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Feasibility Assessment of Microwave Power Beaming for Small Satellites

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While wireless power transmission to fulfill Earth's energy needs has been widely popularized as a potential application of microwave power beaming, one space application that has remained relatively untouched is power beaming between satellites. This paper provides a system-level analysis illustrating the feasibility and limitations of power beaming within a small-satellite cluster. To accomplish this analysis, the simple case of a two-spacecraft system is examined. Parametric models of spacecraft power requirements as a function of eleven design variables allow for an extensive trade-space evaluation, and analysis is divided into four segments. First, the existence of feasible designs in the context of the small-satellite problem is verified with a Monte Carlo sweep of the design space. Next, a feasible baseline (reference) design is defined, and sensitivity of that baseline to individual variables is assessed. Finally, the design space is visualized with respect to distance between spacecraft, antenna diameter, and power independence factors. Despite optimistic assumptions in the setup of the problem, it is demonstrated that the small satellite power beaming design space is severely constrained. Only 6% of the design space falls under a suggested 250 W small satellite power constraint. Designs that are feasible involve very high transmission frequencies (>33 GHz), large antenna diameters for a small satellite (>0.93 m), and stringent proximity operations between satellites (within 740 m). Furthermore, full dependence of one spacecraft on power provided by another is shown to be effectively infeasible. These results do suggest, however, that inter-spacecraft microwave power beaming may deserve some consideration as a supplementary power mode for future small-satellite clusters in short-term emergency or atypical situations.

Nomenclature

DC	=	Direct Current	m	=	spacecraft mass
G_r	=	gain of receiving antenna (rectenna)	P	=	spacecraft power requirement
G_t	=	gain of transmitting antenna	$P_{DC,rec}$	=	usable power output of Spacecraft #2 rectenna
L_{ra}	=	rectenna collection efficiency	$P_{DC,trans}$	=	power input to Spacecraft #1 transmitter
L_{rl}	=	rectenna RF-to-DC efficiency	RF	=	Radio Frequency
L_s	=	space loss	X	=	sunlit period power independence
L_{ta}	=	transmitting antenna efficiency	Y	=	eclipse period power independence
L_{tl}	=	transmitter DC-to-RF efficiency	Z	=	general power independence

I. Introduction

SINCE the days of Nikola Tesla, efficient wireless power transmission has been highly sought-after as a technology enabler for a variety of engineering applications. One widely-examined method of wireless power transmission is microwave power beaming, and one popularized application in the aerospace world has been that of transmitting solar energy from space to fulfill Earth's power needs. While studies have often found that this application is likely infeasible (and certainly economically unviable) with current technology, a space application that has remained relatively untouched is microwave power beaming between small satellites. In particular, the Defense Advanced Research Projects Agency (DARPA) has recently expressed interest in such inter-satellite power beaming as a potential enabling technology for the concept of a fractionated spacecraft.[1,2,3]

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This paper provides a system-level analysis of the feasibility of microwave power beaming for the small-satellite problem. Following a brief literature review on the topic, assumptions and models are described, and several parametric results are discussed which support the conclusion that, although impractical as a primary source of power for a small-spacecraft cluster, inter-spacecraft microwave power beaming may find a role in future small-satellite clusters as a supplementary power supply for short-term emergency or atypical situations.

II. Brief History of Microwave Power Beaming

In the 1960s and 1970s, the burgeoning space industry and particularly Dr. Peter Glaser took notice of microwave power beaming in its potential use in for transmitting massive amounts of solar power (measured at the gigawatt level) to Earth from space.[4] While much too ambitious for the space capabilities of the 1960s, Glaser's solar power satellite concept added impetus to several development efforts of the 1960s and 1970s. At Raytheon for example, William Brown demonstrated power beaming on smaller scales, from small microwave-powered helicopters (at power levels around 200 W) to laboratory efficiency tests at power levels around 500 W and tests at the JPL Goldstone Deep Space Network facility at the level of 30 kW.[4,5] In 1976, a milestone was reached when a 91% rectenna absorbed-to-DC-power conversion efficiency was demonstrated for low (on the order of 10 W) power levels. [4]

Notably, the mid-1980s saw the development of a Canadian unmanned aerial vehicle named SHARP (the Stationary High-Altitude Relay Platform) which in 1987 became the first aircraft to achieve sustained flight powered only by beamed power. The SHARP-5 test vehicle utilized 150 W to stay aloft at about 300 ft altitude, with power received from a 10 kW microwave beam tracking the aircraft from the ground. [4,6]

In 1997, NASA initiated a "Fresh Look" study to reassess the solar power satellite concept. While the study concluded that solar power satellite systems are more technically feasible today than they were a few decades ago, it also showed that their economic viability remains dubious.[7] In 2004, the Auburn University Space Research Institute and Texas A&M University Center for Space Power presented analyses of large beamed power systems for lunar and Mars exploration.[8,9] Most recently, in October 2007 the National Security Space Office released a study on space-based solar power which strongly recommended that the U.S. Government invest in an in-space proof-of-concept demonstration within the next decade. One potential use of even only megawatt-scale power beaming was recognized to be the supply of energy to U.S. forward bases which typically rely on expensive (and dangerous) convoy resupply.[10]

As a final note, in 2007 DARPA called for proposals to develop the F6 concept (Future Fast, Flexible, Fractionated, Free-Flying Spacecraft united by Information eXchange), a challenge to the paradigm of traditional monolithic spacecraft aimed at architecturally improving flexibility and responsiveness of Earth-orbital space systems. In 2006, Brown and Eremenko described fractionation as the decomposition of a space system into modules which interact wirelessly to deliver the capability of a functionally equivalent monolithic system.[1] Brown and Eremenko suggested that power beaming can be an enabling technology of fractionation and is deserving of an in-depth analysis. The present work provides one contribution within this context and in this direction.

III. Modeling

A. Assumptions and Power Paths

The analysis which follows is intended as a first-order feasibility analysis. To accomplish this, the simple case of a two-spacecraft system is examined in terms of the power requirement after solar power is converted to direct current (DC). It is assumed that the two spacecraft lie in a perfect line of sight with the capability of continuous microwave power transfer and no pointing losses. The first spacecraft carries all power transmission equipment, and the second spacecraft carries power receiving equipment and the payload[‡].

The assumed power pathways are shown graphically in Figure 1. Power is generated by solar arrays on Spacecraft #1, and that power is regulated to the proper direct-current (DC) voltage and current level. Enough power is generated to power the internal subsystem loads of Spacecraft #1 as well charge its battery. Additionally, part of Spacecraft #1's power is transmitted at microwave frequencies to Spacecraft #2, where it is received (after space losses) by a rectenna. Power from the rectenna is added to power generated independently by Spacecraft #2 to power internal subsystem loads plus those of the payload. Internal spacecraft loads are determined based on typical satellite power-consumption estimation relationships[11] and are assumed to be constant with time.

[‡] Note that having Spacecraft #1 carry the payload would leave Spacecraft #2 without a purpose in this simple scenario, which is why the payload is assumed to be carried by the power-receiving spacecraft.

Power losses are considered throughout the power distribution chain, especially in the power beaming hardware. Losses occur through the transmitter (e.g. a magnetron or other microwave-generating device), antenna (due to dish imperfections and feed array interference), and rectenna (in terms of percent power received and percent converted to useful DC power). Wiring losses are budgeted as internal loads, and a path efficiency is associated with battery discharge during eclipse.

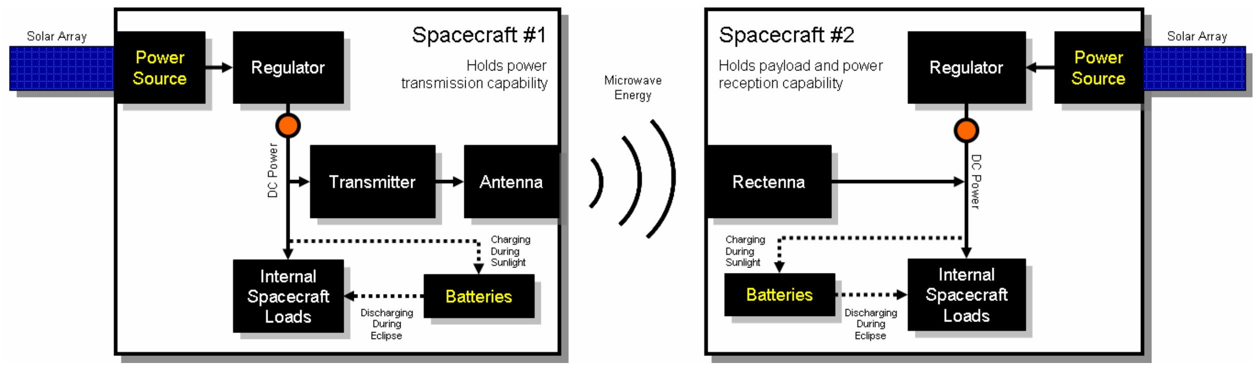

Figure 1. Power Path Model used for this study.
Orange dots indicate "pick-off points" at which DC power requirements in this study are reported.

The degree of power beaming fractionation is considered on a continuous basis from 0% to 100%. In the present study, a parameter Z named power independence is introduced and is defined in Equation 1 below. Z can have values ranging from zero to unity. As defined in Equation 1, when Z is zero, the payload-carrying spacecraft (Spacecraft #2 in Figure 1) draws all its power from power beaming (such a spacecraft would have no solar arrays and no batteries). When Z is unity, the payload-carrying spacecraft is entirely power-independent and has no need for power beaming. For all cases within $0 < Z < 1$, the payload-carrying spacecraft also carries solar arrays, batteries, and a rectenna and receives a certain percent of its power from each (although solar arrays and batteries do not provide power at the same time).

$$Z = \frac{\text{Power Generated by Spacecraft \#2}}{\text{Total Spacecraft \#2 Power Requirement}} \tag{1}$$

It is recognized, however, that a one-dimensional degree of power fractionation may be insufficient to capture all principal effects. Because it is typically more costly to power a spacecraft during eclipse (due to the size of batteries and the fact that batteries must be charged during sunlit periods), two new power independence parameters are introduced in the present study, namely X and Y where X is Z evaluated during the sunlit period and Y is Z evaluated during the eclipse period. Both X and Y have values ranging from zero to unity (see Equations 2 and 3 below).

$$X = Z\big|_{sunlit} = \frac{\text{Power Generated by Spacecraft \#2 during Sunlit Period}}{\text{Total Spacecraft \#2 Power Requirement during Sunlit Period}} \tag{2}$$

$$Y = Z\big|_{eclipse} = \frac{\text{Power Generated by Spacecraft \#2 during Eclipse}}{\text{Total Spacecraft \#2 Power Requirement during Eclipse}} \tag{3}$$

One final assumption of note is that space losses are assessed via the variant of the traditional link equation shown in Equation 4 below (see nomenclature for definition of terms). However, in some cases analyzed next, the spacecraft are in too close proximity for the traditional link equation to be valid (the key assumption of the link equation is that the receiving spacecraft is in the far-field with respect to the transmitting spacecraft). In cases where this is an issue, the model used in this study optimistically assumes that there are no space losses.

$$P_{DC,rec} = P_{DC,trans} L_{tl} L_{ta} G_t L_s G_r L_{ra} L_{rl} \tag{4}$$

B. Historical Spacecraft Power/Mass Correlation

Since this paper aims to assess the feasibility of power beaming for small satellites (defined here to be 300 kg or less in wet mass), the final modeling consideration is how the power values obtained may be converted to spacecraft mass. Rather than use a full set of mass estimating relationships to accomplish this, this study uses a curve-fit relating mass and power for previously flown Earth-orbital satellites.

Power and mass data are compiled from 28 previous satellites[11,12,13,14] and result in the curve fit shown in Figure 2 and Equation 5. The R^2 value for the curve fit is 0.739 and predicts a power of 250 W for a 300-kg spacecraft. This 250 W number is not a rigid definition of a small satellite but is rather meant as a rough estimate of the power one would expect a small satellite to consume. Since this study only computes spacecraft power requirements and does not directly compute spacecraft masses, this 250 W number helps to bound the trade space which will be shown next.

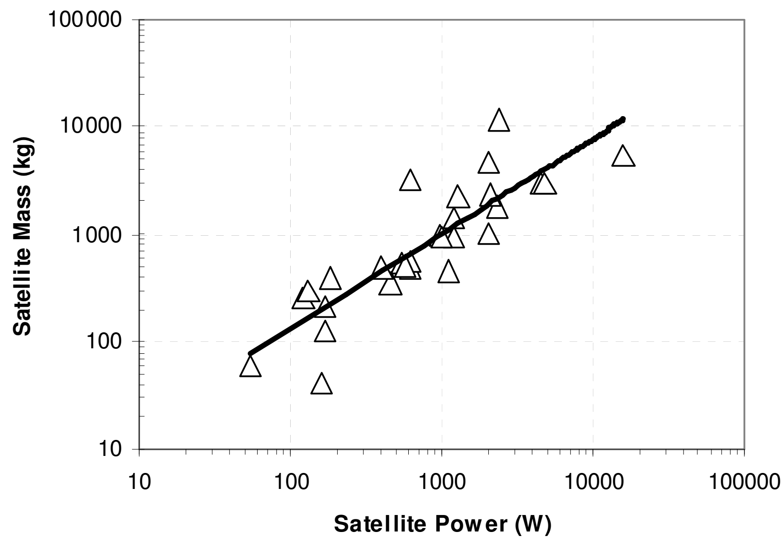

Figure 2. Historical Spacecraft Power/Mass Data (including curve fit).

$$m = 2.3175P^{0.8805} \tag{5}$$

IV. Results and Discussion

The models defined above allow for both an analysis of a single point design and, more importantly a parametric trade-space evaluation for a wide range of possible designs. As such, analysis is divided into four segments. First, the existence of feasible designs satisfying power generation constraints within the context of the small-satellite problem is verified with a Monte Carlo sweep of the eleven-variable design space. Next, a feasible baseline (reference) design is defined, and sensitivity of that baseline's feasibility to individual design variables is assessed. Finally, cross-sections of the design trade space are visualized through plots in the dimensions of power independence, distance between antennas, and antenna diameter.

A. Design Space Cumulative Distribution Function

To begin the analysis process, an initial design space exploration is performed. The eleven design variables considered are antenna diameter (both transmitting and receiving antennas are assumed to be the same diameter), distance between the antennas (assumed constant with time), power independences X and Y (as defined in Equations 2 and 3), payload power requirement (assumed constant during sunlight and eclipse), orbital altitude (for an assumed circular orbit), power transmission frequency, and four efficiencies. The efficiencies include the DC-to-RF conversion efficiency of the transmitter, the RF-to-DC conversion efficiency of the rectenna, and antenna efficiencies for both transmitting antenna and rectenna.

With these variables defined, ranges are assigned in the context of the small satellite problem (see Table 1). Uniform distributions are assigned to each variable in each range and a 10,000-case Monte Carlo simulation is run to assess the feasibility of the design space.[15] The output tracked is the sunlit power requirement of Spacecraft #1 (i.e. the power that must be provided by solar arrays after regulator and converter losses) since it is typically the largest of the power requirements.

Shown in Figure 3 is the resulting cumulative distribution function of this Spacecraft #1 power requirement. While this figure does not tell the whole story of the power beaming problem, it does give a preview of the constrained nature of the design space: Only 6.0% of the designs evaluated are capable of meeting a 250 W notional small-satellite power cutoff (discussed earlier). Even if a generous 500 W constraint is allowed, there is still only a 12.9% feasibility. In other words, over 87% of the designs in the defined design space are well outside of the small satellite class of vehicle.

Table 1. Design Variable Definitions.

Variable	Minimum	Maximum	Baseline
Sunlight Power Independence, *X*	0	0.99	0.9
Eclipse Power Independence, *Y*	0	0.99	0.9
Distance Between Antennas	2 m	2000 m	100 m
Antenna Diameter	0.2 m	2 m	1 m
Payload Power Requirement	5 W	100 W	32 W
Orbit Altitude	200 km	2000 km	700 km
Power Transmission Frequency	2 GHz	100 GHz	35 GHz
Transmitter DC-to-RF Efficiency	0.5	1	0.6
Transmitting Antenna Efficiency	0.75	1	0.85
Receiving Antenna Efficiency	0.75	1	0.85
Rectenna RF-to-DC Efficiency	0.5	1	0.7

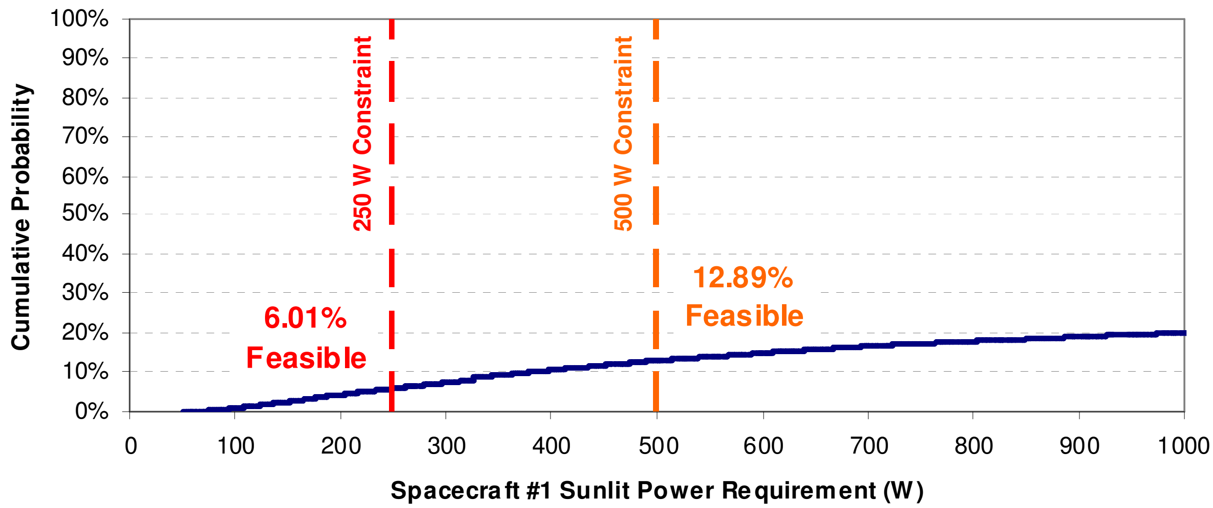

Figure 3. Cumulative Distribution Function of Spacecraft #1 Power within the defined Design Space.

Furthermore, if all designs that exceed a 250 W power requirement are filtered out of the data set, the inputs which correspond to the remaining feasible designs can be plotted as in Figure 4. In each plot of Figure 4, each black dot represents a single feasible design (defined here as a design requiring less than 250 W of power generation for Spacecraft #1). Axes of the plots are inputs to the power beaming model, and areas with high concentrations of black dots indicate regions of the design space of high feasibility (and empty areas of the plots indicate areas of poor feasibility).

Note that only six key inputs are shown Figure 4 (the remaining five inputs, which are less influential, are omitted for clarity). Four of these inputs, spacecraft separation distance, antenna diameter, energy transmission frequency, and payload power requirement, show very strong correlations with feasibility. Weaker but still significant correlations can be seen in terms of the *X* and *Y* power independences. It can be shown that of the 6.01% of the design space that met a 250 W power requirement, 90% of those designs required spacecraft separation distances less than 740 meters, 90% required antenna diameters larger than 0.93 meters, 90% required transmission frequencies above 33 GHz, and 90% required payload power to fall under 68 W. Similarly restricting results are obtained if the filter threshold is set at 500 W (with the exception of payload power, which becomes a more minor factor). Again, this illustrates the highly constrained nature of the power beaming problem.

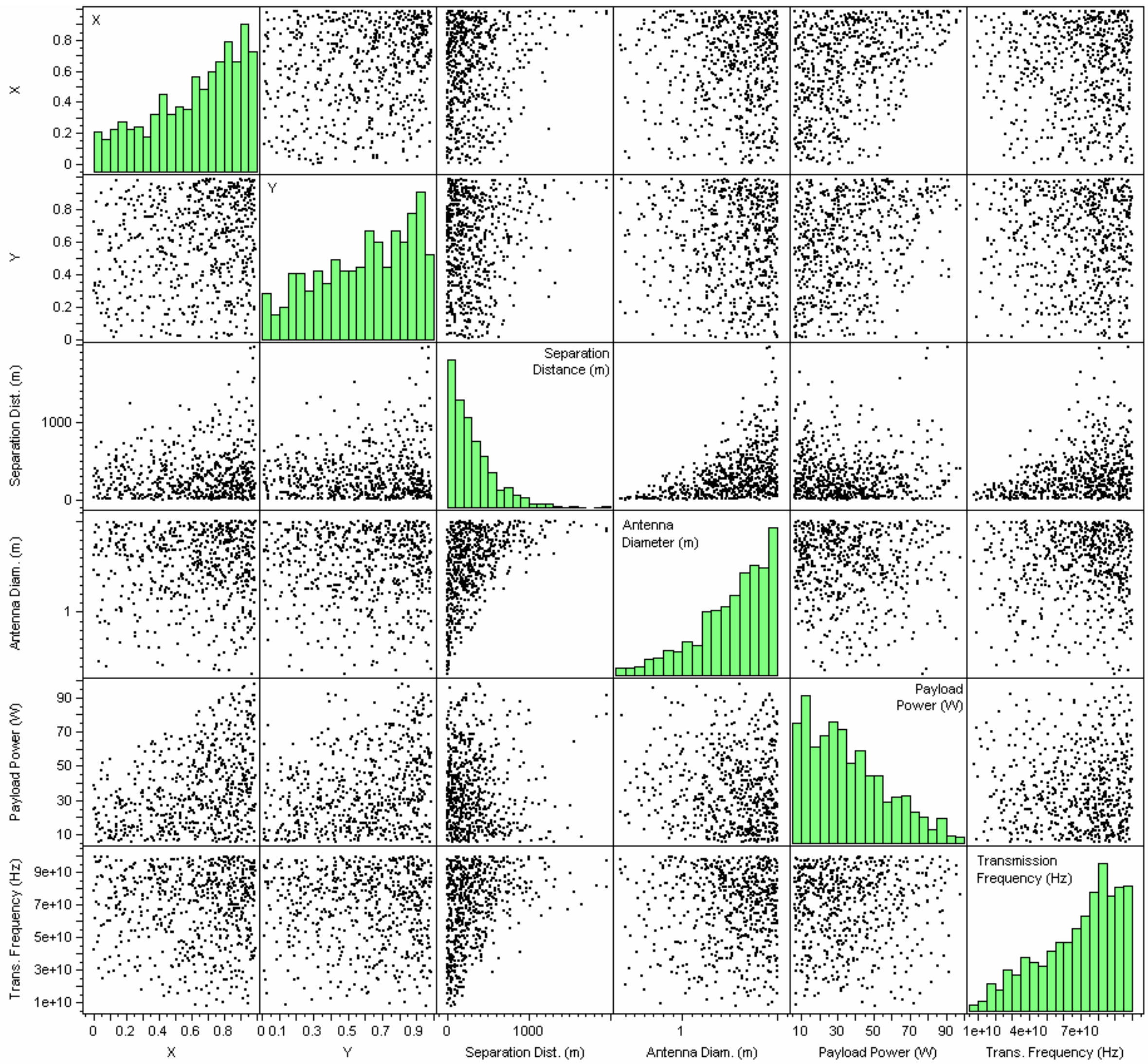

Figure 4. Multivariate plot showing trends among six key input parameters for all 601 feasible (<250 W) designs from the Monte Carlo design space exploration.

B. Baseline Definition

For reference purposes, a baseline configuration in the design space is defined as one reasonable implementation of power beaming. This baseline (see the rightmost column of Table 1) is decided upon after an initial exploration of the design space and uses $X = Y = 0.9$ (i.e., during sunlight and eclipse, the payload-carrying spacecraft generates 90% of its required power and receives the last 10% from beaming). Distance between the antennas of the two spacecraft is set to 100 m and the antennas of the spacecraft are assumed to be 1 m in diameter. The payload power requirement (32 W) and orbit altitude (700 km) are based on a notional remote sensing mission.[11] The power transmission frequency is taken as 35 GHz, an upper limit on the frequency at which reasonably-developed power beaming technology currently exists.[1,5,16] Transmitter efficiency is based on an assumed 35 GHz klystron efficiency,[5] and rectenna efficiency is based on an assumed 35 GHz rectenna.[5] Antenna efficiency is a slightly optimistic estimate based on typical parabolic antenna and rectenna collection efficiencies.[11,17,18]

The power budget for the baseline configuration is shown in Table 2 below. For this baseline, both Spacecraft #1 and #2 have a sunlight power requirement of about 100 W, placing both well within the small satellite category. In this particular scenario, during the sunlit period, Spacecraft #1 must provide 38.0 W to its transmitter in order for 9.7 W to reach Spacecraft #2 in the form of DC power. Similarly, during the eclipse period, Spacecraft #1 must provide 21.1 W to its transmitter in order for 5.4 W to reach Spacecraft #2 as DC power.

Table 2. Power Budget associated with the Baseline Configuration.

Item	Spacecraft #1 Sunlight Power (W)	Spacecraft #1 Eclipse Power (W)	Spacecraft #2 Sunlight Power (W)	Spacecraft #2 Eclipse Power (W)
Payload	0.0	0.0	32.0	32.0
Spacecraft Subsystems				
Propulsion	0.0	0.0	0.0	0.0
Attitude Control	0.0	0.0	0.0	0.0
Communications	15.0	15.0	15.0	15.0
Command & Data Handling	5.0	5.0	5.0	5.0
Thermal	0.0	0.0	0.0	0.0
Power	77.8	22.6	44.9	1.9
Structures & Mechanisms	0.0	0.0	0.0	0.0
TOTAL	**97.8**	**42.6**	**96.9**	**53.9**

C. Sensitivities about a Baseline

To gain a physical understanding of the influence of design variables, the sensitivities of the Spacecraft #1 power requirement are tracked and shown in Figure 5 below. These plots show the variation of the Spacecraft #1 power requirement with respect to one of the eleven variables while the other ten are held constant at the baseline values defined in Table 1[§]. Note that the *y*-axes of all plots are held to the same scale (minimum of 0 W and maximum of 1000 W) and the *x*-axes encompass the entire range of each variable as defined in Table 1.

From Figure 5, three very influential variables are identified: distance between antennas, antenna diameter, and power transmission frequency. All three of these appear to exhibit inverse square relationships with power until an "equilibrium" point is reached at which variations in the parameter no longer have an effect[**].

Three additional variables have a moderate effect: sunlight power independence, eclipse power independence, and payload power requirement. All three have an approximately linear effect on the Spacecraft #1 power requirement (the bend in the eclipse power independence and payload power requirement graph is due to a smoothed discontinuity in the subsystems power model used).

Finally, surprisingly, the four efficiency parameters have a relatively minor effect on the Spacecraft #1 power requirement, at least compared to the other variables. It is also of interest to note that increasing orbit altitude has a slightly favorable effect since a higher altitude correlates with shorter eclipse times.

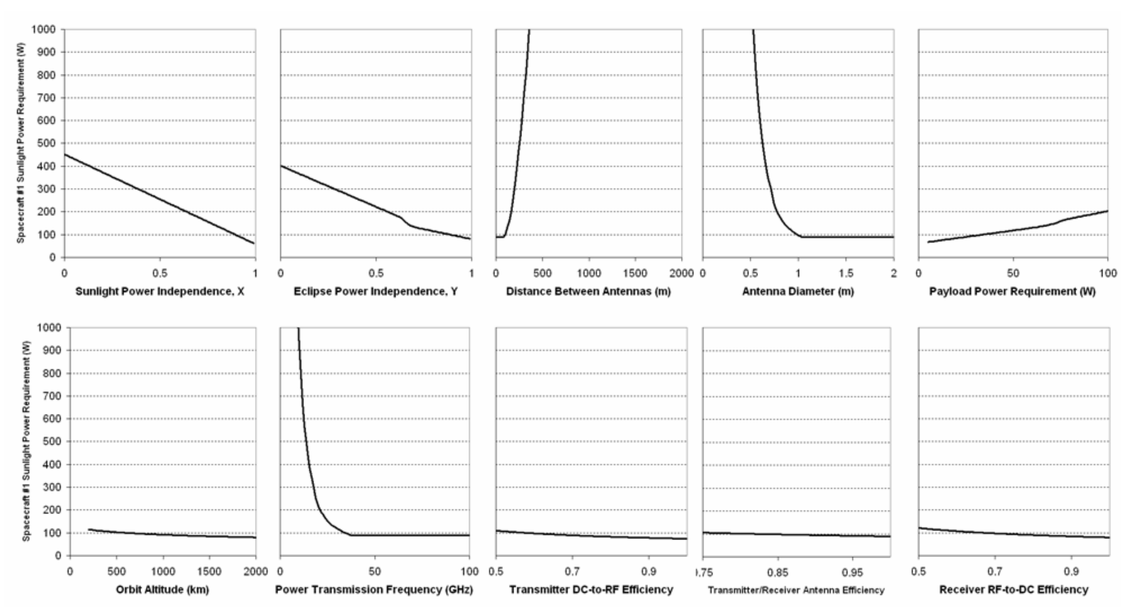

Figure 5. Sensitivities about the defined Baseline.

[§] Note that the transmitter and receiver antenna efficiencies have the same effect on the power requirement output and are shown by a single plot.

[**] This flat-line behavior is due to the fact that the traditional link equation is not valid at very close relative proximities, as noted earlier. Also as noted earlier, the flat-line drawn is actually an optimistic estimate.

D. Parametric Design Space Visualization

In this final step, the design space is visualized by presenting plots in the four dimensions of *X*, *Y*, distance between antennas, and antenna diameter. Two "external" axes are shown as antenna diameter and antenna separation distance. The two "internal" axes of the plots are *Y* versus *X*. The effects of other design space variables are handled by showing the sensitivity of the entire plot with the change of the baseline value.

1. Baseline Design Space

In Figure 6, the payload power requirement, orbit altitude, power transmission frequency, and efficiencies are defaulted to the baseline values in Table 1 while the remaining variables are allowed to vary. The figure clearly shows power requirements increasing with increasing spacecraft separation distance and with decreasing antenna diameter. Plots which are entirely maroon indicate that every combination of *X* and *Y* results in a design with Spacecraft #1 requiring over 500 W in sunlight, well over the bounds for a small satellite (see discussion in Section III.B). Only combinations of very close spacecraft and spacecraft with large antennas generate *X*-*Y* spaces with some feasible region. For a reasonable (1.5 m or less) antenna diameter, 400 m spacecraft separation appears to be the limit of the feasible space. Furthermore, it is noted that feasible regions exist only at *X*-*Y* combinations in the upper right corner of individual plots. For example, in many plots, the $X \geq 0.7$, $Y \geq 0.7$ region is feasible (i.e., it contains no values over 250 W). In other words, even in the most benign cases with close spacecraft and large antennas, power independence of the payload-carrying spacecraft must generally be at least 70%.

2. Sensitivity to Payload Power Requirement

Figure 7 is identical to Figure 6 except that the payload power requirement is set to 60 W instead of the baseline payload power requirement of 32 W. Note that general trends are the same, but far more of the design space lies above 500 W (and thus well outside the limits of a small spacecraft). In this case, blue feasible regions appear at about $X \geq 0.9$, $Y \geq 0.9$, indicating that even in the most benign cases, power independence of the payload-carrying spacecraft must generally be at least 90%.

3. Sensitivity to Power Transmission Frequency

Figure 8 is identical to Figure 6 except that the power transmission frequency is defaulted to 5.8 GHz instead of 35 GHz. This reflects what a potential decision to pursue more traditional power beaming technology would mean for the design space (the 2.5 GHz and 5.8 GHz bands are the ones most prevalent in literature on the topic of power beaming). Note that this change eliminates virtually the entire design space; almost all that remains is a space with antenna diameters larger than 1 m and separation distances less than 100 m. The clear conclusion from this is that the common 5.8 GHz frequency is likely not sufficient to support power beaming in the context of small satellites.

V. Conclusions and Implications

In conclusion, this paper has presented a first-order study of microwave power beaming as applied to small satellites. Several optimistic assumptions were made, including:

- Only two spacecraft exist in the system (the power-transmitting spacecraft must only provide power to one payload-carrying, power-receiving spacecraft).
- No pointing losses are incurred (both spacecraft point directly at each other).
- Power is beamed continuously (no storage penalty is incurred for occasional "power dumps").
- In close-proximity scenarios (i.e. when the traditional link equation gives incorrect received power estimates), it is assumed that there are no space losses (all power transmitted from the antenna of the transmitting spacecraft reaches the antenna of the receiving spacecraft).

However, even with these optimistic assumptions, it has been demonstrated that the small satellite power beaming design space is severely constrained. Only 6% of cases within the defined design space fall under a suggested 250 W small satellite power constraint. The vast majority of cases that are feasible involve very high transmission frequencies (over 33 GHz), large antenna diameters (over 0.93 m), and proximity operations (with spacecraft separation less than 740 m). Payload power requirement is also a driver, and sunlight and eclipse power independences must generally be 70% or greater. Interestingly, power beaming hardware efficiencies are some of the least influential parameters (within the ranges defined by this study), indicating that the technology improvements will have little impact if the system engineer has freedom to vary other design parameters such as transmission frequency, antenna diameter, and spacecraft separation distance.

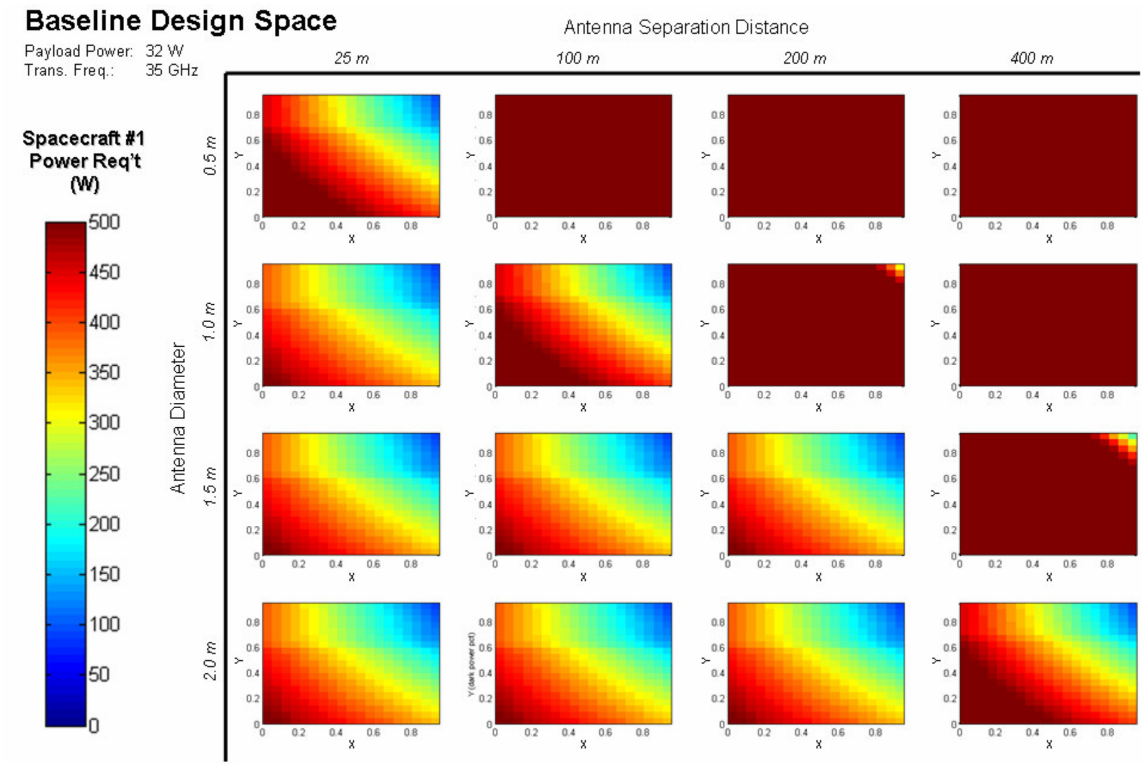

**Figure 6. Baseline Design Space Parametric Visualization.
Note that the color scale has a maximum of 500 W (maroon).**

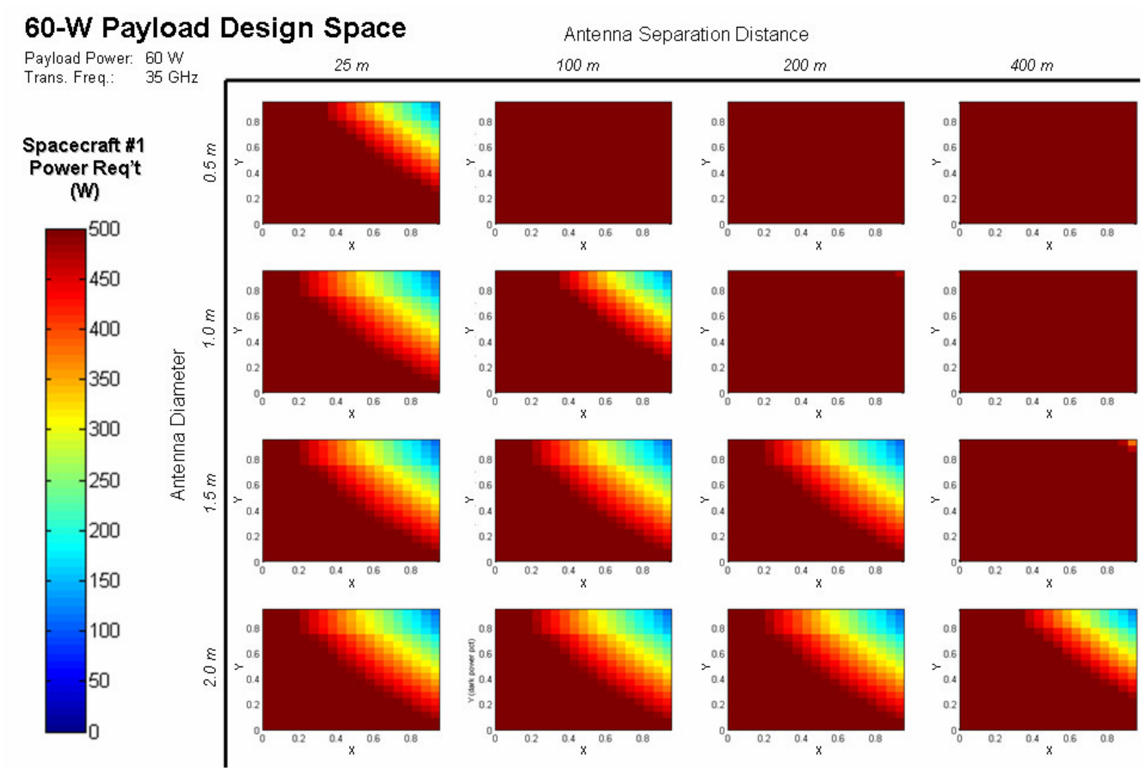

**Figure 7. 60-W Payload Design Space Parametric Visualization.
Note that the color scale has a maximum of 500 W (maroon).**

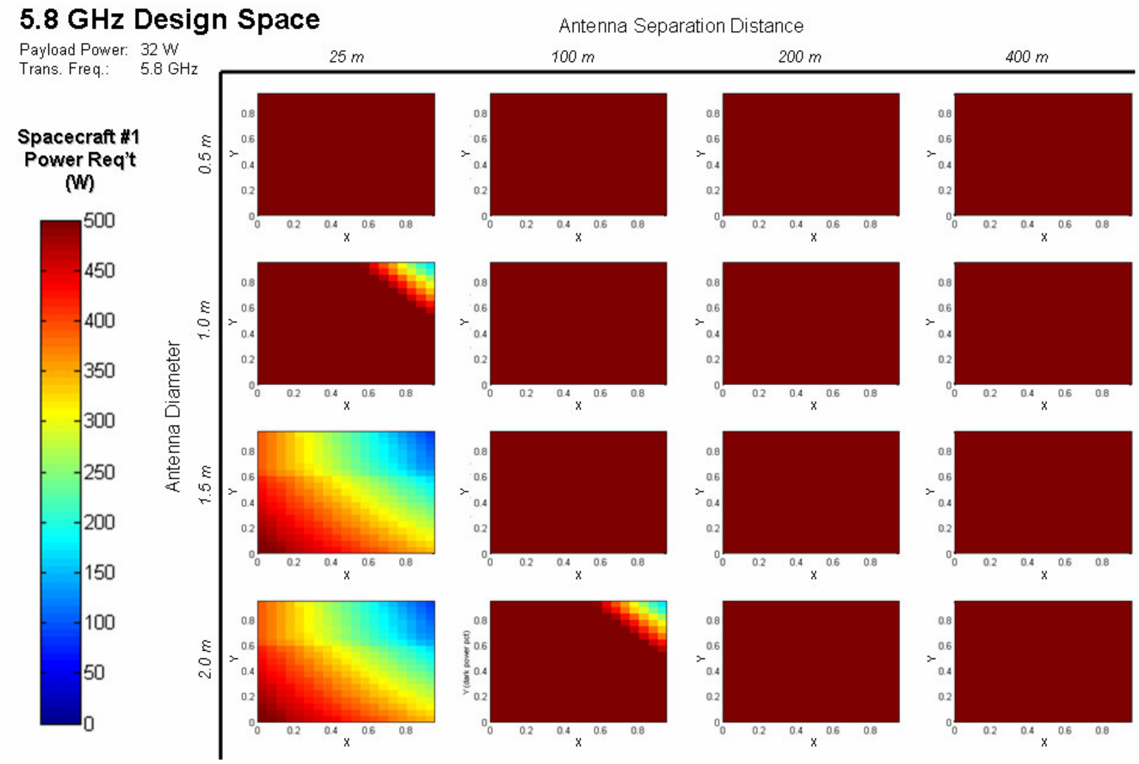

Figure 8. 5.8 GHz Transmission Frequency Design Space Parametric Visualization. Note that the color scale has a maximum of 500 W (maroon).

One important fact to point out is that complete power dependence never appears as a feasible possibility. Small satellites receiving power from other small satellites must have the capability to generate a substantial amount of their own power, using beamed power only as a supplement. Given this fact, it is a fair question to ask whether it would even be worth implementing power beaming for small satellites (since it is clear the spacecraft must carry power-generating equipment anyway, it may not be cost-effective to add additional power beaming equipment to glean the last few percent of the power needed). While this question is beyond the scope of this analysis, it should also be noted that the supplementary power role may be useful in the event of primary power system failure. In this case, a power-transmitting spacecraft could be used to provide keep-alive power to a defunct spacecraft.

Finally, one point to acknowledge is that this study did not consider the implications of using multiple power-transmitting spacecraft. For example, a scenario with two power-transmitting spacecraft and one power-receiving spacecraft was not considered, which would likely have produced more feasible possibilities for the small-satellite problem. However, this approach would require the precise pointing and positioning of three spacecraft and the inclusion of two receiving antennas aboard the payload-carrying spacecraft. The operational complexity introduced by this option would be significant, as would the mass penalty of an additional spacecraft plus an additional receiving antenna on the payload-carrying spacecraft. For these reasons, this option was considered beyond the scope of this study.

Overall, it is reasonable to conclude that microwave power beaming is not currently suitable as a primary mode of power within clusters of small satellites. However, one area of valuable future work may lie in the analysis of the trades required (particularly with respect to mass sizing, costing, and value returned) to allow power beaming to serve as an effective supplementary or even emergency power supply for clusters of small satellites in the future.

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