring any nuclear material.

9. COSTING METHODOLOGY

In order to perform a costing analysis, Able Engineering[1] made several baseline assumptions were made about the program that would lead to such a spacecraft. Thus to produce the 898 kW spacecraft, there would be two solar arrays, 16 3x2 "Aurora" power modules as shown in Figure 8, and thus a total of 96 of the Aurora units (nominal 8 kW, but actually 9.36 kW for the 2003 technology level). Furthermore, it was assumed that proper manufacturing planning steps occurred in the non-recurring phase of the program and that qualification arrays have been successfully produced.

The delivery was assumed to be 1 year from start of production and the production was phased in three distinct phases: Low Rate Initial Production (LRIP), Initial Production (IP), and Production (P). A baseline factory was outlined and concurrent manufacturing and design engineering was used. The latter included such elements as manufacturing design and value engineering, thorough production planning, customer coordination and scheduling, and procurement economies to set the stage for cost estimation. Specifics of the layout of the optimum facility included 3200 ft^2 for program administration and engineering, 9200 ft^2 for shipping, receiving, kitting and inventory, 25,450 ft^2 for assembly and test and 7000 ft^2 for solar power module and unit staging. Automation would be used where warranted and there would be 13 work cells. Eleven of these are day-to-day work cells (some duplicated) with two off-line work cells in reserve.

The process would use concurrent manufacturing and design engineering with the categories of: manufacturing design, thorough production planning, customer coordination and scheduling, procurement economies and learning curve methodology. Value engineering would include: standardization, reduced parts count, modularity, optimized fabrication processes, reduced touch labor and optimized production flow. Relevant ground support equipment would include: assembly and handling tools, test fixtures and systems and shipping and storage containers. Concurrent engineering is the first step to cost savings. Production planning is also crucial. Careful planning includes: work cell planning, proper lot size determination, procedure and production control documentation, and plans for maintenance, rework and systematic product improvement. This type of planning is key to a successful and cost effective program. To do proper work cell planning, work cells should be based on similar disciplines and work flow. The work flow capabilities include the number of cells, the number of shifts, the role and usefulness of automation and supply chain capacities. A gating hierarchy must be established and the work should fit to existing facilities and personnel insofar as possible. As appropriate, new resources would be created to conduct the work.

A single controlling document is essential for team evolvement and production control. All steps must be accounted for in a work flow diagram: procurement, inspection, kitting, assembly steps and processes, testing and shipping. All GSE must be tracked. Items can include: storage and stocking needs, handling equipment, assembly tools, test fixtures and systems, shipping containers and other specialized facility requirements. Careful procedure documentation is also essential. Likely items include: work and inspection instructions, test plans and procedures, kitting instructions, and handling and rework instructions. Timely preparation and proofing of new or revised procedures and their documentation will reap additional benefits.

Customer coordination and scheduling is an often overlooked element. Team coordination will yield the best overall solution. It is important that all parties understand and communicate the production plateaus that occur and the work should be planned within comfort zones. Supply chain management, the numbers of shifts and work cells and the use of automation help create these comfort zones. With all these considerations, an optimized product delivery schedule that is sensitive to customer needs can be established. A strategy for procurement economies is based upon including vendors as team partners, working to take advantage of price break plateaus, bulk purchases shipped "just in time" and long term purchase order agreements. Clearly, controlling procurement is the quickest avenue to cost savings.

With the realistic phasing breakdown of the program described above, it is possible to assess the greater production efficiencies for each phase using a logarithmic learning curve:

$$COST_{production} = TFU \times N_{units}^{[1-\ln(1/LC)/\ln 2]}$$

Where TFU is the theoretical first unit cost, N_{units} is the production quantity and LC is the learning curve factor. This factor is set at 95% for the LRIP phase where the first ten items are delivered. The second phase is based on a quantity of 10 to 50 units and has a learning curve of 90%. The third phase covers any quantity beyond 50 units and has an LC value of 85%. It should be noted that this approach is approximate and does not capture "learning" effects versus the step-wise addition of more efficient tooling or other one-time events. It also does not take into account any new technologies that may be implemented that will make significant downward steps in costs. With the planned production cost model, the program is divided into these three phases. It is important to consider this expanded model as most production efforts are impacted by changes either in technology or production methodologies as they progress. The model shown above is used in each phase with the appropriate LCs and TFUs. The equation for this model becomes:

$$COST_{production} = TFU_{LRIP} \times N_{LRIP}^{[1-\ln(1/LC_{LRIP})/\ln 2]} + TFU_{IP} \times N_{IP}^{[1-\ln(1/LC_{IP})/\ln 2]} + TFU_{P} \times N_{P}^{[1-\ln(1/LC_{P})/\ln 2]}$$

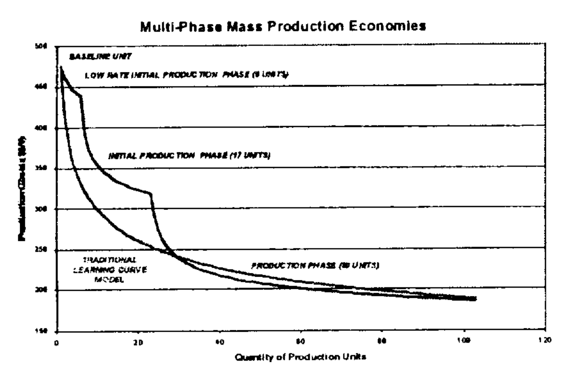

Figure 15: Production economies with phasing

The results of this modeling are shown in Figure 15 and the step-wise approach compared to a traditional learning curve model. The results clearly show that the phased approach matches the traditional learning curve methodology when the program enters the production phase. The traditional learning curve methodology tends to underestimate costs at the earlier stages of production. An additional benefit of the modular design of the POWOW array technology is that other markets exist for the 8 kW building block units. Thus descending this learning curve is not dependent upon a single large program, but is a program that can benefit multiple users simultaneously and provide early cost benefits to all. Note that costs under \$200/W are achieved at production levels of about 60 units. A robust planetary exploration program coupled with aggressive government and commercial use of near-Earth orbits can provide timely and cost effective implementation of this technology. However, critical to acceptance of concentrator arrays for space is successful demonstration in space. Hopefully Deep Space 1 will provide validation of both the stretched lens linear Fresnel lens array and electric propulsion technologies.

10. SUMMARY

A preliminary design of an 898 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. Laser beamed power options used current technology. Trip times to Mars were obtained from the program All SEP Mars 1.0 obtained from the NASA Glenn Research Center. These elements combined to yield the following observations.

Trip times to Mars and the IMHELIO mass were dependent upon the Isp of the thrusters used. The Xe ion and applied field MPD thrusters gave trip times of about 279 days with small IMHELIO masses of 29 MT. The Xe Hall thruster system provided the lowest trip times of 230 days at the expense of a 44 MT IMHELIO. The self-field MPD thruster fell in between with a trip time of 271 days but IMHELIO of 39 MT. Using lasers to beam power to the surface of Mars was the preferred option. Laser wavelengths of 10.6 μm were eliminated from consideration due to concerns over adverse effects to the Martian CO_2 atmosphere. 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. Assuming that only 700 kW of the 898 kW spacecraft power were available for power beaming, and that the efficiency of converting the laser beam to electricity was 50%, then 150 kW could be delivered to the Mars surface. Use of a modified learning curve methodology showed that with proper phasing of the program, costs would be reduced to less than \$200/W, even at a total production of only 96 units.

11. RECOMMENDATIONS

The most critical element of the entire system is the 8 kW stretched lens array building block. Even in its early stages demonstrated here, it has achieved performance levels that promise to far exceed conventional planar solar array technology. An aggressive program to further mature the designs developed under this program by Able Engineering and Entech, Inc. should be undertaken. The designs outlined herein have progressed enough to show the assembly, packaging and deployment of the stretched lens array at an 8 kW level. The assessment of the dynamic characteristics suggests some additional improvements would be warranted. Continued development and production of engineering models of this array that would ultimately lead to a space demonstration of a minimum 1 kW array of this design would be warranted.

This study also demonstrated that electric propulsion systems when coupled with lightweight array systems can provide encouraging trip times to Mars. Only two electric propulsion options were delineated here. It would be desirable to develop a significant program to bring a wide range of electric propulsion technologies to full space use. Such a program should encompass strong system studies to support technical decisions. Mass reductions of the propulsion units as well as the power processors should be a strong element of such an effort.

The demonstration in this work that beaming power via laser energy is a superior system comes as no surprise. However, in the past, power beaming was thought to be exclusively the arena of microwaves. The benefit of higher frequency electromagnetic radiation is well known, but the limitations that atmospheric distortions and beam absorption by various species has not been extensively studied by NASA. Past work in NASA and the DoD focused on beaming laser energy from Earths' surface to objects in space. It is appropriate to build upon that past work and the work performed herein to reawaken the use of laser energy for beaming to a planetary surface from space. As in the previous recommendations, an aggressive experimental demonstration program beginning with point-to-point laser power beaming on Earth would appear advisable.

Finally, with successful demonstrations of the relevant technologies needed for space to surface laser power beaming, a space demonstration of significant yet affordable size should be implemented.

12. ACKNOWLEDGEMENTS

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