Energizing the Future of Space Exploration: Applications of Space Solar Power

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[Abstract] As NASA's Space Exploration policy takes shape, one of the key issues that will affect all aspects of the exploration architecture is the ability to provide electric power to various surface locations on the moon, Mars, other planets or asteroids. This power should be available through daylight times as well as at night. The moon is an especially difficult case in this regard. While nuclear reactors and radioisotope power sources may be one choice for continuous power, it is the purpose of this paper to explore the advantages of power beaming by lasers and/or microwaves to provide power to a wide range of locations on the surface of the moon, Mars and Venus.

I. Introduction

AS NASA's Space Exploration policy takes shape, one of the key issues that will affect all aspects of the exploration architecture is the ability to provide electric power to various surface locations on the moon, Mars, other planets or asteroids. This power should be available through daylight times as well as at night. The moon is an especially difficult case in this regard. While nuclear reactors and radioisotope power sources may be one choice for continuous power, it is the purpose of this paper to explore the advantages of laser and/or microwave power beaming to provide power to any location on the surface of the moon, Mars and Venus.

Inherent in this discussion is the use of electric propulsion as the transportation mode of choice. The major benefit of solar electric propulsion (SEP) is that more payload can be delivered to the moon, Mars or Venus for less cost than by chemical means. In addition, SEP allows for high delta-V orbital adjustments to permit a range of orbital locations that best fit the application. This will be important for Mars. A corollary benefit of using electric propulsion is the availability of that high "transportation" power for beaming power from the relevant orbit about the body.

Many lunar orbits can be used for power beaming. All have system impacts depending on the location of the receiving station. The L1 (or L2) location requires a beaming distance of about 56,000 km to provide continuous power to near-equatorial lunar locations. The constraints on laser power beaming over this distance have been described previously.[1] If a Molniya-type highly elliptical orbit is chosen, the apogee need be only about 12,000 km, hence beaming issues are eased for a near-equatorial site. The length of time the lunar surface site is in view by the satellite is important for keeping the amount of energy storage on the surface small. In the same way, circular orbits of varying heights above the moon encounter the same site view time issue. Thus, maximum elevation of the beaming spacecraft, the angular distance, the precession of orbits around the moon the perturbations of lunar gravity and the number of spacecraft all combine to complicate the analysis. The situation is eased greatly if the receiving site is at a polar location. The results of these analyses will be presented.

For Mars, the situation is completely different. In this case, the SEP spacecraft would enter the Mars areosynchronous orbit (17,000 km) and beam power to the surface from there. This orbit is equivalent to Geosynchronous Earth Orbit (GEO); hence the spacecraft remains virtually stationary over the Martian equator and could beam power to a range of latitudes. The most significant issue on the Martian surface is dust storms. While this is an important issue and may affect the laser wavelength used, dust obscuration should not be a major issue on the receiving array. Martian dust is dry and has no sharp edges hence by tilting the receiving array; the dust will likely not adhere. For Mars, the use of microwave beaming will offer several additional complex issues as well.

For Venus, the issue is much clearer. Only microwave power beaming is feasible due to its dense cloud cover. Although the surface temperature is a major issue, with power beaming, a long-life rover may be feasible. However, the major restriction on this option is the exceptionally long beaming distance for a synchronous-orbiting satellite over Venus.

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For laser beaming, a wavelength range of 830 - 850 nm has recently been selected because it is well suited for peak performance of GaAs (or Si) solar cells. It is also a wavelength band that penetrates the earth's atmosphere with minimal absorption (of course the moon and Mars will be fully transparent to these wavelengths). However, the earlier studies discussed here generally used a 1.06 μm wavelength for conversion with solar cells. These single junction cells are expected to be less expensive than the triple junction cells, but have not been in production for some time for space use. As a relevant point, multiple junction cells cannot be used to receive single wavelength laser beams. Testing of GaAs solar cells in a concentrator array with a ~840 nm laser has yielded efficiencies over 45% at room temperature. For microwave power beaming, 5.8, 10 and 35 GHz wavelengths have been chosen.

II. Lunar Power Beaming

The ultimate necessity for effective human habitation on the moon is a continuous source of power for 24 hours of every earth day and ~29 earth days of the lunar day. Without continuous power, human habitation and development of a viable infrastructure and economy are simply unattainable. Past studies have examined the use of regenerative fuel cells powered from solar arrays, and the use of a buried nuclear reactor power system. Both approaches have their limitations and benefits. However, the alternative approach of power beaming[1] may lead to a lower infrastructure on the lunar surface and potentially lower costs as well as offering greater mission flexibility.

A key benefit for power beaming from space to the lunar surface is that the moon has no atmosphere to attenuate the beam. This means that virtually any frequency of microwave or light can be used for power transmission (the exceptions are communications frequencies and, for microwaves, their subharmonics). One difference between microwave and laser wireless power transmission to the moon, is that the photovoltaic cells used for conversion of the laser beam can also convert sunlight during the two week lunar day, while the same is not true for microwave rectennas. As pointing limitations permit, outpost power could be significantly augmented with this option. That study examined issues with beaming both laser and microwave radiation from the L1 libration point. Because of the long beaming distance, two other options were studied: circular equatorial orbits[2] and elliptical polar orbits[3].

A. Lunar Power Beaming from the L1 Libration Point

That past study[1] aimed at producing 10 MW of lunar surface power from the L1 libration point located about 56,000 km above the lunar surface using both microwave and laser power beaming. This power level was very large and should be taken as exemplary only. Those analyses could be extended to lower power levels. The results of each beaming option were briefly discussed. Power beaming to the lunar surface from the cis-lunar libration point can be based on either a single central power source, or a distributed group of satellites. Either microwave or laser light wireless energy transportation could be used in a central or distributed architecture. In either case, the transmission system will be retrodirective and will use a pilot beam from the center of the receiving array to provide the information to steer the power beam and maintain it on the target. However, for laser beaming to the moon, no adaptive optics will be required.

(1) Microwave Power Beaming: Based on technical maturity and frequency, 5.8 GHz was selected as the wavelength for a microwave based power transmission system. Power to the microwave system came from high efficiency, triple junction solar cells. In this design, the transmitting antenna aperture was larger than that required for a similar antenna coupling efficiency of a geostationary-to-terrestrial system. Transmission efficiency η_B for Gaussian beams is related to the aperture sizes of the transmitting and receiving antennas:

$$\eta_B \cong 1 - e^{-\tau^2} \quad (1)$$

and

$$\tau = \frac{\pi}{4}\frac{D_T D_R}{\lambda R} \quad (2)$$

where D_T is the transmitting array diameter, D_R is the receiving array diameter, λ is the wavelength of transmission and R is the range of transmission.

To achieve 90% aperture coupling efficiency, a transmitting antenna of 1.5 km diameter and a rectenna of 4 km diameter was adopted as a compromise between a too-large in-space antenna and a too-large lunar rectenna. The design power of 10 MW output to the lunar station required that 17.6 MW be transmitted, giving a maximum beam density of 20.6 W/m^2 at the center of a Gaussian beam. Furthermore this low intensity may not be sufficient to activate some of the diodes in the rectenna depending on beam pattern.

For this system, the primary lunar surface energy source was a thin film rectenna of 4 km diameter. The rectenna would be assembled from 200 m by 10 m strips stretched on a light frame designed to provide the proper angle of inclination such that the surface of the rectenna is normal to the incident beam. Each panel had its own conversion electronics and will be tied into a collection grid before being fed into the base power system. The system provided the requisite power to the lunar surface, however, the size and design of the beaming satellite was complex and a detailed study was not performed.

(2) Laser Power Beaming; For a lunar laser power beaming system, the size of the receiver is dictated either by the diffraction limited beam or by the amount of auxiliary photovoltaic energy desired from conversion of sunlight. To provide 1 MW of auxiliary power at full illumination required a field of approximately 50 m diameter. Limits to the beam intensity (H) which can be delivered at a given distance is inversely proportional to the square of the distance (R) and directly proportional to the power (P) and the ensemble averaged antenna gain (⟨G⟩). Under ideal conditions, the gain is limited by diffraction and for light of wavelength λ and an antenna of diameter D,

$$G_{DL} = \left(\frac{\pi}{4}\right)\left(\frac{D}{\lambda}\right)^2 \tag{3}$$

The ratio of actual beam intensity to ideal diffraction limited beam intensity, the Strehl ratio (I_{REL}), is defined as $\langle G\rangle/G_{DL}$, and the intensity at a given target becomes:

$$H = I_{\mathrm{Re}\,l}\,\frac{P}{R^2}\left(\frac{\pi}{4}\right)\left(\frac{D}{\lambda}\right)^2 \tag{4}$$

For approximately 1μm wavelength laser light, the diffraction limited spreading is less than the ~50 m diameter photovoltaic field required to provide the desired auxiliary power. Therefore, the lunar receiving photovoltaic receiver will be sized at ~ 50 m diameter. Thus, to continuously provide 10 MW power from the laser system 14.6 MW incident on the surface photovoltaic array is required for a Strehl ratio of 0.9. However, the beam intensity at the receiver may rise to about 4 suns equivalent intensity, so beam expansion may be needed.

B. Lunar Laser Power Beaming from Elliptical Equatorial Orbit

Many options exist for orbits around the moon. We did an initial study[2] using Satellite Tool Kit® V7.1 with altitudes as high as 55,000 km. After extensive analysis, an equatorial Molniya-type orbit with an apogee of 30,000 km and a perigee of 500 km was chosen as it appeared to be the best match of long coverage time over a two year period and a somewhat short time with no coverage whatsoever. The orbital condition required that beaming would occur only when the site was in shadow and the satellite was in sunlight. Of course, beaming when the site and the satellite were illuminated is possible. In this orbit, the satellite can beam power to locations within 45º north and south of the equator when it is in sunlight and the site is in darkness. Based on this initial study that covered a two-year time period from July, 2008 to June, 2009, the maximum length of time with no coverage of the 45º N site by this one satellite for these sites to be 164 hours or nearly 7 days. This would require a massive storage system, assuming the night-time power requirement is the same as in day-time. Over the two year period, there were only ten times when the dark time exceeded 84 hours. Thus a single satellite would not suffice.

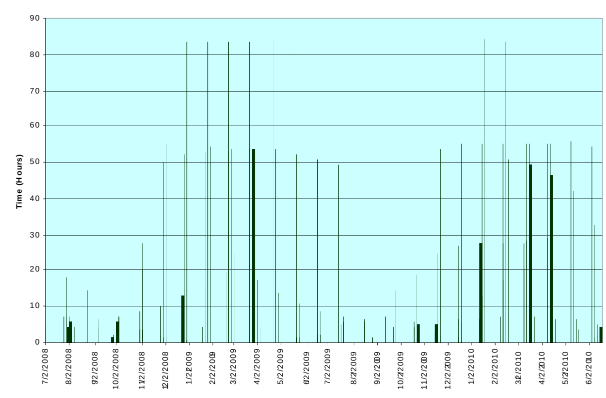

Figure 1. Time spans when neither satellite views a ground station at 45º N.

In order to reduce the dark period length, a second satellite was added. However, one of the critical issues is the angular relationship between the first satellite and the second one. Based on a detailed analysis, a separation of 55º gave the most overlap in coverage. The times when no satellite is able to cover the lunar sites at 45º north or south has dropped dramatically to a maximum of 84 hours (3.5 days) as shown in Figure 1. Furthermore, there are only eight of these periods over the two-year span. In addition, in many cases the duration drops to 54 hours (2.25 days). Both these options represent significant mass savings for a lunar site. It should be remembered that sites closer to the equator will have shorter "dark" times.

Laser beaming from the satellite in this elliptical orbit yields a complex beam pattern and intensity. Fortunately there is no atmosphere so adaptive optics are not required to compensate for atmospheric variations as is the case for earth. For the 45º north site, the solar arrays will be tilted at the latitude. The satellite will rise over the horizon and move about 180º in azimuth. At the same time the satellite will increase in elevation to 45º. A laser power level of 40 kW was assumed, with a wavelength of 850 nm, which is ideal for GaAs cells. Full satellite power was 100 kW. It is important to note that the intensity of the laser beam is approximately 0.2 equivalent suns over most of its traverse and peaks at only 2 “suns”. Assuming a nominal 60 kW GaAs solar array on the surface, the laser beam will not exceed the array dimensions even at the highest altitude. Based on a laser beam power of 40 kW and a 4 m^2 aperture, and with a 45% monochromatic beam conversion efficiency into electricity by the GaAs cells, the power delivered to a the ground site is shown in Figure 2. Beaming from the second satellite will increase this power, depending on whether the array is tracking or not. However, given the eight 84 hour times with no access, energy requirements are still significant, but much reduced.

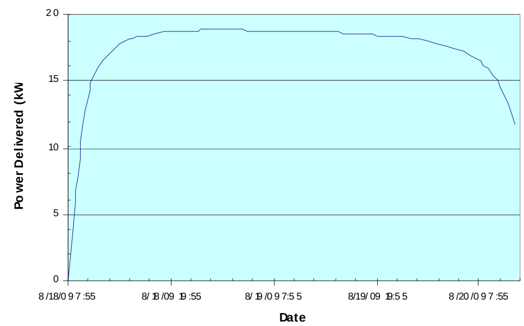

Figure 1. Laser power delivered to a 45º N site.

C. Lunar Power Beaming from Elliptical Polar Orbits

Because NASA’s attention appears to be focused on establishing a camp near the south pole, the previous study was extended to include a polar location as well as the 45º locations. In this study, two satellites in 5000 x 500 km orbits offset from each other by 180º in that orbit were used as the baseline. The orbital time was about 7.5 hours and the apogee was over the lunar South Pole. These orbits were exceptionally stable over time, and can be assumed to be “frozen” orbits about the moon. A GaAs single junction tracking receiver was used, but a fixed array was acceptable, but at a somewhat reduced power. A fixed array would have to be essentially horizontal with the lunar surface which would compromise its ability to capture sunlight. The laser beam aperture was 1.5 m^2 with a wavelength of 850 nm.

Figure 3 shows the access times for this situation. The single satellite provides excellent coverage from the pole to about 30º. Of course the higher the altitude, the longer is the coverage time. Apogee heights of 20,000 to 30,000 km provide the best coverage with a peak of about 340 days. The power delivered to the site(s) also was about 18 kW as noted above, however, now the time when no beaming is possible to any of the sites drops to only 1.5 hours and that is only for the 45º site!

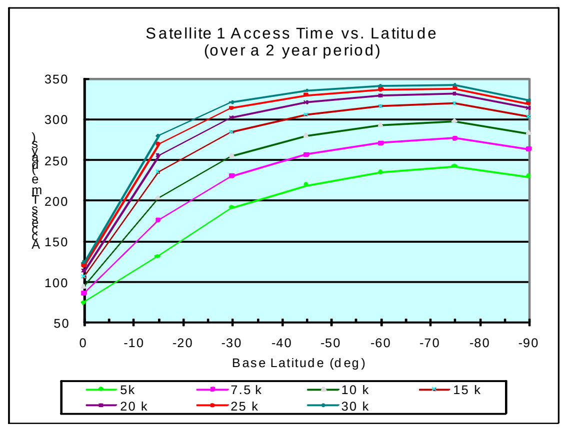

Figure 2. Access time for a single satellite as a function of lunar latitude.

D. Lunar Rover Power

The possible location of a camp at the lunar South Pole on the rim of Shackleton crater was selected to explore that crater for the presence of water. However, the crater is essentially in continuous darkness, hence extensive exploration will require either a rover powered by a radioisotope-fueled source (e.g. with a Stirling convertor), an extremely large battery or fuel cell, or capitalize upon beamed power. All these options are possible, but all have impact on mission duration or other factors. The radioisotope option will provide a limited amount of power indefinitely but will have cost implications as well as using up part of the limited supply of Pu^{238} fuel. However, as the Apollo Lunar Science Experiment Packages (ALSEP) showed, this is a realistic option. Similarly, the fuel cell/battery power source is realistic and inexpensive. However, stay time in the dark crater will be constrained by the amount of hydrogen and oxygen that can be carried for the fuel cell or the charge time of the battery. It’s too early to say what limitation the fixed fuel supply will place on the mission.

The third choice, which can also impact the fuel cell option is to provide beamed laser power to the rover. Because there is no significant atmosphere on the moon, special mirrors won’t likely be required, but there will still be the need for a “guide beam” of some nature to ensure tracking of the rover. Design of such a power system seems straightforward, knowing the efficiency of conversion of the laser light into electricity. If GaAs solar cells are used with a laser wavelength near 850 nm the conversion efficiency will be at least 45% (and perhaps larger due to the low temperatures within the crater). The laser intensity and coverage area can be independently selected. For example, if the solar array on the rover is 1m diameter with a laser beam intensity of 136 mW/cm^2 (AM0), a total power of about 500 W could be supplied when temperature effects are included. The array would include either side reflectors or linear Fresnel lenses to ensure capture of the beam with modest off-pointing due to terrain issues. In addition, a battery would also be a reasonable option to cover times when the line of sight between the beaming source and the rover is obscured by terrain. This power level would roughly equal the power delivered by three Stirling Radioisotope Power sources.

Similarly, the beamed laser power could be used to regenerate hydrogen and oxygen from water produced in the fuel cell and thus increase the mission time of that option. However, the straightforward option of directly powering the rover with a small solar array seems to be the most cost effective. Thus laser power beaming to a rover within a permanently dark crater appears to be a very attractive option. More detailed mission profiles that take into account the terrain of the crater interior and potential "interesting" locations must be developed to assess feasibility of this option.

III. Mars Power Beaming

Electric propulsion has emerged as a cost-effective solution to a wide range of satellite applications and it is a logical choice for Mars exploration as well. Deep Space 1 successfully demonstrated electric propulsion as the primary propulsion source for a satellite. The POWOW (POwer WithOut Wires) concept[3,4] was a solar-electric propelled spacecraft capable of significant cargo and acceptable trip times to Mars. Once at Mars, it would enter areosynchronous orbit (Mars GEO equivalent ~17,000 km altitude) and beam power to surface installations by either lasers or microwaves. Two different approaches were used for the spacecraft: 1) a 10 MW, highly modular vehicle with power, propulsion and beaming modules arranged in 50 kW sections[3] and 2) a conventional designed 898 kW solar array with only laser power beaming from the main spacecraft body[4].

A. 10 MW POWOW Spacecraft

The schematic of the 10 MW (AM0) vehicle is shown in Figure 4. It was made from modular power sections with AM0 power of about 36 W each. These were assembled in hexagonal sections that totaled 8 kW/section. Six of these sections (totaling 48 kW) were then used as a further building block. However, one important aspect of this configuration is that the central hexagon that is surrounded by the six power units housed both the electric propulsion units as well as laser power beaming units (especially lasers). To be specific, in this building block, there were 200 power units and 34 propulsion/beaming units. At Mars, the power level in these units would drop to between 17 and 26 kW due to the distance from the sun and Mars' orbital ellipticity.

The nominal 48 kW units were further assembled into hexagonal sections each having a power level of 1.6 MW with an edge dimension of 48 m. Six of these sections were assembled around a central core which housed the microwave power beaming source. Of course, this section could house another laser power unit instead of the microwave power beamer. Because of the high degree of modularity of this concept, the cost of the solar arrays would drop significantly given that the "learning curve" concept would work for space arrays as it has for terrestrial photovoltaic systems. Were that to be the case, the cost of these arrays would drop by over two orders of magnitude.

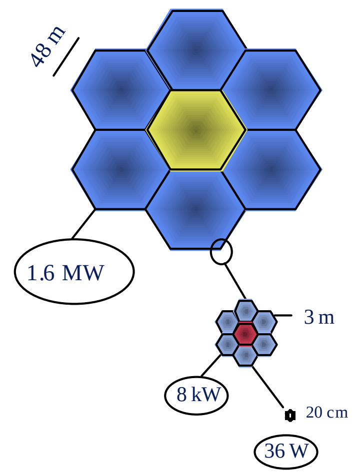

Figure 4. POWOW spacecraft concept

B. 10 MW POWOW Spacecraft Mission Issues

(1) <u>Spacecraft mass and trip times:</u> The spacecraft would weigh on the order of 350 MT in earth orbit! Admittedly this is a concept for the distant future, but the analysis of trip time to Mars illustrated the value of electric propulsion. Several propulsion cases were examined but one having 5 kg/kW for the power system specific power coupled with a 5 kg/kW propulsion system that had an Isp of 4000 s, yielded trip times to Mars of only 279 days. This vehicle also carried an additional 70 MT of payload.

(2) <u>Power beaming options - microwaves:</u> Two microwave options with frequencies of 10 and 35 GHz were examined. Because the areosynchronous altitude is 17,000 km above Mars, severe challenges exist in achieving the goals of this mission. The maximum diameter of the spacecraft was 258 m leading to a maximum transmitter area of just over 52,000 m^2. For beaming applications, a 10 dB Gaussian taper was assumed and a total RF transmitter power of 2.77 MW. Several surprising conclusions were reached as a result of this study. For example, at 10 GHz, for 80% collection efficiency, the rectenna diameter was 3.24 km and 169 kW were delivered to the surface.

However, the key finding was that the rectenna efficiency was only 7.5%! This is caused by the exceptionally low transmitted beam power density. It is so low that the rectenna elements are not being driven hard enough to operate at high efficiency. This would lead to many issues within the diode array where many might not be sufficiently excited to even work, and worse might become a drain on the rest of the array.

The situation did not improve significantly with the 35 GHz case. Once again, at 80% collection efficiency, the rectenna diameter dropped to 0.93 km and 194 kW was delivered to the surface. This is an increase in power over the 10 GHz case, but the rectenna efficiency was still low – about 8.8%. Once more, this was caused by the low transmitted beam power density limited by the maximum diameter of 258 m. Thus microwave power beaming was dropped from further consideration for future studies of this concept.

(3) Power beaming options – lasers: Because of the limitations of the microwave system. optical power beaming was considered. One of the benefits of Mars is that it has virtually no atmosphere - just about 7 millibars of CO_2. Therefore the use of both a 10.6 µm CO_2 laser and a 1.06 µm laser were examined. It is possible that the 10.6 µm laser might excite unwanted modes in the Martian CO_2atmosphere and suffer additional losses, but that issue was not considered in the study. Neither was the need for adaptive optics on the mirror and use of a guide beam to adjust for variations in the Martian atmosphere due to the low density – however that was probably overoptimistic. Furthermore, the infrared wavelengths may not be significantly affected by the dust storms in the atmosphere given the small size of Martian dust. At the time, none of these issues was studied in detail.

The results of that study indicated that about 800 kW could be delivered to the surface. However, some errors were made in the area calculations that need to be corrected. Specifically, for a 1.06 µm laser beam with a 1 m diameter aperture and a Strehl ratio of 0.9, the beam intensity at the surface is about 2.8 mW/cm^2. However, with a 4 m diameter beaming aperture the intensity at the receiving solar cell array becomes 44.3 mW/cm^2 and with a 6 m diameter aperture it rises to 100 mW/cm^2. Assuming 40% conversion efficiency from power to laser beams and an average solar intensity at Mars of 46% AM0, a 90% transmission efficiency and a 45% solar cell conversion efficiency yields a surface power of about 800 kW. Of course this assumes no adverse atmospheric effects. The 10.6 µm laser case would require a thermal conversion system because there are no solar cells that operate at that wavelength. No detailed study was made of thermal engine conversion or thermophotovoltaic conversion subsystems for this laser. Thus the wavelength choice is critical for significant power transfer.

C. 898 kW POWOW Spacecraft

The conceptual design[4] of the 898 kW POWOW spacecraft is shown in Figure 5. This 2001 study used various technologies that were expected to be available in 2003. Each wing was built from eight 3x2 ATK "Aurora" modules, that used advanced cell and Stretched Lens concentrator Array technologies that boosted the total power to just over 56 kW EOL per module. Thus each wing produced 449 kW and has dimensions of 53 m wide by 50 m in length. Each wing has a mass of approximately 2488 kg (180 W/kg). The electric propulsion module is centrally located in the body of the spacecraft and had a 45-degree plume clearance from the closest array element. There are 48 Xe Hall thrusters arranged in a 6x8 configuration. Thirty of the Hall thrusters are active with the remaining sixteen standing in reserve. The dry propulsion module weighed 10,700 kg .

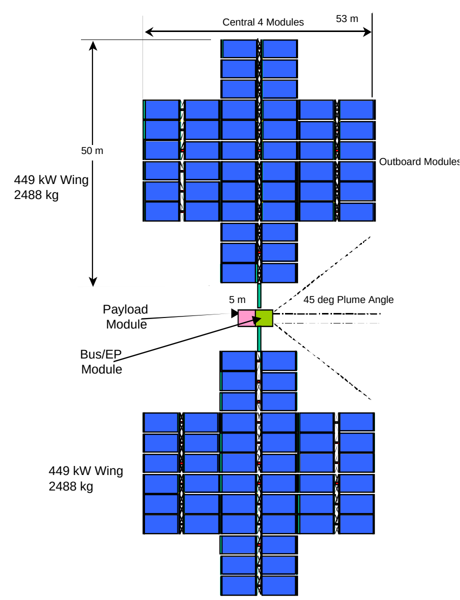

Figure 5. 898 kW POWOW spacecraft

The trip times were calculated assuming that the spacecraft had been boosted to near escape velocity so travel through the radiation belts was unnecessary. A program called All SEP Mars 1.0[5] was used for these calculations. The results of the modeling using the Xe Hall thruster yielded 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 (230 days total). The capture orbit in this case is a lower orbit than areosynchronous so the total time is conservative. The total propellant mass was 24 MT. The trip times obtained were reasonable for payload transits to Mars and were similar to chemical propulsion results.

D. 898 kW POWOW Spacecraft Subsystem Issues

(1) Solar cell and array technologies: The baseline solar cell was 26% efficient triple junction cell operating at 80 °C with 78% of the power remaining (p/p_o) after a dose of $1x10^{15}$ electrons/cm^2. The 2003 projection proposed a 32% efficient, four-junction cell operating at 80 °C under concentration with a p/p_o of 83%. Unfortunately that projection was overly optimistic – but current production technology is approaching that value. These cells would be used in the Stretched Lens Array (SLA) shown in Figure 6. The SLA consists of linear Fresnel lenses made from DC 93-500 silicone rubber, a linear array of triple junction solar cells mounted on a thin composite radiator. The silicone lenses are flexible and are deployed by spring action. Their present lens thickness is about 180 µm but can be substantially reduced. Confirmed lens efficiency is 92% and the concentration ratio is 8.5x. At the time of the study this design produced an array of 180 W/kg. An even lighter weight design, 330 W/kg, is being demonstrated today with further increases likely. Thus these past calculations were somewhat pessimistic on mass. The six panel, 48 kW modules package neatly into a 2m x 2m x 4m envelope.

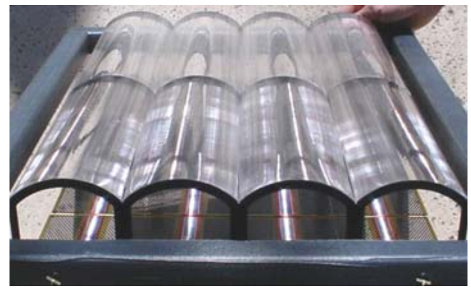

Figure 6. Lightweight stretched lens array (SLA) model.

(2) 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 that study, four suitable candidates were identified from a wide range of propulsion candidates. The selected options fell into two general categories – 25 kW class thrusters and very large thrusters with power levels above 200 kW per thruster. The selected options were: 25 kW Xe Ion (Isp 6000s), 25 kW Xe Hall (Isp 3242s), 256 kW Li Applied Field MPD (Isp 5382s), and 768 kW Li Self Field MPD (Isp 3405s). Trip times were calculated using a program for performing quick estimates of trip times for electrically propelled spacecraft traveling to Mars called All SEP Mars1.0[5]. As noted above, the Xe Hall thruster yield a payload of 4.0 MT, IMHELIO of 44.4 MT and a trip time to Mars of 230 days including capture.

(3) Power beaming technology: For this study, a 1.06 µm laser was used once again due to the extensive work at that wavelength, but a promising new avenue of a 0.9 µm diode bar laser was mentioned. That 192 bar laser diode array that yielded 23 kW of peak power at 0.9 µm wavelength with an efficiency of 43%. This wavelength is much more compatible with GaAs solar cells that the Si cells that might have been used with the 1.06 µm laser. Of course with all laser options, cooling will be a major issue and assessment of that issue was not included. The hope was that as laser technology advances, efficiencies will continue to increase with a concomitant decrease in cooling needs. Using the previous assumptions for the 10 MW POWOW, with an initial power of 898 kW, a 43% conversion of electricity into laser light, an average solar intensity at Mars of 46% AM0, a 90% transmission efficiency and a 45% solar cell conversion efficiency yields a surface power of about 72 kW. That paper failed to include the reduction in power of the satellite at Mars, which has been corrected here.

E. Costing Methodology

Detailed learning curve modeling was done and assumed three levels of production: initial low rate, a second phase that yielded 17 flight units and a final stage producing the rest of the arrays. At the end of production, costs were projected to drop below $200/W at an array level.

IV. Venus Microwave Power Beaming

To the best of this authors' knowledge, no study of power beaming to the Venusian surface has ever been conducted. Because of the thick cloud cover, the use of laser power beaming with a photovoltaic receiver on the surface is not likely due to significant atmospheric absorption of the beam. Secondly, the high surface temperature (735 K or 462 ºC) would render most photovoltaic systems unusable. Finally, the both high surface temperature and pressure have been significant barriers to long-duration surface exploration. However, two factors make Venus a very interesting target for beamed microwave power for robotic exploration. First, the diameter of Venus is 6,052 km (or 95% that of earth. Furthermore, it is located at about 0.72 AU from the sun; hence the solar intensity is about double that of earth, making satellite solar arrays smaller. However, a quick estimate of the Venus Synchronous Orbit (VSO) suggests that it would be approximately 925,000 km from the surface due to the very slow Venusian rotation rate of 6.52 km/hr caused by a sunrise to sunrise time of 116.75 days (not the 243 day transit about the sun). This makes microwave power beaming transmit apertures very large in order to keep the receiving antenna within reasonable bounds.

Recently, SiC, GaN and other types of diodes that can operate at temperatures up to 500 ºC have emerged. These might serve as the basis of a rectenna on the Venus surface. If the receiving antenna is 3 m diameter, the transmitting array area can be estimated for a 5.8 GHz frequency. For the VSO noted above, the area of the transmitting antenna becomes about 3×10^{14} m^2 or a diameter of 2×10^4 km! Obviously this is not realistic for a single antenna.

However, a distributed satellite design may provide an alternative. Although the "curse of the sparse array" is usually cited as prohibiting a dispersed microwave transmission system, it must be noted that the grating lobe spacing is inversely related to the spacing of the radiating elements, and that for very widely spaced virtual aperture elements, the first "n" grating lobes will fall onto the rectenna. A constellation of small microwave transmitting satellites, occupying a regularly spaced grid of halo orbits about the VSO point would form a synthetic transmission aperture. The beam pattern on the rectenna from such an array would be complex, with a central lobe, surrounded by side lobes and then grating lobes. Although a complete analysis of the beam pattern is beyond the scope of the present work, certain important characteristics may be noted. Grating lobe characteristics are given by:

$$R_{GL} = R \sin \Theta_{GL} \tag{5}$$

and

$$n(2\pi) = \frac{2\pi}{\lambda} X_D \sin \Theta_{GL} \tag{6}$$

Where: R_{GL} = range from rectenna center to first grating lobe, R = range from satellite to rectenna (925,000 km), X_D = distance between phase centers on subarrays, λ = wavelength of microwave transmission, Θ_{GL} = grating lobe angle when the outputs from each subarray add in-phase. The calculation is based on the Gobau expression for aperture coupling efficiency in the far field. To achieve close to 90% aperture coupling over that distance would require a transmitting antenna effective diameter of approximately 30,000 km. A solid aperture of such dimensions would make a tremendous solar sail. A sparse array would require an interesting and extreme set of halo orbits to maintain the constellation. To increase the efficiency of a sparse array through spacing the transmitting elements (satellites in this case) such that the first grating lobe falls on the 3 m diameter surface rectenna would require spacing the satellites almost 32,000 km apart, which is larger than the proposed pseudo aperture diameter.

Thus this idea does not solve the problem of power to the surface of Venus – not to mention any atmospheric effects due to sulfuric acid or other components. In fact, given the extreme altitude of the VSO, it is highly likely that the Sun would win out in the competition for the satellite system not to mention the impact of solar pressure!

V. Conclusion

The combination of highly efficient electric propulsion and beamed power are in natural synergy with each other. The high power needed for propulsion becomes the source for power beaming and thus can enhance exploration of the planets. Power beaming to the surface of the moon was shown to be especially feasible using elliptical polar orbits but beaming from equatorial or from L1 proved difficult except for laser beaming. At Mars, the areosynchronous orbit was particularly beneficial. For the case studied here, microwave beaming was limited by the transmitter aperture and was eliminated. Laser beaming was successful, but may require adaptive optics and a wavelength selected that would minimize the impact of atmospheric dust. Neither microwave nor laser power beaming to Venus was possible from the Venus Synchronous Orbit due to its large distance from the surface.

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