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System-Level Feasibility Assessment of Microwave Power Beaming for Small Satellites

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Although wireless power transmission to fulfill Earth's energy needs has been a widely popularized application of microwave power beaming, one space application that remains relatively unexplored is power beaming between satellites. This paper provides a system-level analysis of the feasibility and limitations of microwave power beaming within a small-satellite cluster. This analysis consists of four parts using parametric models of spacecraft power as a function of 11 key design variables. In the first part, the existence of feasible designs is verified with a Monte Carlo design trade-space sweep. Next, a feasible baseline (reference) design is defined, and then sensitivity to individual variables is assessed. Finally, the design space is visualized with respect to six influential variables. Despite several optimistic assumptions, it is demonstrated that the small-satellite power-beaming design space is severely constrained. Feasible designs involve high transmission frequencies (greater than 33 GHz), large antenna diameters (greater than 0.93 m), and stringent proximity operations between satellites (within 740 m). Furthermore, full dependence of one small spacecraft on power provided by another is shown to be effectively infeasible. These results do suggest, however, that microwave power beaming may deserve consideration as an auxiliary or short-term emergency power mode for future small-satellite clusters.

Nomenclature

G_r	=	gain of receiving antenna (rectenna)
G_t	=	gain of transmitting antenna
L_{ra}	=	rectenna collection efficiency
L_{rl}	=	rectenna RF-to-dc efficiency
L_s	=	space loss
L_{ta}	=	transmitting antenna efficiency
L_{tl}	=	transmitter dc-to-RF efficiency
m	=	spacecraft wet mass, kg
P	=	spacecraft power requirement, W
$P_{\mathrm{dc,rec}}$	=	usable power output of spacecraft no. 2 rectenna
$P_{\mathrm{dc,trans}}$	=	power input to spacecraft no. 1 transmitter
$P_{\mathrm{SC1,sun}}$	=	sunlit power requirement of spacecraft no. 1
X	=	sunlit period power independence
Y	=	eclipse period power independence
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 solar energy transmission from space to fulfill Earth's power needs. Although studies have often found that this application is likely infeasible with current technology (as well as economically unviable and possibly environmentally hazardous), a space application that has remained relatively unexplored is microwave power beaming between small satellites. In particular, the Defense Advanced Research Projects Agency (DARPA) has recently expressed interest in such intersatellite power beaming as a potential enabling technology for the concept of a fractionated spacecraft [1–4].

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

II. Highlights from the History of Microwave Power Beaming

In the 1960s and 1970s, the space community took notice of microwave power beaming in its potential use for transmitting massive amounts of solar power (measured at the gigawatt level) to Earth from space [5]. Although much too ambitious for the space capabilities of the 1960s, the concept of the solar power satellite [5] added impetus to several development efforts of the 1960s and 1970s. Raytheon, for example, 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 Jet Propulsion Laboratory's Goldstone Deep Space Network facility at the level of 30 kW [5,6]. In 1976, a milestone was reached when a 91% rectenna absorbed-to-dc-power conversion efficiency was demonstrated for low power levels (of the order of 10 W) [5].

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 used 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 [5,7,8].

In 1997, NASA conducted a Fresh Look study to reassess the solar power satellite concept. Although the study concluded that solar power satellite systems were more technically feasible than they were a few decades earlier, it also showed that their economic

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viability remains dubious [9]. 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 [10,11]. Most recently, in October 2007 the National Security Space Office (NSSO) released a study on space-based solar power that strongly recommended that the U.S. Government invest in an in-space proof-of-concept demonstration within the next decade. One potential use of megawatt-scale power beaming was recognized by the NSSO to be the supply of energy to U.S. forward bases, which typically rely on expensive (and dangerous) convoy resupply [12].

As a final note, in 2007 DARPA called for proposals to develop the F6 fractionated spacecraft 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-orbiting space systems. In 2006, Brown and Eremenko [1] described fractionation as the decomposition of a space system into modules that interact wirelessly to deliver the capability of a functionally equivalent monolithic system. Brown 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. System Modeling

A. Assumptions and Power Paths

The following analysis is intended as a system-level feasibility assessment. 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 perfect line of sight with the capability of continuous microwave power transfer and no pointing losses. The role of the first spacecraft is to carry the power transmission equipment, and the role of the second is to carry the payload (and power-receiving equipment).

The assumed power pathways are shown graphically in Fig. 1. Power is generated by solar arrays on spacecraft no. 1, and that power is regulated to the proper dc voltage and current level. Enough power is generated to power the internal subsystem loads of spacecraft no. 1 as well as charge its battery. Additionally, part of spacecraft no. 1's power is transmitted at microwave frequencies to spacecraft no. 2, in which it is received (after space losses) by a rectenna. Power from the rectenna is added to power generated independently by spacecraft no. 2 to power internal subsystem loads plus those of the payload. Internal spacecraft loads are determined based on typical satellite power-consumption estimation relationships [13] and are assumed to be constant over time. A consequence of this assumption is that only the power budget of the power subsystem differs between sunlight and eclipse periods. Additionally, it is worth noting that for very low total power requirements (less than 100 W), the power model from [13] allocates no power for attitude determination and control. This conflicts with the assumption of no pointing losses and results in the optimistic assumption that perfect pointing can be achieved without providing power for attitude control functions.

Power losses are considered throughout the power distribution chain, particularly in the power-beaming hardware. Losses occur through the transmitter (e.g., a magnetron or other microwave-generating device), the antenna (due to dish imperfections and feed array interference), and the 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.

The white and gray circles in Fig. 1 indicate the points at which sunlit and eclipse power requirements, respectively, are tracked in this study. The placement of the white circles indicates that sunlit power requirements quoted in this paper refer to useful dc power after regulation (i.e., unregulated power from solar arrays will need to be higher than the post-regulation powers quoted here). The placement of the gray circles indicates that eclipse power requirements quoted in this paper refer to the useful dc power output of the battery.

A key requirement in this modeling is specification of, in the relative sense, how much power beaming must occur. For example, at one end of the modeling spectrum is a situation in which spacecraft no. 2 receives its entire power requirement via power beaming (i.e., through spacecraft no. 1). At the other end, spacecraft no. 2 may provide all of its own power through its own solar arrays (and, during eclipse, its battery). A continuum of possibilities exists between these extremes, which may be considered as a degree of power fractionation that varies between 0 and 100%. To account for this, a parameter Z named power independence is introduced and is defined in Eq. (1). Z can have values ranging from zero to unity. As defined in Eq. (1), when Z is zero, the payload-carrying spacecraft (spacecraft no. 2 in Fig. 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

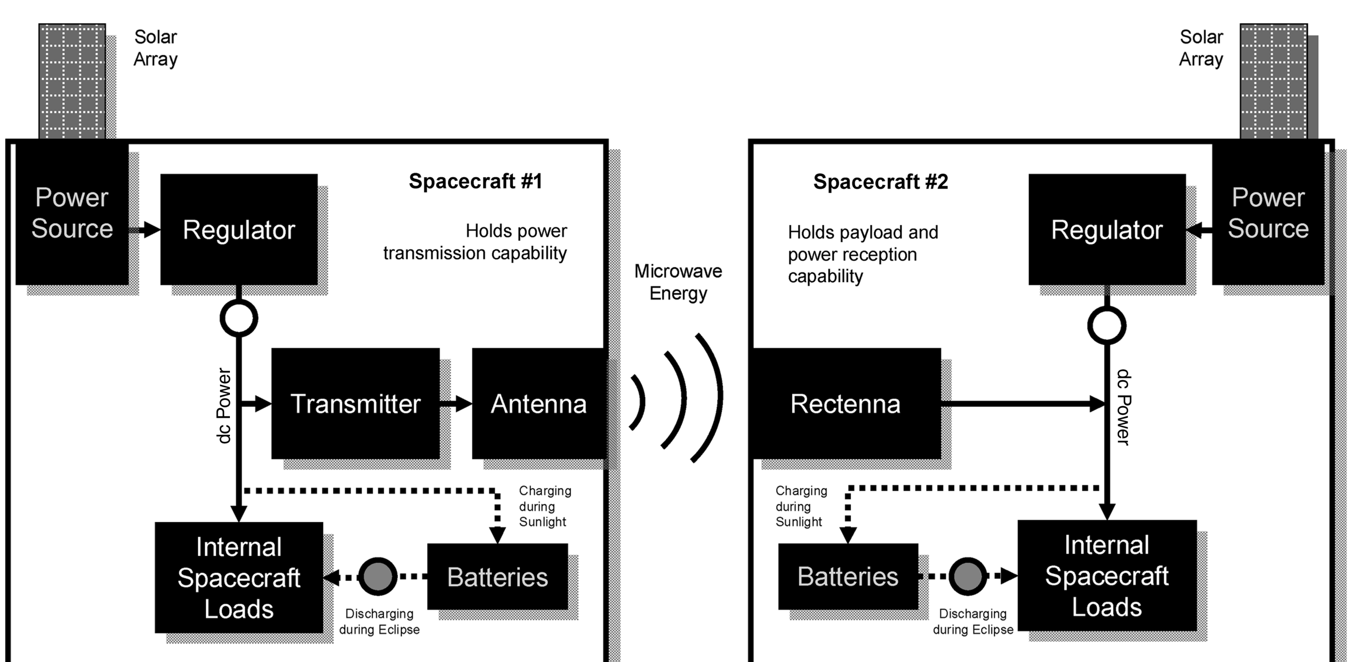

Fig. 1 Power path model.

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 it is assumed that solar arrays and batteries do not provide power simultaneously):

$$Z = \frac{\text{power generated by spacecraft 2}}{\text{total spacecraft 2 power requirement}} \tag{1}$$

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 Eqs. (2) and (3)]:

$$X = Z|_{\text{sunlit}} = \frac{\text{power generated by spacecraft 2 during sunlit period}}{\text{total spacecraft 2 power requirement during sunlit period}} \tag{2}$$

$$Y = Z|_{\text{eclipse}} = \frac{\text{power generated by spacecraft 2 during eclipse}}{\text{total spacecraft 2 power requirement during eclipse}} \tag{3}$$

One final assumption of note is that space losses are assessed via a variant of the traditional link equation shown in Eq. (4) (see the Nomenclature for a 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 in which this is an issue, the model used in this study optimistically assumes that there are no space losses:

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

B. Design Variables

In the system defined previously, as in any engineering system, a number of key variables can be controlled by the engineer designing the system. For example, the modeling assumptions previously have not restricted the engineer's choice of the payload, the distance between the two spacecraft, the frequency at which power is transmitted, or any of several other parameters. In the terminology of this study, the numerical variables under the control of the engineer designing the system are called design variables.

In this study, 11 design variables are considered, and they may be divided into the categories of physical design variables and technology design variables. The physical design variables, which deal with decisions that the engineer can directly make involving physical components and mission design, 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 Eqs. (2) and (3)), payload power requirement (assumed constant during sunlight and eclipse), orbital altitude (for an assumed circular orbit), and power transmission frequency. The technology design variables used here, which are efficiencies reflecting the amount of investment an engineer might decide to place on further developing a given component, are the dc-to-RF (direct-current-to-radio-frequency) conversion efficiency of the transmitter, the RF-to-dc conversion efficiency of the rectenna, and antenna efficiencies for both transmitting antenna and rectenna. Although it is a simple matter to multiplicatively combine these four efficiencies into a single net efficiency, they remain separated in this study to provide physically meaningful information at the component level, particularly because these individual efficiencies are not identical and may require different levels of time or effort to attain a given level of improvement.

C. Selection of Output Metric

The models in this study take as inputs the design variables described previously and produce several metrics that can be defined as outputs. The most important of these metrics are the power requirements during sunlit and eclipse periods for each spacecraft (referenced at the white and gray circles in Fig. 1, as described in Sec. III.A). Of these four metrics, the output tracked throughout this study as the feasibility criterion is the sunlit power requirement of spacecraft no. 1, denoted as $P_{\text{SC1,sun}}$. During sunlit periods, spacecraft no. 1 must generate enough power through its solar arrays to power its own internal loads, charge its own battery, and power a portion of the loads on spacecraft no. 2 through beaming (which includes a portion of charging the spacecraft no. 2 battery). As a result, the sunlit power requirement of spacecraft no. 1 ($P_{\text{SC1,sun}}$) is typically the largest of the power requirements and, consequently, the power requirement for which the larger of the spacecraft must be sized.

D. Historical Spacecraft Power/Mass Correlation

Because this work 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 subsystem mass estimation relationships to accomplish this, the present study uses a curve-fit relating wet mass and power for previously flown Earth-orbiting satellites.

Power and mass data are compiled from 28 previous satellites [13–16] and result in the curve fit shown in Fig. 2 and Eq. (5). The coefficient of determination R^2 for the curve fit is 0.739 and predicts a power of 250 W for a 300 kg spacecraft. When necessary, this study uses the 250 W estimate as a suitable upper bound for the power requirement of a small satellite. Note, however, that although the following analysis will often quote metrics associated with the 250 W bound, many of the plots can be used to show the consequence of selecting higher or lower bounds. For example, the last two figures in this paper display data through 1000 W. As a result, much of the data and results generated in the present study may still be useful if individual engineers prefer to apply different upper bounds on small-satellite power requirements:

$$m = 2.32 P^{0.881} \tag{5}$$

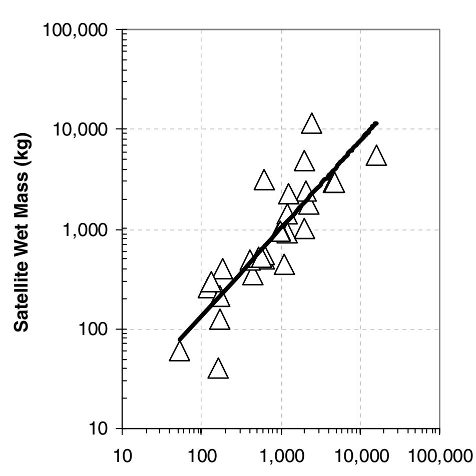

Fig. 2 Historical spacecraft power/mass data (including curve fit).

IV. Results and Discussion

The models defined previously ultimately allow an engineer to select values for the 11 design variables and track the sunlit power requirement of the power-beaming spacecraft (spacecraft no. 1), which can be correlated to typical satellite wet masses through the relationship in Eq. (5). Thus, a set of selections for each of the 11 design variables allows analysis of a single point design. More important, by allowing each of the design variables to vary, these models allow a parametric trade-space evaluation for a wide range of possible designs. This evaluation is documented in the following sections and 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 11-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. Each of these steps is described in detail in the sections that follow.

A. Design Space Cumulative Distribution Function

To begin the analysis process, an initial design trade-space exploration is performed. Ranges are assigned to each of the 11 design variables 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 [17]. For example, for any given case in the Monte Carlo simulation, the selection of distance between antennas is allowed to be anywhere between 2 and 2000 m, with no bias toward any particular value within that range. The bounds on the distribution are selected as approximate limits to the small-satellite problem of interest (it is not expected that spacecraft of interest would be less than 2 m apart or greater than 2 km apart). As described in Sec. III.C, the output tracked as the feasibility criterion is the sunlit power requirement of spacecraft no. 1 ($P_{\mathrm{SC1,sun}}$).

Shown in Fig. 3 is the resulting cumulative distribution function of $P_{\mathrm{SC1,sun}}$. Although this figure does not reveal which variables are design drivers, it does give a preview of the constrained nature of the design space: only 6.0% of the 10,000 cases 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. More formally, because in this step the design variables are assigned as random variables, the resulting output $P_{\mathrm{SC1,sun}}$ is a random variable here. Thus, this allows the formal statements in Eq. (6); that is, if an engineer were to select design variables from the trade space with no preferences, the probability of selecting a design meeting 250 and 500 W $P_{\mathrm{SC1,sun}}$ requirements is 6.0 and 12.9%, respectively. In other words, over 87% of the designs in the defined design space fall well outside of the small-satellite class of vehicle:

$$\mathrm{pr}\,(P_{\mathrm{SC1,sun}} < 250\ \mathrm{W}) = 0.060 \qquad \mathrm{pr}(P_{\mathrm{SC1,sun}} < 500\ \mathrm{W}) = 0.129 \tag{6}$$

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	0.50	1	0.60
Transmitting antenna efficiency	0.75	1	0.85
Receiving antenna efficiency	0.75	1	0.85
Rectenna RF-to-dc efficiency	0.50	1	0.70

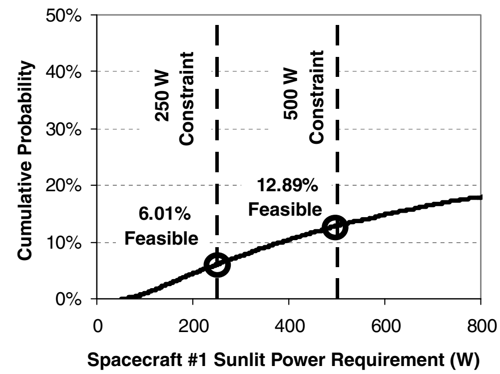

Fig. 3 Segment of $P_{\mathrm{SC1,sun}}$ cumulative distribution function.

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 Fig. 4. In each offdiagonal plot of Fig. 4, each black dot represents a single feasible design (defined here as a design with $P_{\mathrm{SC1,sun}} < 250$ W). Axes of these 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). For example, the plot of antenna separation distance versus antenna diameter (see row 4, column 3 in Fig. 4) shows strong clustering toward low values of separation distance and high values of antenna diameter, indicating that all feasible designs (i.e., with less than 250 W power requirements) had combinations of large antennas and/or close-proximity spacecraft. Because similar plots on the diagonal of Fig. 4 would be trivial (these would show plots of one variable versus itself), on the diagonal are histograms approximating the likelihood that the value of a single variable will fall within a given interval for the filtered (feasible) set of designs. If a variable has no effect on feasibility, its histogram should show an approximately uniform distribution because the input distribution was uniform.

Note that only six key inputs are shown Fig. 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 m, 90% required antenna diameters larger than 0.93 m, 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 for small satellites.

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 [13]. The power transmission frequency is set at 35 GHz, an upper limit on the frequency at which reasonably developed power-beaming technology currently exists [4–6,10,18]. Transmitter efficiency is based on an assumed 35 GHz klystron efficiency [6], and rectenna efficiency is based on an assumed 35 GHz rectenna [4,6]. Antenna efficiency is a slightly optimistic

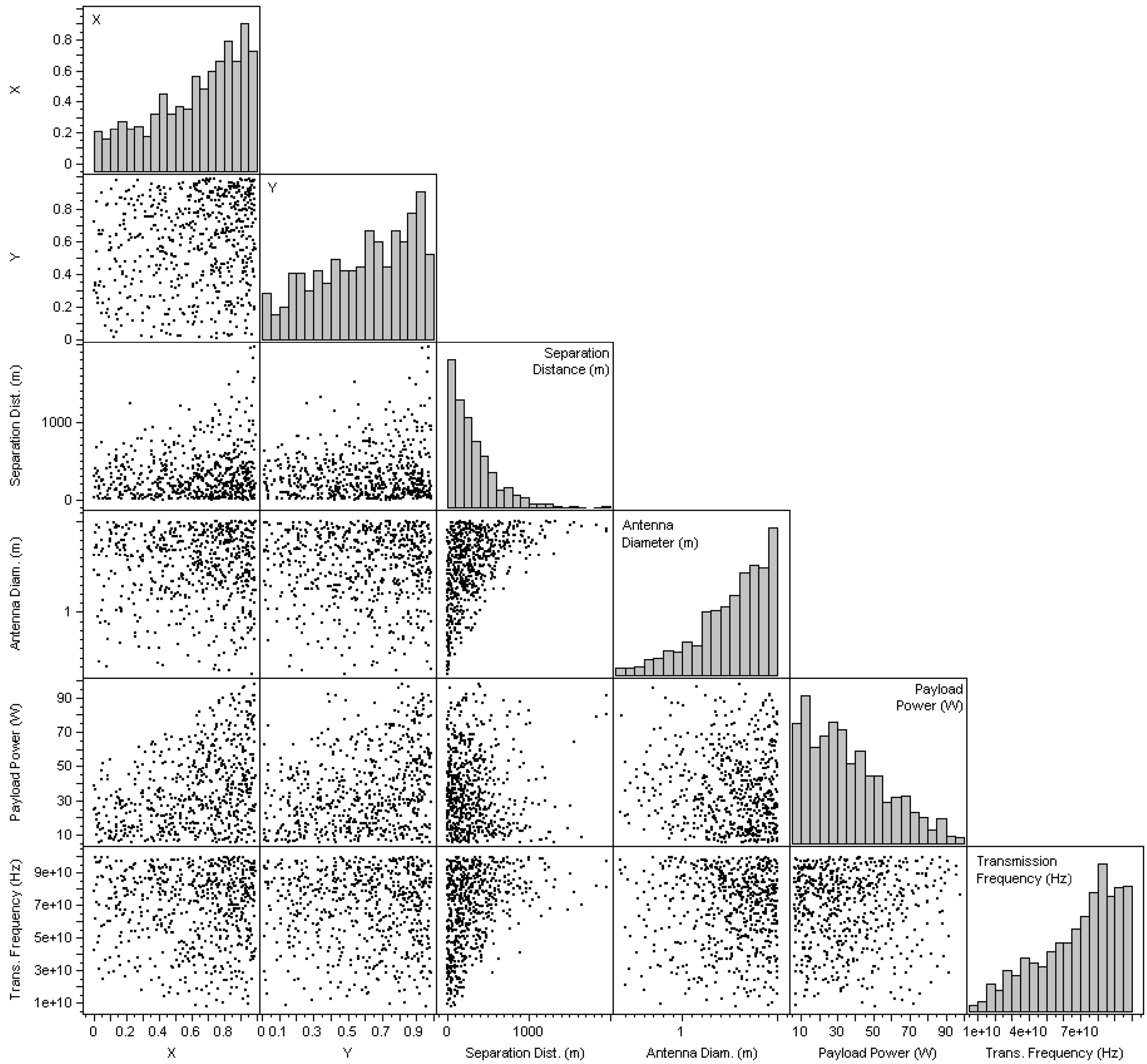

Fig. 4 Multivariate plot showing trends among six key input parameters for all 601 feasible (less than 25 W) designs from the Monte Carlo design space exploration.

estimate based on typical parabolic antenna and rectenna collection efficiencies [13,19].

The power budget for the baseline configuration is shown in Table 2. For this baseline, both spacecraft no. 1 and 2 have a sunlit power requirement of about 100 W, placing both well within the small-satellite category. In this particular scenario, during the sunlit period, spacecraft no. 1 must provide 38.0 W to its transmitter for 9.7 W to reach spacecraft no. 2 in the form of dc power. Similarly, during the eclipse period, spacecraft no. 1 must provide 21.1 W to its transmitter for 5.4 to reach spacecraft no. 2 as dc power.

C. Sensitivities About a Baseline

To gain a physical understanding of the influence of design variables, the sensitivities of the spacecraft no. 1 power requirement $P_{SC1,sun}$ are analyzed and shown in Fig. 5. Each plot shows the

Table 2 Power budget associated with the baseline configuration

	Spacecraft no. 1		Spacecraft no. 2	
Item	Sunlit power, W	Eclipse power, W	Sunlit power, W	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 and 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 and mechanisms	0.0	0.0	0.0	0.0
Total	97.8	42.6	96.9	53.9

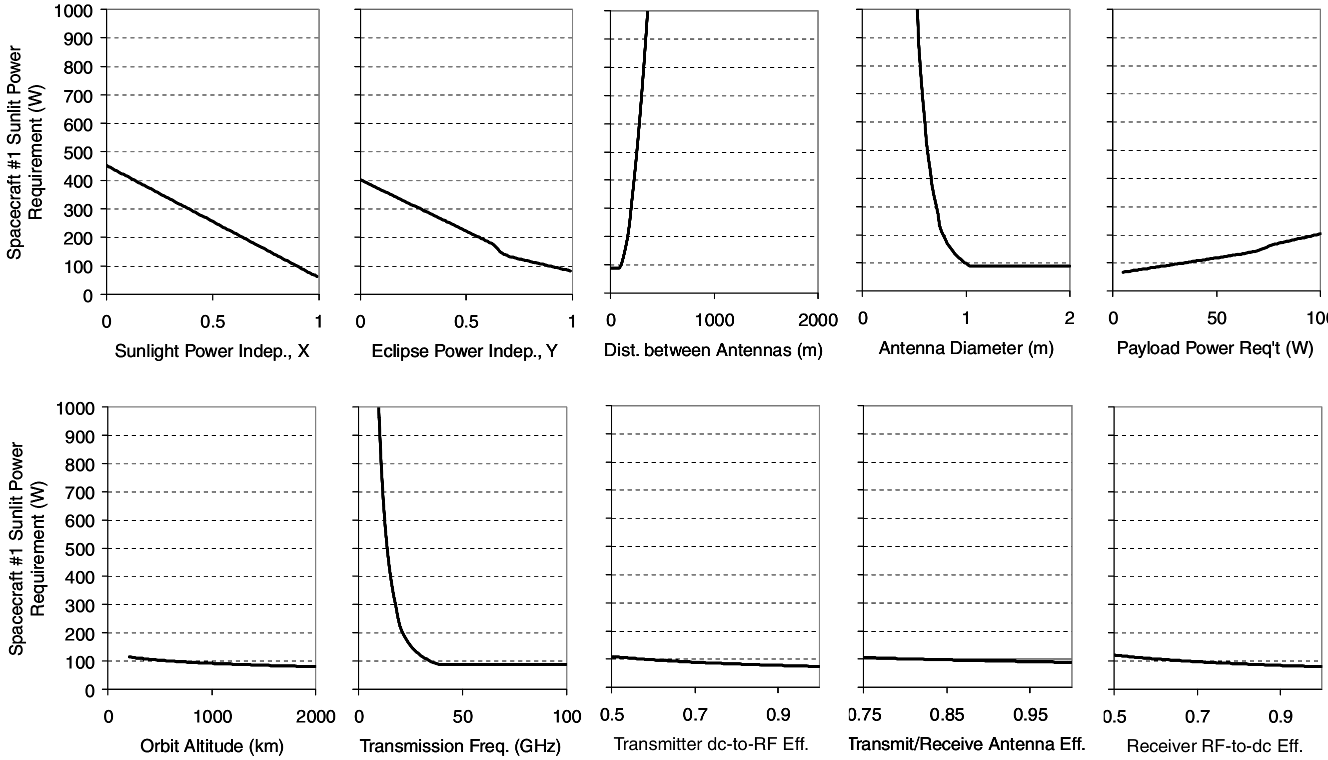

Fig. 5 Sensitivities of $P_{SC1,sun}$ about the defined baseline.

variation of $P_{SC1,sun}$ with respect to one of the 11 variables, whereas the other 10 are held constant at the baseline values defined in Table 1 (only 10 plots are shown in Fig. 5 because the transmitter and receiver antenna efficiencies have the same effect on the power requirement and are shown by a single plot). 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. For example, the upper left plot shows the variation in $P_{SC1,sun}$ with changes in the sunlight power independence X, where X varies from 0 to 0.99. At $X = 0$ (i.e., the situation in which spacecraft no. 2 receives all of its power through beaming during sunlit periods), the power requirement is about 450 W, and at $X = 0.99$ (i.e., in sunlit periods, spacecraft no. 2 generates 99% of its own power), the power requirement is about 60 W. Thus, with respect to the baseline design in Table 1, changes in X from 0 to 0.99 can produce changes in power requirement from 60 to 450 W. At this point, it should become clear that Fig. 5 can assist in the identification of influential variables; if the difference in power requirement between the minimum and maximum values for a given variable is large, then that variable is more influential than one for which the difference is small (again, this is with respect to the baseline design in Table 1).

Thus, from Fig. 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 a critical point is reached at which variations in the parameter no longer have an effect. This flat-line behavior occurs due to the optimistic estimate noted earlier regarding zero space losses at very close proximities.

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 no. 1 power requirement (the bend in the eclipse power independence and payload power requirement graph is due to a smoothed discontinuity in the power model from [13]).

Surprisingly, the four efficiency parameters have a relatively minor effect on the spacecraft no. 1 power requirement, at least compared with the other variables. This suggests that 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. Finally, at a level of influence similar to the efficiency parameters is orbit altitude, which produces a slightly favorable effect when increased because a higher altitude correlates with shorter eclipse times.

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 may be handled by showing the sensitivity of the entire plot with the change of the baseline value. Contours represent the sunlit power requirement of spacecraft no. 1 ($P_{SC1,sun}$) and are shown at every 100 W from 100 to 1000 W. Areas of the design space with power requirements greater than 1000 W are marked as such.

1. *Baseline Design Space*

In Fig. 6, the payload power requirement, orbit altitude, power transmission frequency, and efficiencies are defaulted to the baseline values in Table 1, whereas the remaining variables are allowed to vary. The figure clearly shows the sunlit power requirement of spacecraft no. 1 ($P_{SC1,sun}$) increasing with increasing spacecraft separation distance and with decreasing antenna diameter. Plots with no contours indicate that every combination of X and Y results in a design with spacecraft no. 1 requiring over 1000 W in sunlight, well over the bounds for a small satellite (see the discussion in Sec. III.D). Only combinations of close-proximity 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, feasible regions exist only at X–Y combinations in the upper right corner of individual plots. In many plots, only the $X \geq 0.6$, $Y \geq 0.6$ region is fully feasible. That is, this rectangular region contains no values over 250 W; equivalently, the region $X < 0.6$, $Y < 0.6$ is generally fully infeasible, and the other two quadrants ($X \geq 0.6$, $Y < 0.6$ and $X < 0.6$, $Y \geq 0.6$) are partially feasible. For example, in many plots, $X = 0.5$, $Y = 0.8$ is a solution

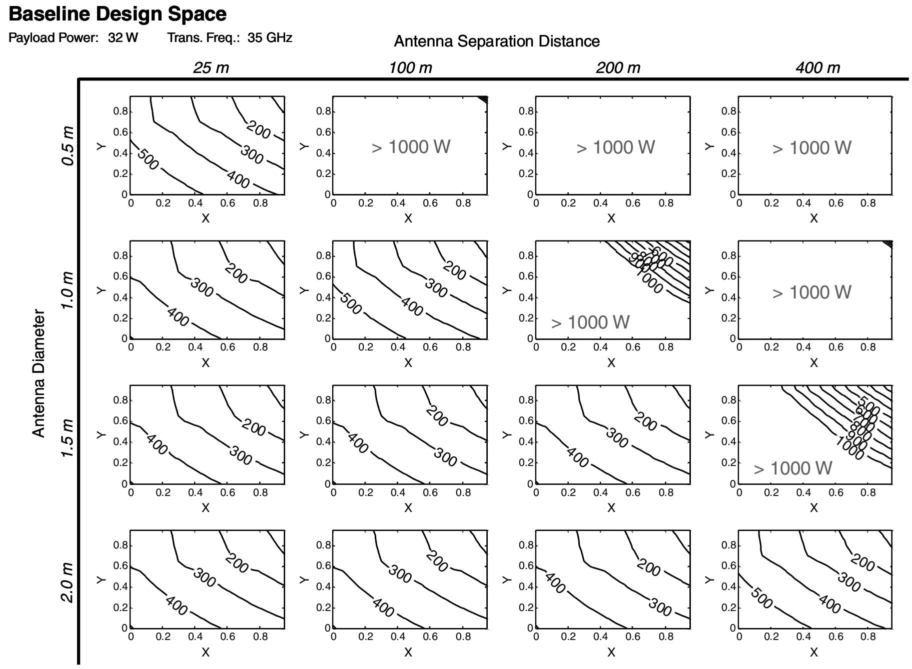

Fig. 6 Baseline design space with contours of spacecraft no. 1 sunlit power requirement ($P_{\mathrm{SC1,sun}}$).

that requires less than 250 W of power; however, for one power independence X to be less than 0.6, the other power independence Y must be greater than 0.6. In short, even in the most benign cases with close-proximity spacecraft and large antennas, at least one power independence (X or Y) of the payload-carrying spacecraft must generally be at least 60%. This definitively excludes full power dependence (i.e., $X = Y = 0$) and further implies that at some point during an orbit, the payload-carrying spacecraft must supply at least 60% of its own power.

2. *Sensitivities to Payload Power and Transmission Frequency*

It is important to recognize that Fig. 6 is only valid for the default selections of payload power requirement, orbit altitude, power transmission frequency, and efficiencies mentioned earlier. If the payload power requirement is set to 60 W instead of the baseline payload power requirement of 32 W, general trends remain the same, but far more of the design space lies above 1000 W and thus well outside the limits of a small spacecraft. In this case, feasible regions appear at about $X \geq 0.75$ and $Y \geq 0.75$, indicating that even in the most benign cases, power independence of the payload-carrying spacecraft must generally be at least 75%. As might be expected, the higher the payload requirement, the more constraints exist on other parameters in the design trade-space (power independences, antenna diameter, separation distance, etc.) if all spacecraft must remain small.

Also of interest is the scenario in which 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 and 5.8 GHz bands are the ones most prevalent in literature on the topic of power beaming). This change eliminates virtually the entire design space; almost all that remains are designs with antenna diameters larger than 1 m and separation distances less than 100 m. The principal 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

This paper has presented a system-level feasibility study of microwave power beaming as applied to small satellites. Several optimistic assumptions were made, including the following:

1) Only two spacecraft exist in the system (the power-transmitting spacecraft must only provide power to one payload-carrying, power-receiving spacecraft).

2) No pointing losses are incurred, and no power is provided to an attitude control system to ensure this (both spacecraft point directly at each other through passive control).[‡]

3) Power is beamed continuously (no storage penalty is incurred for occasional power dumps).

4) 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).

Despite 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 60% or greater. Interestingly, power-beaming hardware efficiencies are some of the least influential parameters (within the ranges defined by this study), indicating that 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.

One important fact to point out is that complete power dependence (i.e., $X = Y = 0$) 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 be worth implementing power beaming for small satellites (because it is clear that 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). Although this question is

[‡]In reality, pointing and tracking is a complex design problem in and of its own. In the simple two-spacecraft system presented here, solar arrays must be sun-pointing, whereas the antenna and rectenna must be precisely aligned. Additionally, the payload may have its own pointing requirements.

beyond the scope of this analysis, it should also be noted that an auxiliary power role may be useful in the event of primary power system failure. In this case, a power-transmitting spacecraft could, for example, be used to provide keep-alive power to a defunct spacecraft.

It should also be emphasized that the 250 W maximum power requirement applied to spacecraft no. 1 in this study applies only if both spacecraft are required to be *small* satellites. This study does not necessarily preclude solutions of large power-beaming spacecraft providing power to small power-receiving spacecraft, and the same methods used here (and much of the same data generated) may also be useful in assessing these concepts in the future.

One final 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 not explored in 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, cost, and value returned) to allow power beaming to serve as an effective supplementary or emergency power supply for clusters of small satellites.

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