L•GARDE
SMART SPACE TECHNOLOGY

L • GARDE INC. CORPORATE PRESENTATION

A Power Antenna for Deep Space Missions

Costa G. Cassapakis and Mitchell Thomas

The American Society of Mechanical Engineers

Reprinted From
Solar Engineering
Editors: J. H. Davidson, and J. Chavez
Book No. HO1046 - 1996

A POWER ANTENNA FOR DEEP SPACE MISSIONS

Costa G. Cassapakis and Mitchell Thomas
L'Garde, Inc.
15181 Woodlawn Avenue
Tustin, CA 92680-6487
714-259-0771

ABSTRACT

Solar power becomes too weak to power spacecraft missions to the outer planets. However, by using a large antenna also as a solar concentrator, useful power levels can be obtained in deep space. This paper presents the results of a design study that determines the effectiveness and size of such a multiple-use reflector. Solar ray tracing defined the size and shape of the solar energy pattern on the power system. The concentrator can also be used to maintain spacecraft temperatures near 390 K throughout its flight. The weights of such a power antenna is competitive with RTG's for the closer missions, but is heavier for the far planets. Further design trades are needed to reduce system weights, and some candidate approaches have been identified to accomplish that.

NOMENCLATURE

D	Location of power system (distance from focal plane measured toward reflector)
RF	radio frequency
RTG	Radioisotope thermalelectric generator
f	focal length
d	diameter.
r	radius

INTRODUCTION

Missions to the far planets currently are powered by radio-isotope generators (RTGs). These devices are very expensive, and may become unavailable because of the decreased activities of the DOE laboratories. A need exists for alternate power systems. However, the weight of solar cells to provide deep-space power becomes very large. We have proposed a system that uses a high degree of solar concentration to reduce the weight of the cells needed. This reflector can also be used to keep the spacecraft warm, and even to telemeter mission data back to Earth. This multi-use reflector was studied under a recent NASA/JPL SBIR contract and we present here a summary of the results of the analysis of the solar power systems. More complete data is presented by Cassapakis, et al (1995).

Figure I shows the concept of the power antenna (Joel Sercel, NASA/JPL). Based on our analysis of weights, volumes, and mechanizations, the power antenna appears to be a feasible and capable substitute for an RTG, at least for a low power, microspacecraft Jupiter and a Saturn mission, with the possibility of providing electrical power throughout the transit from Mars, spacecraft heating, and very fast data transfers. Masses of 20 to 40 kg are foreseen, not much different than an equivalent RTG. Fig. 2 shows the baseline design resulting from the study to scale for the IO fly-by mission ($f/d = 1$, d = 5.24m).

The power antenna is a workable concept either for short term fly-by, or longer-term rendezvous microspacecraft missions with low-power requirements, based on the assumption that with a double bumper shield on either side of the lenticular reflector/canopy structure, the damage to the third film is negligible. Furthermore, the power antenna concept is feasible for the far outer planets (Uranus and Neptune), but the mass needed has been estimated in this study to be larger than 100 kg (See Fig. 3). More work is needed to define how this mass can be reduced through more efficient support structure and going to lower inflation pressures, and using more efficient inflation systems.

Beyond direct power-antenna considerations, there are a host of ancillary spacecraft technology considerations for the described mission concepts. Such considerations play a key role in determining power-antenna feasibility. These include fault tolerance and reliability requirements and capabilities for off-sun events and pointing anomalies. While beyond the scope of the power-antenna requirements addressed in this paper, such considerations and technologies needed to address them are enabling for applications of power antennas in many of the applications described here, especially for those missions beyond the orbit of Mars.

Prepared for the 1996 ASME International Solar Energy Conference.

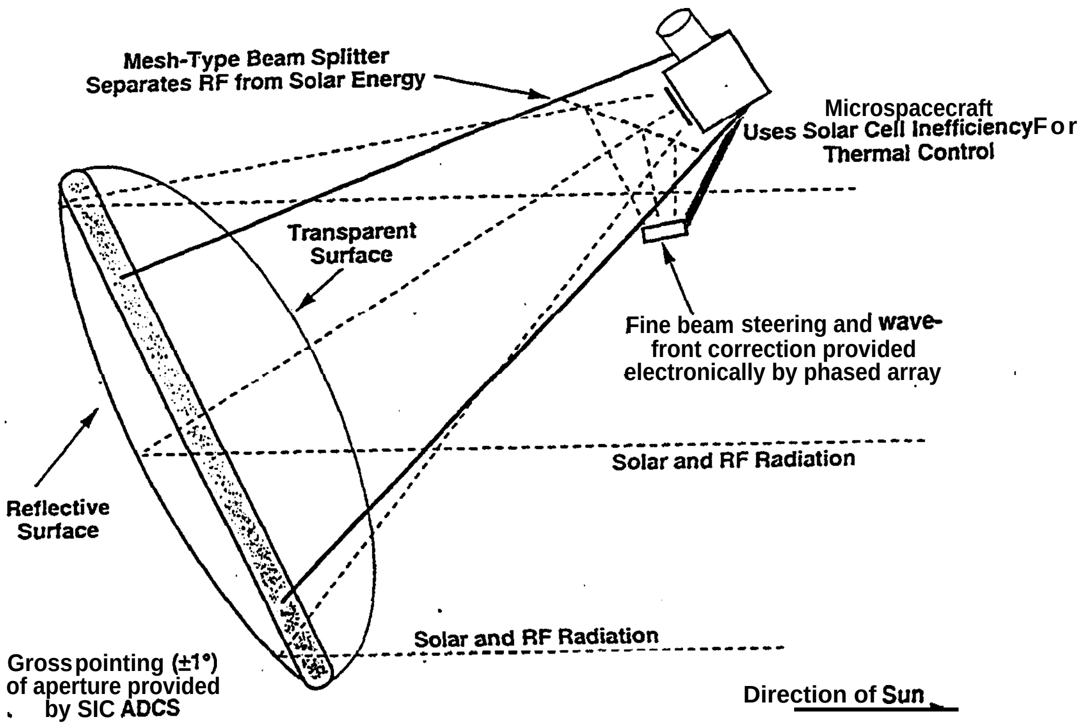


Fig. 1. Power Antenna **Initial Concept**

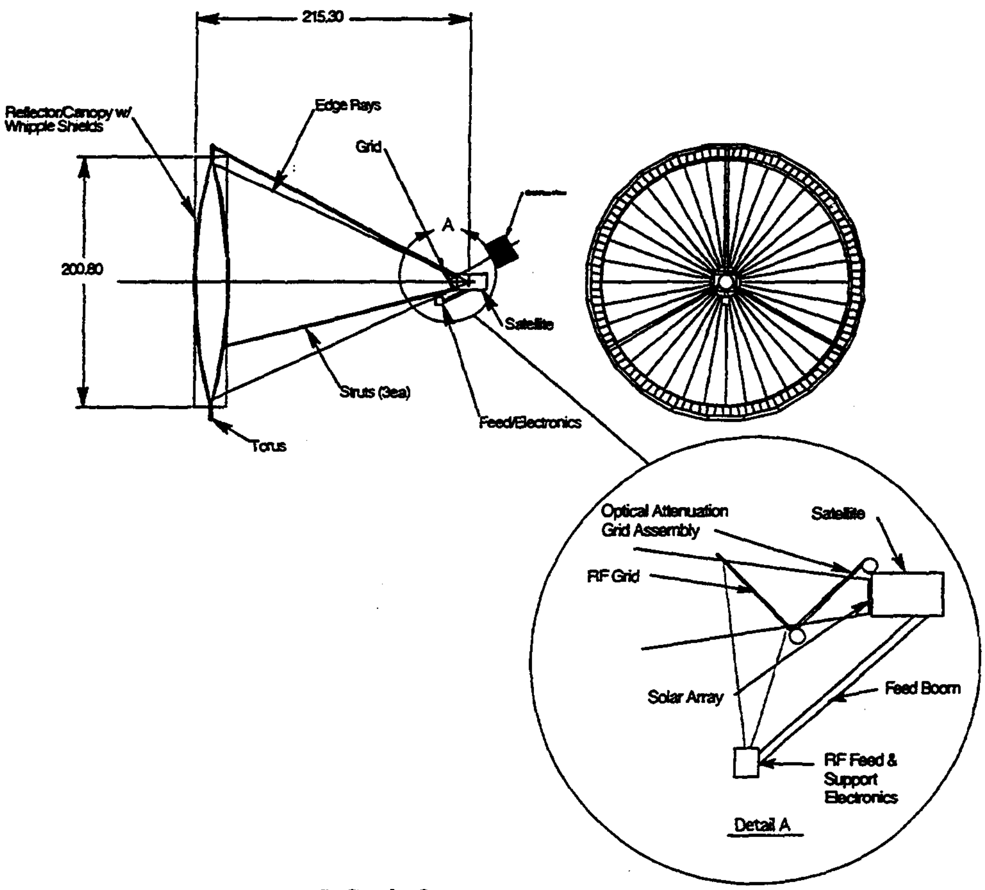


Fig. 2. The L'Garde Concept of the Power Antenna

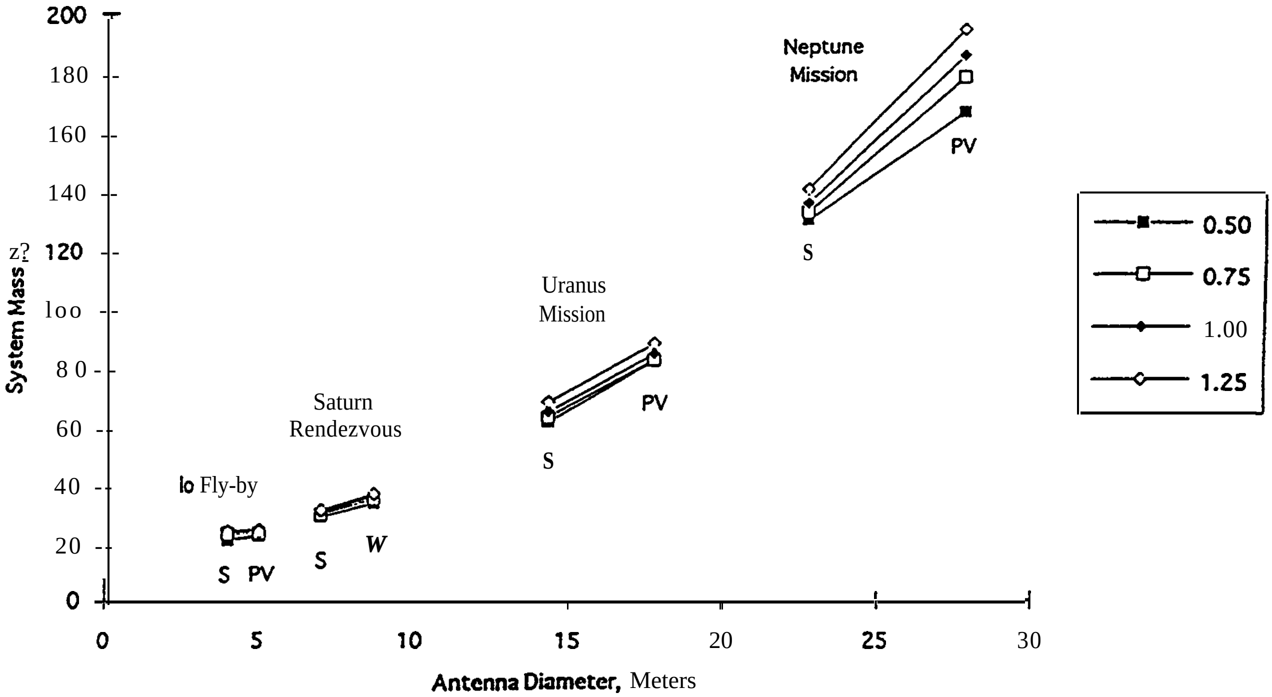


Fig. 3. Power Antenna Mass vs. Diameter Summary for Various f/d

Summarv of Solar Analvsis Results

We have evaluated all the efficiency factors for reflector performance when operating with standard GaAs cells. The total efficiency is about 0.0767. For a 75w power system, the resulting reflector diameters range from 5m (Jupiter) to 28m (Neptune). Using an alkali metal electrical generator, the efficiency improves and the corresponding diameters arc 4 and 23 m, respectively.

Solar Array Svstem

The array was sized to fit on the microspacecraft, 40 cm in diameter. For all the following analysis the COPS ray-tracing code was used (obtained from Sandia NL). The placement of the reflector was studied to get the most uniform illumination combined with high efficiency. The resulting intensity distributions were quite uniform, typically varying from 1 to 3 kw/m^2 for f/d = 1 and from I to 6 kw/m^2 for f/d = .5. A nominal surface accuracy of 1 mm rms was assumed but surface error was not important for this configuration of solar concentrator, although it was very important for the antenna use. Predicted intensity distributions on the solar array were calculated for various placements of the antenna, the two values of f/d, and the distance from the sun.

For f/d's of 1.0 and 0.5, the corresponding rim angles are 53.1301 and 28.0725°, respectively. The solar array is 40 cm in diameter. For the above rim angles, the array should bc located about a distance, D, of 0.375m and 0.15m away from the focal point toward the reflector, for the two cases. This geometry was modeled using the COPS computer code (COPS, 1980). COPS is a ray tracing code that considers the finite size of and limb-darkening on the sun. For this program, we modified COPS to find the intensities at planes displaced from the focal point, and also to allow consideration of the systematic error described above.

Figure 4 shows COPS calculations for the nominal value of D = 0.375m position of the array, and for two variations about this value. In this COPS run, 40,000 random sun rays were chosen. A perfect reflector was assumed. As seen on Fig. 4, the nominal value of -.375 for array placement is a good choice with a high flux evident over the 20 cm radius shown. Moving closer to focal point gives higher fluxes, but they are not very uniform. Moving further from the focal point reduces the fluxes without much increase in uniformity. Therefore, the nominal values for the array displacements from the focal point, calculated from the geometry, were used for all subsequent calculations on the array.

COPS allows blocking to bc considered, in this case blocking by the 40-cm-diameter spacecraft. Figure 5 shows the effect of blocking on the incident intensity. The center of the array is significantly shadowed for this case. Therefore, blocking was included in all subsequent calculations.

Figure 6 compares two COPS calculations with and without a surface error. For this case and all subsequent calculations, we used 200,000 rays. We do not expect surface errors much larger than the 2 mrad slope error shown. This size error has minimal effect on the illumination of the solar array. Therefore, we assumed a perfect reflector for-the remaining calculations for the case of the solar array.

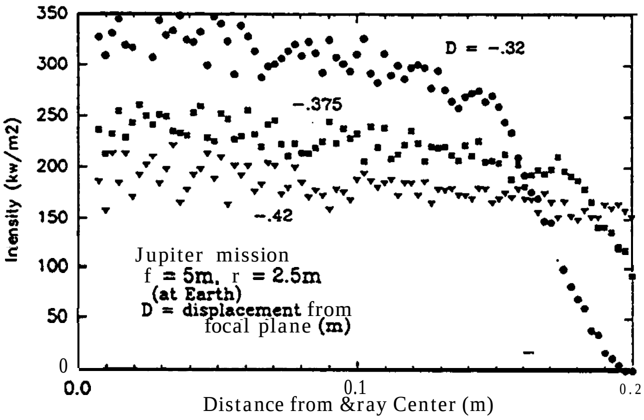
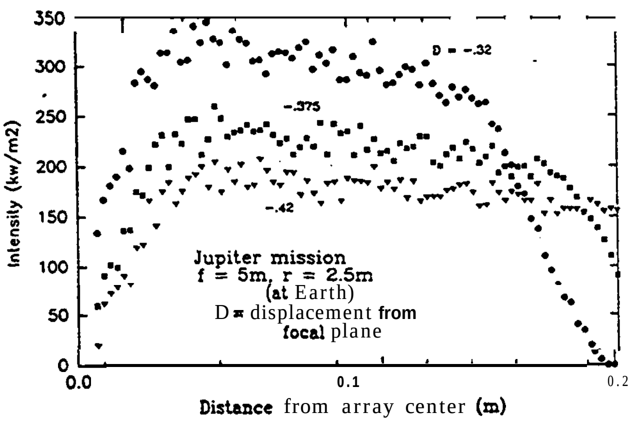


Fig. 4. Array Intensity, No Blocking



Fii. 5. Amy Intensities, With Blocking

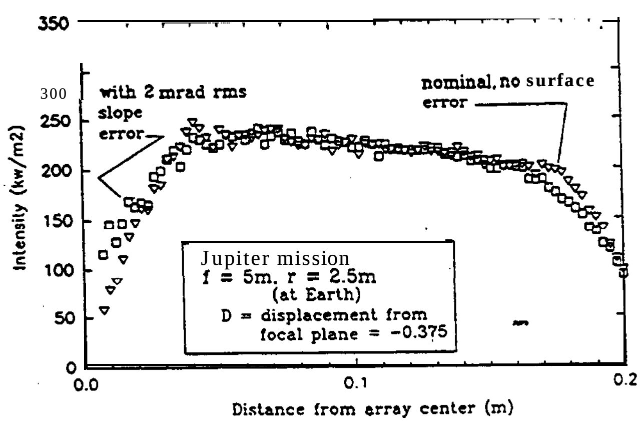


Fig. 6. Array Intensities, With Blocking

Jupiter Mission. Figure 7 shows the calculated power incident on the solar array for the f/d = 1 case, both at the Earth and at Jupiter. Very large powers are incident near the Earth and some sort of shielding is clearly needed. Figure 8 shows similar data for the case of f/d = OS. For this case, the incident intensity is much less even than for the longer focal length. The my tracing calculation assumed a RF grid shadowing efficiency of 0.88. This, plus reflection and transmission losses produce the effective concentrator reflectivity of 0.4166. Similar data were derived for the Saturn, Uranus and 2 Neptune missions (Cassapakis, et al, 1995).

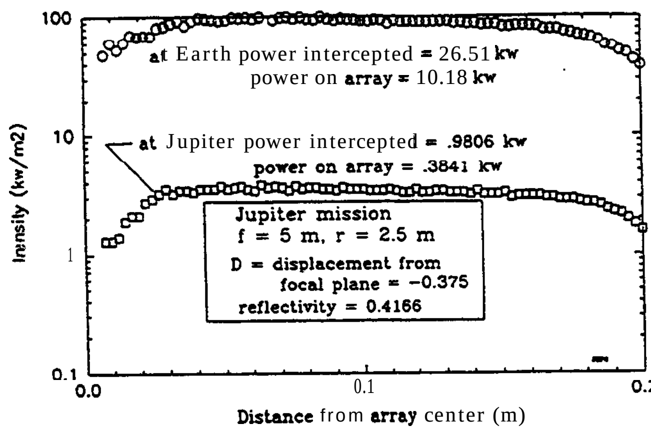


Fig. 7. Array Intensities, With Blocking

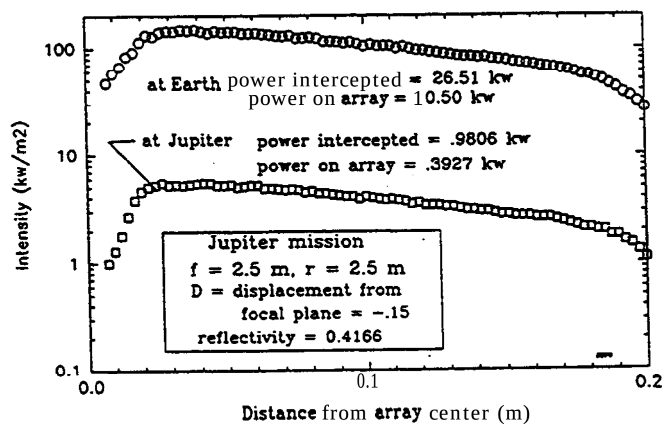


Fig. 8. Array Intensities, With Blocking

The intensity incident on an array designed to go to deep space is very high near Earth. For instance, for the Saturn mission, it peaks at 500 kw/m^2 near Earth. Therefore, an attenuation scheme is needed if the power is to be available throughout the flight. We looked at repositioning the spacecraft so that the reflected energy misses most of the spacecraft.

Figure 9 shows the distribution of flux for a perfectly aligned reflector designed for the Jupiter mission. The solar array is a circle of radius **0.25m** centered at **(0,0)** in this figure. One approach to reduce this intensity is to rotate the spacecraft slightly so that the large flux is shifted off of the array. Figure 10 shows that this approach creates a very uneven heating of the array. Here, the antenna was **misaligned** by 4 degrees in an attempt to allow only a **small** fraction of the collected energy hit the solar array. This rotation actually increases the concentration of the solar energy in the section of the beam nearest the array center. As shown in Fig. **10,** very steep gradients of solar energy occur so that some of the solar cells **will see** concentrations of almost 100, while adjoining cells may see only 1 sun or less. Further work is needed to see if the solar array can be designed to obtain useful power under these conditions.

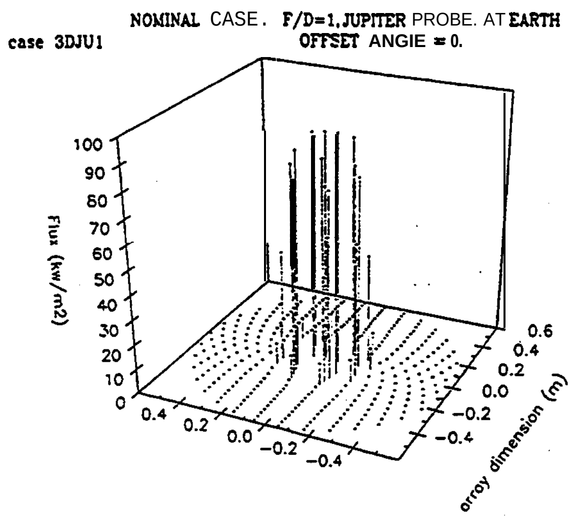


Fig. 9. **Flux** Distribution **- Reflector** Misalignment **= 0°**

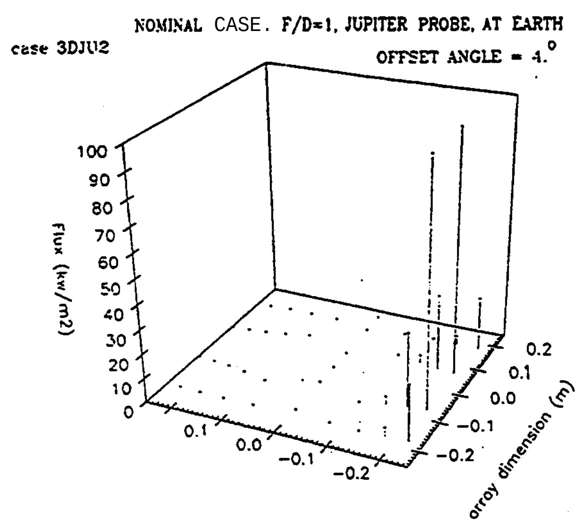


Fig. 10. Flux Distribution - Reflector Misalignment = 4°

Another approach to solving this problem is by using the grid subsystem shown previously in Figs.1 and 2. This subsystem is mounted to the spacecraft and deployed after erection and setup of the Power Antenna, but before reflector inflation. A fixed RF **reflecting** grid **reflects RF** energy to the feed system. Behind the RF grid is a **solar** attenuation grid. It consists primarily of a **roll** of Kapton thin film fastened to a **roller** feed and take-up assembly. The **film** is advanced throughout the mission. The Kapton **film** has a half tone reflective metal or white ink on it that constantly changes. The dots are very dense at the beginning of the mission when too much solar energy is being received. Upon arrival at the planet of **interest,** the thin film would be clear and without dots. With the solar attenuation grid, the Power Antenna produces 75 watts of electrical power regardless of its distance from the sun.

A continuously advancing **roll** always exposes new RF grid material. Therefore, meteoroid or heat damaged material would be refreshed as **the** mission progresses. The grid can be adjusted for unexpected anomalies in the optical **performance;** that is, the grid can be rewound to a previous area if needed.

The solar attenuation grid can operate from **Mars** to about Jupiter and possibly Saturn. If the Power Antenna aperture were sized for a planet further away, the grid's Kapton would not **survive** the sun's heat when the Power Antenna is near Mars.

Cevitv Absorber

A solar-heated cavity to drive an alkali metal power system required more accuracy on the reflector and was more **efficient.** We saw that the size of the solar disk influenced the cavity size for the Jupiter mission, but was negligible at the far **planets.** We have calculated the cavity size needed as a function of focal length and mission range for a nominal 1 mm **rms surface** accuracy. As expected, the cavity width increased as the rms accuracy degraded. At Neptune, the diameter is 2.6 cm for a 1 mm accuracy and 10 cm for a 4 **mm** accuracy.

For the power system that requires a high-temperature source, the system is located at the focal point of the concentrator. Because this system is more **efficient** than the solar array, a smaller concentrator is need. Figure **11** shows COPS ray tracings for the Jupiter mission, at the **Earth,** for eases of a **perfect reflector** and one with a 1 mm absolute surface error. For this size reflector, this **absolute** error corresponds to a 2.271 mrad systematic rms slope **error.** Figure 11 shows the two cases for this **slope** error assuming a systematic and random error. As shown, there is little difference between the results, For the remaining **calculations** of concentrator performance, a systematic error corresponding to 1 mm absolute was assumed. The **perfect** reflector shows a jagged response that apparently corresponds to the solar limb-darkening model used in COPS.

Jupiter Mission. Figure 12 shows the power on the **power-**system aperture as a function of the aperture radius. These arc not kw/m^2 as shown in previous graphs, but is the integrated power from the center of the cavity out to a radius r in kw. The **shapes** of the curves for near Earth and near Jupiter differ because near the **Earth, the** sun's radius is about 4.5 **mrad,** which is significantly greater than the surface slope error. Therefore, the sun is being roughly imaged. However, at Jupiter, the sun's radius has decreased to only 0.9 **mrad;** it is acting **like** a point source.

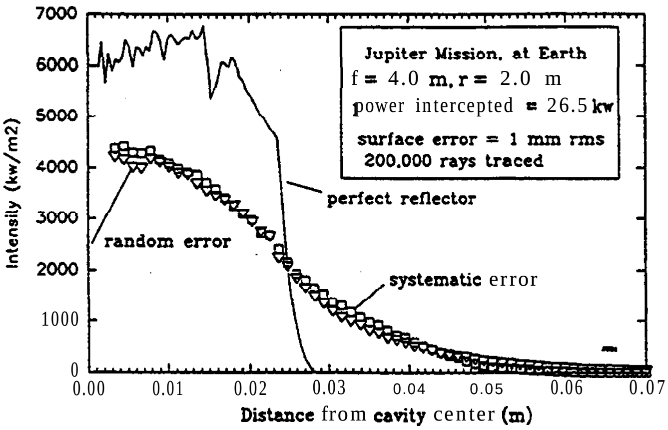


Fig. 11. Intensitiis, With Blocking

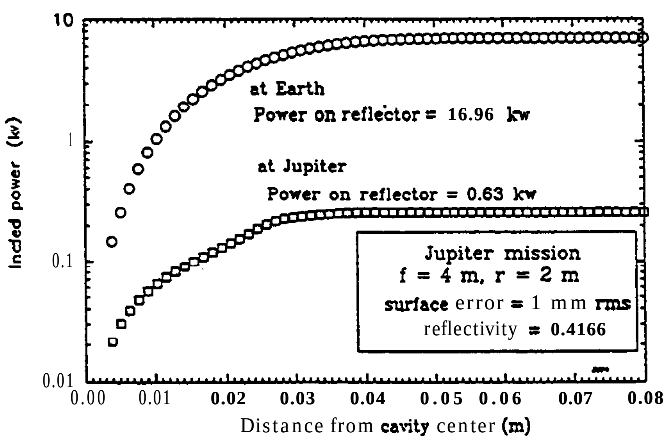


Fig. 12. **Concentrated** Power, With Blocking

Figure 13 shows the data for both the long and short focal-length wnwntrators. Note that the vertical scale is now linear. The radius at which 90% of the concentrated power is included is 0.029m (1.1 in) for the long focal length and 0.0114m (0.45 in) for the short fwal length. Thus, a very small cavity should suffice for the nominal surface error assumed

The shape of the power curves for the Saturn, Uranus, and Neptune is essentially the same as for the Jupiter mission (Cassapakis, et al, 1995).

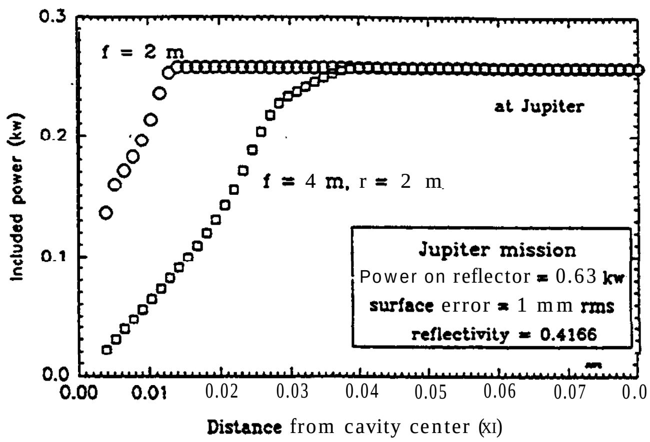


Fig. 13. Concentrated Power, With Blocking

CONCLUSIONS

We have conducted a preliminary feasibility study on the concept of the Power Antenna. In this SBIR Phase I effort we have studied the issues wnwming the Power Antenna's concurrent operation as a solar concentrator and RF antenna, within the constraints imposed by the small mass and volume of a Small Satellite (or microspawcraft). Thus, we have also conducted a large number of conceptual point designs to assess the parameters that strongly affect the masses and volumes of Power Antennas as a function of their distance from Earth. We have concluded the following:

- The Power Antenna is eminently feasible for use as both a power source (electrical and thermal) and a communications antenna for near Outer-Planetary missions.

- An RF reflector grid and a solar energy attenuator can be made to work optimally in a synchronous mode to provide the required power (75 W electric was assumed) and improved RF link margins to Earth. An f/d of I is highly recommended. In addition, a large amount of waste heat is present for spa-raft warming at all times. Should these **requirements decrease, Power Antenna mass and volume will decrease.**

- The solar wncentration profiles on the body mounted solar array arc fairly flat producing a rather uniform illumination of the array.

- The Power antenna masses and volumes for the low-power, microspawcraft Jupiter and Saturn missions are close to what an equivalent RTG would possess, but the price for the Power Antenna is estimated to be an order of magnitude smaller than that of an RTG.

we can show that we can obtain reasonable performance at low stress levels, or by the use of low modulus materials, the mass of the these power antennas is reduced by as much as a factor of three. Furthcrmorc, the make up gas requirements for this case arc also minimal.

Additional work is needed, but the Power Antenna does hold promisc of being practical for microspacecraft missions in the not-too-distant future.

ACKNOWLEDGEMENTS

The work was supported by NASA/JPL (Contract NAS7-1339); the JPL Program Monitor was Bill Niesmith. Significant contributions to the study were made by the following L'Garde personnel: Gil Friese - Systems Analysis, Dr. Allan Love - Antenna Design, Patrick Malone - Mechanical Design, and Dr. Arthur Palisoc - Thermal/Structural Analysis.

REFERENCES

Costa Cassapakis, et al, "A Triple-Use Microspacecraft Subsystem for Outer Planetary Missions (Power Antenna), L'Garde Report LTR-95-CC-026, June 1995.

"Survey Mirrors and Lenses and Their Required Surface Accuracy." Vol 2: Concentrator Optical Pcrformancc Software (COPS) User's Manual. Final Report for Sept. 15, 1978 - Dec. 1, 1979. Report DOE/CS/35348-T1 (Vol 2), January 1980, US Dept. of Energy.

L•Garde Inc
15181 Woodlawn Avenue
Tustin, CA 92780-6487
www.LGarde.com