# A Low-Cost Wireless Power Transmission Experiment

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## Abstract

A solar power satellite (SPS) is a satellite dedicated to collecting solar energy on orbit, transforming it into microwave or laser energy, and beaming it to a receiving station on the ground. The transmitted energy is converted into DC or AC power for consumer use. Even a test version of the SPS would be a challenging and expensive undertaking due to the amounts of energy which need to be collected. Satellites measured in hundreds of meters and hundreds of millions of dollars are required for an accurate demonstration. It has been suggested that useful measurements can be made for substantially less if the beaming direction is reversed. A 30 kg, cubic satellite with 31 cm square faces costing under two million dollars is presented. The spacecraft is to receive, rectify and measure microwave power beamed from a high-power, ground-based radar station. Secondary payload opportunities have been analyzed in conjunction with the applicability, availability, and cost of government radar stations. Primary and back-up radar stations have been chosen. Orbital constraints recommended the selection of respective primary and back-up launch vehicles. The satellite is being designed to survive the launch environments of both launchers to increase the likelihood of mission success.

## Introduction

In 1968 Peter Glaser introduced the concept of a solar power satellite (SPS)[1,2]. An SPS is an application of power beaming or wireless power transmission (WPT). Solar energy is collected on orbit, converted into microwave or laser energy, and beamed to a receiver on the ground. The receiver converts the captured energy into useful AC or DC power. In 1977 NASA and the DOE initiated the Concept Development and Evaluation Program (CDEP). In order to advance the understanding of the technical, social, safety, and economic issues associated with an SPS, a "reference system" was developed. While the reference system was never intended to represent all possible forms of an SPS, its scale provides a data point against which the magnitude of the present design's results can be compared. Table 1 gives the reference system characteristics.

Table 1: Reference System Characteristics (Single Satellite)

| | |
|---|---|
| Power Delivered to Ground | 5 GW |
| Overall Dimensions | 10 x 5 x 0.5 km |
| Mass | 35-50 * $10^6$ kg |
| Orbit | GEO |
| Transmission Antenna Diameter | 1 km |
| Transmission Frequency | 2.45 GHz |

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The National Research Council (NRC) evaluated the expansive work which came out of CDEP. In its report[3], the NRC concluded

> "High capital cost per unit of installed power capability is not the only deterrent to SPS deployment. The complexity of the system, its "all or nothing" character, and the necessity of investing on the order of $100 billion [1980$] before the first unit can begin operation would all strain our ability to introduce an SPS..."

Post-CDEP studies have refined the reference system, introduced alternate architectures and developing technologies, and attempted to define a system which could be implemented gradually, be less costly, and provide an early return on investment.

Studies by Keith Rogers[4] and the University of Alaska at Fairbanks[5] (UAF) in 1993, introduced the idea of reversing the beaming direction. Figure 1 presents the concept by Rogers. A ground-based radar station is used to beam microwave power to an orbiting microsatellite. The satellite is launched as secondary payload (SPL) aboard an Ariane IV, and consists of a cubic structure attached to a 10-meter inflatable reflector which would focus the microwaves onto a receiver on the cube. Arecibo was chosen as the radar station because of its capability of delivering high, continuous-wave power at 2.38 GHz to low Earth orbit (LEO). A system of control cables connect the reflector and the cube keeping the microwaves targeted on the receiver. Costs were estimated to be between eight and 10 million dollars for a five-year program.

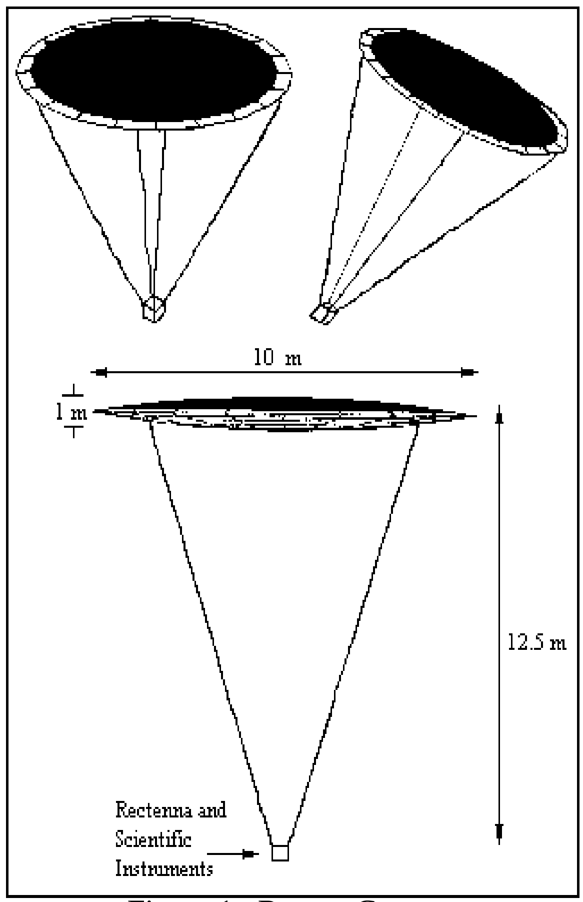



Figure 1: Rogers Concept

UAF included both microwave and laser beaming in its proposed wireless power transmission (WPT) experiment. A NOAA satellite tracking station at Fairbanks provided microwave power at 35 GHz. A 14-meter, inflatable reflector captured and focused the microwaves. Program costs through one year of operation were estimated at 48 million dollars including a dedicated Pegasus launch.

The mission described herein will collect microwave energy from a ground-based radar station, capture it with a rectifying antenna, or "rectenna," convert the energy to DC, and measure the amount of energy converted. Mission requirements have been developed. Mission and system analysis has been completed, and two mission architectures have been downselected. The satellite configuration has been defined, and subsystem design drivers have been identified.

## Mission Architecture Development

Mission requirements were derived to meet WPT data needs. They were developed from the above-mentioned studies, from SPS literature, and from surveying investigators in the field of radio wave transmissions through the atmosphere. The requirements were assessed from a cost-constraint perspective. Engineering solutions and science needs were retained and enabled only when they did not violate the cost constraints. Requirement generation and analysis are discussed in the companion paper presented in the Better, Cheaper, Faster session of this conference and its proceedings[6].

From these requirements a system architecture was developed. The architecture elements include a radar station, launch vehicle, spacecraft, and ground-control station.

## Preliminary Radar Station Selection

A facility survey was performed of United States government military radars, NSF and NASA research radars, space tracking radars, FAA air traffic control radars, and weather radars. The survey was filtered by frequencies appropriate to high-efficiency, trans-atmospheric WPT, by the power levels radars were capable of providing, aperture size, and tracking capability. The combination of high power production and a large focusing aperture yields power densities large enough to be measured in LEO.

Frequencies considered for atmospheric WPT are 2.45, 5.87, 10, 34, 95 and 245 GHz, due to low constituent and particulate attentuations experienced by transmissions at these frequencies. Atmospheric attenuation increases with frequency. Equipment efficiencies decrease.

Transmittable power in the megawatt range was required to provide power densities sufficient to excite the spacecraft rectenna. The CDEP rectenna was designed for 500 W/m$^2$ [7]. A rectenna designed for a Japanese in-space WPT demonstration received 24 W/m$^2$ [8]. A low power density rectenna was developed in 1990 to operate at 50 mW/m$^2$ [9]. Therefore, ground radars capable of generating at least 50 mW/m$^2$ were sought.

Another selection criterion was that the radar must have the capability to track at least 1.5 degrees per second. This requirement was derived from considering a two-minute encounter during a 180 degree pass.

Four transmitters at two facilities were initially selected as characteristically favorable in all four categories. These were the NASA radar SPANDAR at Wallops Island, Virginia, and TRADEX, ALCOR, and MMW at the Army facility on Kwajalein Atoll in the Pacific. A fifth transmitter, Arecibo, was retained for further consideration because it is capable of delivering extremely high power densities to LEO when compared with the others' capabilities. Table 2 gives their operating characteristics. The center frequencies are acceptably close to those previously considered for WPT. For example, existing hardware designed for 2.45 GHz will experience small reductions in efficiency when operating at 2.8 or 2.38 GHz. Also, if necessary, frequency-specific hardware can be designed.

Table 2: Operating Characteristics of Selected Radars

| | SPANDAR | TRADEX | ALCOR | MMW | Arecibo |
|---|---|---|---|---|---|
| Center Freq. | 2.8 | 2.95 | 5.67 | 35.0 or | 2.38 |

| | | | | | |
|---|---|---|---|---|---|
| (GHz) | | | | 95.5 | |
| Peak Power (MW) | 1 or 4 | 2 | 3 | 30 | 1 |
| Aperture Diameter (m) | 18.29 | 25.6 | 12.2 | 3 | 305 |
| Slew Rate (deg/sec) | 6 | 11.5 | 10 | 12 | 0.4 |

Figure 2 plots the power densities of the selected radars as a function of slant range. Dual scales are necessary because the Arecibo power densities are an order of magnitude greater than those which can be provided by the other facilities' instruments. The horizontal 50 mW/$m^2$ constraint limits the slant range to the 375 to 800 km range for radars other than Arecibo. Arecibo delivers 35 W/$m^2$ at 200 km and is capable of delivering 50 mW/$m^2$ out to nearly 5,300 km. Note that SPANDAR's 4 MW transmitter is similar in frequency and performance to the TRADEX radar.

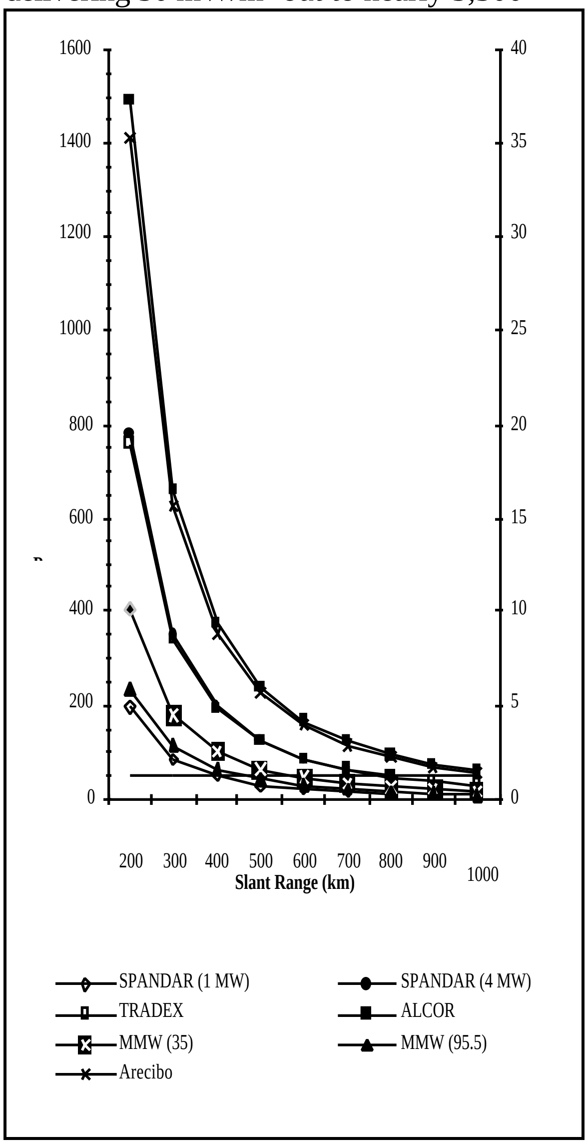



Figure 2: Power Densities of Candidate Radars

Technology readiness is greatest for rectenna hardware at 2.45 GHz. It is the only hardware which has been flight tested[10]. While hardware at higher frequencies has been developed, the combined losses due to lower hardware efficiency and greater atmospheric attenuation favored use of S-band transmitters (2-3 GHz). Spacecraft contacts with SPANDAR, TRADEX and the Arecibo facility were examined. Since TRADEX is co-located with ALCOR and MMW, mission analyses with respect to TRADEX also serve to illustrate opportunities with these instruments.

## Mission Analysis and Launch Vehicle Selection

Launch as a secondary payload (SPL) was indicated heuristically. Comparison of launch costs, provided in Table 3, confirmed this assessment. Only launch vehicles providing primary as well as secondary opportunities and those which had been successfully launched were considered in the comparison.

Representative launch vehicle companies were contacted to obtain the profiles of upcoming missions. These do not claim to be an exhaustive set of orbits available to secondary payloads, but were considered representative of those which might be used by the mission. Six opportunities in the 1997-1999 time frame were obtained. Orbital elements by vehicle are presented in Table 4.

Table 3: Launch Vehicles Providing Secondary Payload Services

| Vehicle Name | Ariane IV | Pegasus XL | Delta II | Space Shuttle (STS) |
|---|---|---|---|---|
| Cost for SPL | $150, 000 - $1 million | $4-5 million[11]<br><br>$1-2 million for microsats | Integration and launch free[12] to university missions | $1 million for reimbursable mission<br><br>$3,000 for Educational Get Away Special |
| Cost[13] for Primary Payload Launch | $45 - 50 million | $7-12 million | $60-120 million | $130 million for commercial user |

The Orbital Launch Service Project and NASA Goddard provided data for the Landsat 6 and GPS IIR missions which would launch aboard a Delta II. The Orbital Science Corp. publication cited in Table 3 described two missions, C and D. A tenth of a degree in inclination was all that differentiated orbit C from orbit D, so only D was considered. Arianespace provided some dimensions of a typical geosynchronous transfer orbit. A more complete description was obtained from the literature[14] and was the one analyzed for this study. While the Ariane IV ASAP launch is

Table 4: SPL Orbit Parameters

| Orbit | Landsat 6 | GPS IIR | Pegasus D | Ariane IV | Shuttle 200 | Shuttle 400 |
|---|---|---|---|---|---|---|
| Coordinates | ECI | ECI | ECI | ECI | ECI | ECI |
| Eccentricity | 0.3757726E-03 | 0.07909975 | 6.34682351E-7 | 0.731076 | 0 | 0 |
| Semi-major Axis (km) | 7086.36 | 7138.87 | 6298.14 | 24461.07 | 6582.14 | 6785.14 |
| Inclination (deg) | 98 | 37.62112 | 97.6 | 7.0 | 28.15 | 57 |
| Arg. of Perigee (deg) | 343.3152 | 172.75162 | 164.92302 | 25 | 0.0 | 0.0 |
| Right Asc. of Ascending Node (deg) | 175.3388 | 238.5807 | 61.2819 | 255 | 278 | 289 |
| Mean Anomaly (deg) | 359.9654 | 337.1685 | 0.0 | 0.0 | 0.0 | 0.0 |
| Period (min) | 98.89 | 100.05 | 95.65 | 634.56 | 88.57 | 92.70 |
| Mean Motion (revs/day) | 14.52 | 14.36 | 15.01 | 2.27 | 16.21 | 15.49 |

discussed, a similar capability being planned for the Ariane V was suggested.

Two nominal Shuttle orbits were considered, a 407 km, circular orbit at 57 degrees inclination and a 204 km, circular orbit at 24.5 degrees inclination.

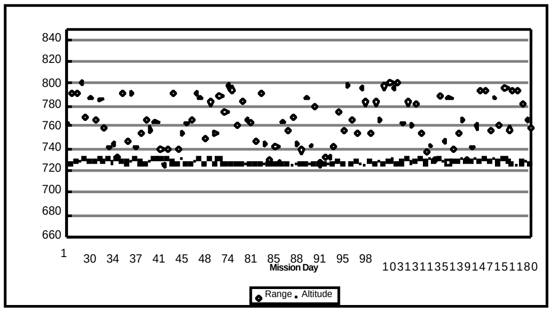
PC-TRAK™ version 3.1 was used to simulate the coverage of TRADEX, SPANDAR, and Arecibo radars from GPS IIR, Landsat 6, Pegasus D, Ariane IV and Shuttle orbits. Drag effects were not included in the analyses.

Station-to-satellite contacts recorded by PC-TRAK™ are line-of-sight. A subset of the time during which the satellite was visible, when power densities at the satellite were greater than 50 mW/m$^2$, was termed an encounter. For SPANDAR and TRADEX this meant the satellite slant range had to be below 750 km.

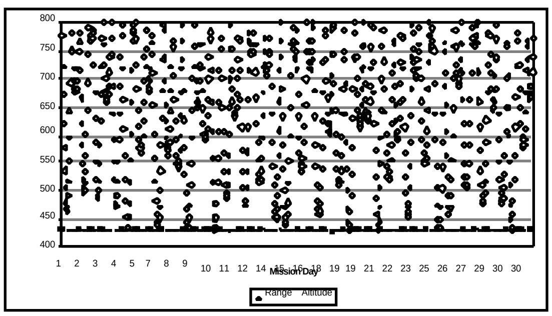
Arecibo encounters are further constrained by the brief time the vehicle will be within the essentially zenith-looking radar beam, because that station can not track the satellite. In order to keep costs down, radar/satellite combinations for which encounters were dense early in the on-orbit life of the satellite were favored. The term dense refers to encounters of relatively long duration that occur several times during a 24-hour period. Costly preparations are required to prepare radar stations for the encounters.

Precise ephemeris must be generated, and the appropriate transmitter adaptations made. In SPANDAR's case, this means arranging for the 4 MW transmitter rather than the more customary 1 MW. Repeated station preparation would be expensive.

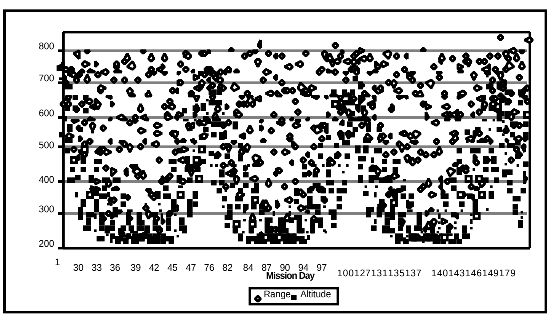
SPANDAR Encounters




A: Landsat-SPANDAR




B: STS 400-SPANDAR




C: GPS IIR-SPANDAR

Figure 3: Altitude and Range between SPL Orbits and SPANDAR

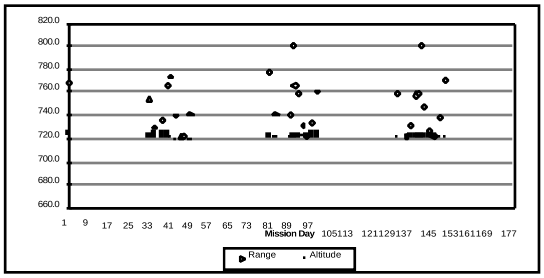
Figure 3 presents the plotted satellite altitudes and ranges from SPANDAR by mission day. Neither Ariane IV nor STS 200 orbits have encounters with SPANDAR due to their low inclinations. Pegasus D simulations did not yield any encounters with SPANDAR over the six-months simulated in this study. Figures 3-A through 3-C depict favorable orbit-SPANDAR combinations. For the Landsat orbit, there are at least 45 encounters within the acceptable power density range spread out over six months. Encounter durations are roughly one minute. The 57 degree inclination of the STS 400 orbit permits two to two-and-a-half minute contacts with SPANDAR, however there are fewer than two encounters during a 24-hour period. Of the three, the GPS IIR encounters of Figure 3-C provide the greatest encounter density. One day of data taking between days 45 and 50 would yield three to four encounters of two to two-and-a-half minutes in duration.

TRADEX Encounters

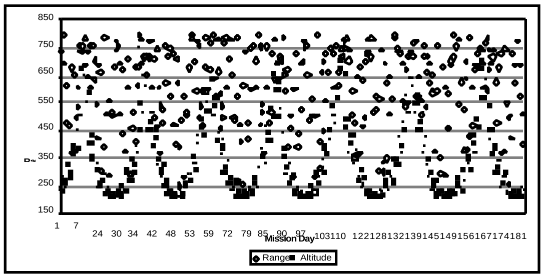
Figure 4 shows that five of the six orbits were favorable in conjunction with TRADEX. Ariane IV yielded only two links during the first four mission days followed by 220 days of inactivity. Landsat 6 encounters have a regular periodicity, but a 20-day data taking period would yield only eight encounters during that time. GPS IIR has a periodicity similar to SPANDAR, but with a lower encounter density. The STS 200-TRADEX encounter density was great enough that the time scale in Figure 4-E was expanded for clarity. The encounter densities are better than for GPS IIR and SPANDAR with two to three encounters per day of about 3 minutes in duration. The STS 400 and Pegasus D encounter densities were much lower by comparison.

Arecibo Encounters

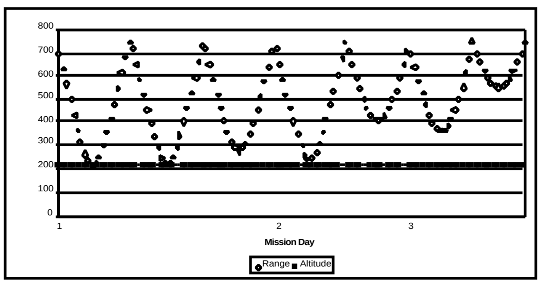
Figure 5 presents the plotted satellite altitudes and ranges from Arecibo. With its much smaller beam width and inability to track, Arecibo exhibited very low encounter densities. The high altitude, low inclination orbits of GPS IIR and the Ariane IV fared the best in conjunction with Arecibo.



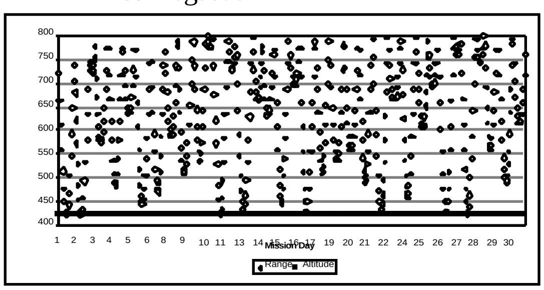

A: Landsat-TRADEX



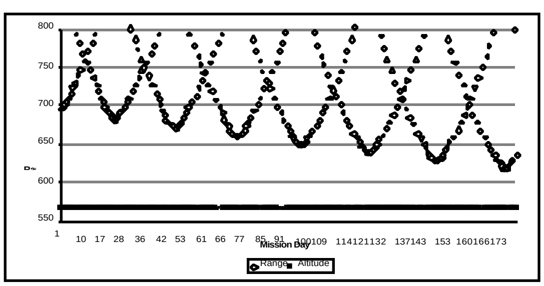

B: GPS IIR-TRADEX




C: Pegasus D-TRADEX



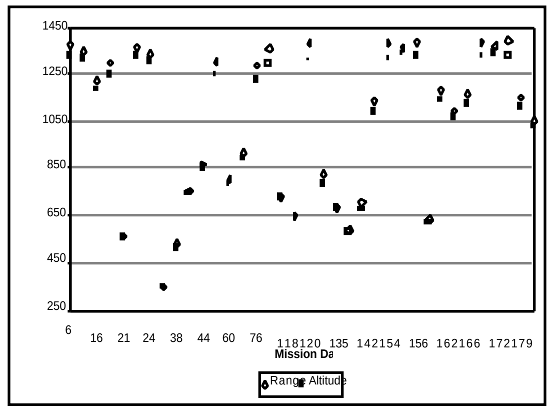

D: STS 400-TRADEX



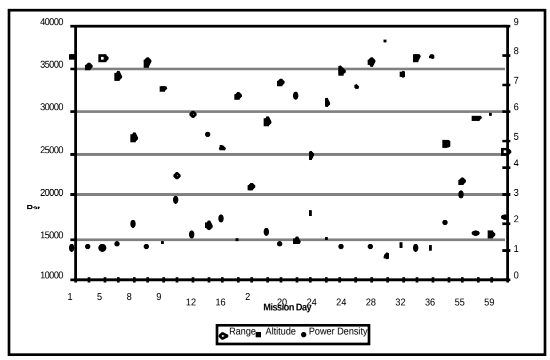

E: STS 200-TRADEX

Figure 4: Altitude and Range between SPL Orbits and TRADEX

Ariane IV was initially thought to show the most promise. A pass at perigee would be very brief – on the order of 70 milliseconds if the satellite is passing through the center of the beam. The power received would be approximately the same amount generated if you flashed a 60 W light bulb on for 1.5 milliseconds, about 0.09 Joules. It could be difficult to distinguish the signal from background noise in so brief an engagement.




A: GPS IIR-Arecibo




B: Ariane IV-Arecibo

Figure 5: Altitude and Range between SPL Orbits and Arecibo

Apogee encounters would yield periods of up to an hour when the satellite would linger near Arecibo's beam. During that time, the satellite could pass through the signal several times. Unfortunately, even the high power densities Arecibo can deliver do not acceptably extend as far as 36,000 km. The right axis of Figure 5-B calibrates the power densities (shown as filled circles) delivered during Ariane IV-Arecibo encounters as calculated based on slant range to the transmitter. The highest power density would be 8 mW/m$^2$ which is less than the 50 mW/m$^2$ minimum required.

Summarizing, using encounter density as a figure of merit, the most favorable encounters were between a satellite in the GPS IIR orbit and SPANDAR, and a Shuttle in a 204 km, 28.15 degree orbit and TRADEX. Costs for a single day's worth of coverage were solicited from SPANDAR and TRADEX operating authorities. For SPANDAR, three, 150-second encounters separated by 1.75 hours were costed. For TRADEX, the cost of three, three-minute encounters during a 24-hour period was solicited. Table 5 presents the results.

Table 5: Radar Station Costs

| SPANDAR | Cost ($) |
|---|---|
| Government-sponsored Mission | $7,000 |
| W/out Government Sponsorship | $17,000 |
| TRADEX | |
| Irradiation | $33,000 |
| Irradiation w/ Correlated Metric Data | $200,000-$400,000 |

SPANDAR costs include beaming time, the master tracking radar to which SPANDAR slaves, intercoms between the two radars, and pre- and post-encounter calibration balloons for determination of atmospheric conditions. The advisability of government sponsorship is evident.

TRADEX costs were for a university, minimum-manpower effort for irradiation of the satellite during a 1997-1998 time frame. Sponsorship was not a factor in the costing. A discriminator for TRADEX would be the requirement for true metric data such as tracking and other correlated information. This added data would require communications, mission control, and data center personnel which would result in the order-of-magnitude cost increase.

## Space Segment

The space segment was broken into the payload and the bus. Mission science will first be discussed. The spacecraft to support the achievable science goals will then be described.

### Mission Science

The science requirements were divided into the categories core requirements, atmospheric science, and supplemental science. This order is also the ranking of their importance to the mission.

#### Core Science

Core science tasks included

- Receive microwave power from the ground
- Convert microwave power into useful power
- Measure amount of power received
- Measure time dependence of rectenna efficiency
- Measure range dependence of rectenna efficiency

Reception and conversion of the beamed, microwave power is achieved by a rectifying receiver. The term rectenna, or rectifying antenna, refers to a field of half-wave dipoles (HWD). The dipole output is filtered using a low-pass filter for harmonic reflection and impedance matching, rectified using a half-wave Schottky barrier diode, DC filtered, and fed into a power bus[15]. A schematic is shown in Figure 6.

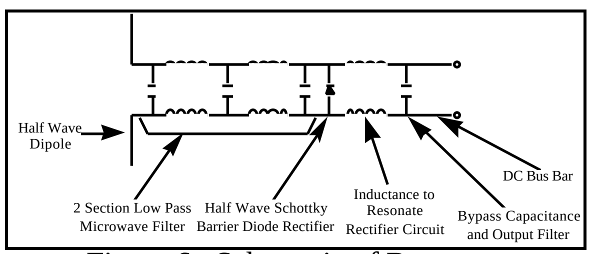



Figure 6: Schematic of Rectenna Element (Brown)

The previous studies included inflatable antennas to capture and focus the microwave beam. In this way, acceptable power densities, on the order of at least 25 W/m$^2$, were generated. The current study ruled out an inflatable dish because of the ground rule that required an operating prototype to exist early in the design process in order for the technology to be included. Only recently has the L'Garde reflector been flown. It was an optical, not an RF dish. Attitude control of a spacecraft equipped with an inflatable antenna has yet to be achieved.

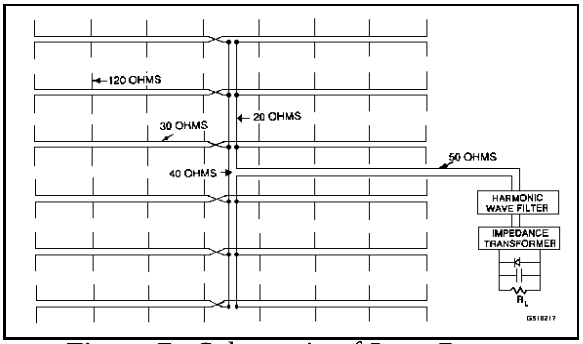



Figure 7: Schematic of Low-Power Rectenna (Brown)

A low-power rectenna[16] which aggregates several rows of dipoles with one rectifying element has been demonstrated. Shown in the schematic of Figure 7, the circuit can rectify power densities as low as 50 mW/m$^2$ with an efficiency of 54%. This efficiency is anomalously low when compared to other rectennas operating at 2.45 GHz. Efficiency can be improved somewhat through diode redesign, but the voltage drop across the diode will cause a 14% reduction when operating at low powers. Additionally, linking of the dipoles gives this rectenna a directionality around the axis parallel to the dipoles which is not present in receivers having a one-to-one relationship between HWD and rectifier.

Compensation for directionality can be accomplished by combining the low-power rectenna design with a design which was flown on the Japanese ISY-METS experiments[17]. Shown in Figure 8, HWDs are manufactured in two layers with the rectenna elements at 90 degrees to one another, thus mitigating polarization losses. In effect, a crossed-dipole antenna is created. Such an arrangement reduces attitude control requirements on the spacecraft since the most the satellite would need to rotate in the plane is 45 degrees rather than 90 to attain maximum performance.

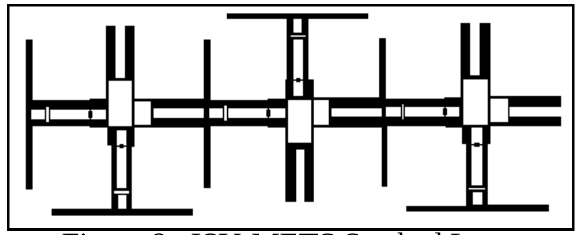


Figure 8: ISY-METS Stacked Layer Rectenna (McSpadden)

Requiring a new, hybrid design of these two rectenna versions violates the same ground rule which eliminated the inflatable antenna. However, both technologies had been demonstrated and tested at the study's inception. One was flight proven. This made the new rectenna the more attractive solution to the low power density problem when compared with an inflatable reflector. The rectenna design must be able to receive and rectify power densities from 50 to 760 mW/m$^2$. These are the maximum and minimum power densities received by the satellite in the orbits selected. Thus, with the arrangements described so far, power can be received, rectified and range-dependent performance can be assessed.

Of the radar stations considered, only Arecibo is capable of providing continuous-wave transmissions to the satellite. TRADEX can deliver high-power pulses of 3-9 microseconds with a maximum pulse repetition frequency of 1500 pulses per second. SPANDAR is capable of 0.25 to 5 microsecond pulses with a maximum of 1280 pulses per second. The response time of the rectifying circuit was a concern. Reverse recovery times (RRT), the time it takes for the diode to pass from forward bias to maximum reverse or quiescent current, was used as a figure of merit in determining the impact. Fast diodes have RRTs of 1-1000 nanoseconds. RRTs of this order would allow the rectenna to track the radar signal. Horowitz and Hill[18] refer to Schottky diodes as being applicable for high speed applications. Since these are employed in the rectenna, it is believed that the rectenna developed will be able to handle the

TRADEX and SPANDAR duty cycles.

To measure equipment efficiencies, an independent receiver is required to assess the actual power densities received at the spacecraft. A calibrated, microstrip, patch antenna will be used. A WR-350 rectangular waveguide will interface with a coaxial stub attached to the patch feed line. A bolometer will be used to measure the magnitude of the power received. A square patch resonating in the mutually perpendicular Transverse Electric, $TE_{01}$ and $TE_{10}$, modes will be used for the linear polarized waves received from the radar stations. The patch will be aligned with the rectenna dipole axes to experience the same degradation due to polarization mismatch.

Measurement of rectenna time dependence is contingent upon the sponsorship the mission receives. For the lowest cost, the mission can be completed within 24 hours of launch. This would give scant data on rectenna degradation with time. Designers proposing intersatellite WPT would find data on in-space rectenna performance over an extended period useful. A repeat of the 24-hour encounter after a hybernation period could provide the required data for only the cost of the radar station and ground station personnel.

Of the two orbit-satellite combinations, GPS IIR-SPANDAR best tests the rectenna's range performance. On a single day, slant distances from the spacecraft to the radar range over 350 km, while the minimum and maximum Shuttle-TRADEX distances differ by only 100 km.

Atmospheric Science

Atmospheric science included

- Test power density necessary to alter ionosphere
- Perform tests of ionospheric interaction effects
- Test communications interference with the microwave beam
- Measure atmospheric absorption under different weather conditions
- Examine effects of tropospheric microwave transmission

The troposphere and the ionosphere have different effects on microwave transmissions. The troposphere refracts the beam and causes time delays in wave propagation. Precipitation and temperature inversions cause absorption of microwaves and multipath-related signal fading. Atmospheric turbulence and constituent particles can cause scattering. The major ionospheric impacts on high-power transmissions are from non-linear behavior due to ohmic heating at low altitudes and thermal self-focusing at higher altitudes. Scintillation can also degrade transmissions.

Electromagnetic (EM) wave propagation will occur through the ionosphere so long as the beam frequency exceeds the plasma frequency (3-30 MHz). This is necessarily the case for transatmospheric WPT. At frequencies high when compared to the gyrotron frequency (3-10 MHz), non-linear behavior will occur when the amplitude of the incident field intensity is of the same order as the characteristic plasma field intensity. At 2.5 GHz, power densities of up to 400,000 $mW/m^2$ propagate without the development of non-linearities[19]. Since this three orders of magnitude great than the highest density which will be produced by TRADEX or SPANDAR, measurements of interactions with the ionosphere can not be successfully made. Even the more powerful Arecibo facility is not strong enough to create non-linear behavior. For similar reasons, communications interference measurements had to be eliminated from consideration for the mission.

The troposphere will cause signal attenuation due to hydrometeor scattering at the frequency of interest. The integrated water content near TRADEX and SPANDAR could be significant since both propagate over water. Weather condition sampling is integrated into the SPANDAR and TRADEX systems through associated support facilities and procedures. Correlation of coarse water content measurements with power received at the satellite can be easily incorporated into the mission data set. As with rectenna degradation performance, measurements over time are required to study these effects under different weather conditions.

Atmospheric refraction of radio waves will also have to be considered. Analytical models place signal refraction at 2.45 GHz on the order of 10 meters at altitudes of interest. Measurements of atmospheric temperature, pressure, and water vapor content are required to determine the actual index of refraction.

Supplemental Science

Supplemental Science included

- Measure radar beam sidelobe strengths and distributions

Making beam sidelobe measurements, suggested by Rogers, was not looked upon so much as a mission objective but as a potential advantage. Radar performance calibration is valuable. It could be that returning data on sidelobes could be exchanged for reducing the cost of using the radar. This will be explored during negotiations with the facility(s).

In summary, only the core science and tropospheric science goals can be accomplished by the mission. Beam sidelobe measurements could prove to be financially advantageous.

Microsatellite Bus

To increase the probability of mission success, both of the launch vehicle-radar combinations were carried through the spacecraft design. In this way if either a spacecraft was grounded or a radar station was taken off line, a backup would be available. The Delta II, GPS IIR launch was preferred because it was viewed as the more user friendly-vehicle, because the launch and integration costs were free, and because SPANDAR was the less costly of the radar stations.

Launch Vehicle Constraint Comparison

The feasibility of executing a design compatible with both launchers was examined by collecting launch environment data for both and chosing worst case, or composite, features which would suit either rocket. The Shuttle environment was generally more stringent resulting in some over-design in the satellite for the Delta II launch. As mass margins are tight, it may not be advisable to retain the Shuttle option as the design proceeds. Alternately, if the Delta II mass constraint is exceeded, launch on the Shuttle will be mandatory. Table 6 gives the constraints for the two rockets. Bolded entries indicate the more demanding requirement and the one to which the satellite is being designed.

For the Shuttle, electromagnetic interference (EMI) with normal Shuttle operation was thought to be an issue. Extrapolating a graph of EMI threshhold levels given in the Hitchhiker Customer Accommodations and Requirements Specifications (CARS)[20] from its 1 GHz upper limit out to 3 GHz showed that the maximum power TRADEX delivers is only a third of the permissible intensity for unintentional radiated narrowband emissions from payloads, therefore EMI was not a problem.

Both the Shuttle and Delta II permit secondary payloads to remain fixed to the vehicle or to a stage of the vehicle. The feasibility of doing so was investigated for both launchers.

The area available on a fixed Shuttle payload is only 45% that available from a 31 cm cube with body panels deployed. Shuttle orientation is dictated by the need of the primary mission(s) and tasks. Maneuvers to accommodate SPLs are permissible, but limited. For these reasons, it was determined that an ejected payload was preferable.

GAS missions no longer eject payloads. A program related to GAS is Hitchhiker (HH). The Hitchhiker Ejection System (HES) is available to accepted HH payloads, but its use requires that the payload be sponsored by a NASA Science or Technology office[21]. The price of a standard HH mission is given as $1.11 million for reimbursable organizations. However, since the ejection system requires NASA sponsorship, the mission would fall into the non-reimbursable category. Standard transportation and integration costs would be free. An ejected payload is not a standard service, so cost of a recontact analysis, the ejection system itself, etc. would be billed to the mission.

Delta II data on fixed and separable SPLs was taken from the Delta Launch Vehicle Secondary Payload Planners Guide for NASA Missions[22]. A fixed payload remains attached to the second stage throughout its life. The advantages of this option, besides eliminating separation complexity, are that an additional 4.5 kg is available for the satellite, and the nominal permissible dimensions grow from a 31 cm cube to 50.0 cm x 33.6 cm x 36.8 cm. Pointing and power would be available from the second stage for 7200 seconds – about one orbit. Subsequently, the attitude is at the mercy of the uncontrolled 921 kg second stage. Mission analysis showed that the most favorable GPS IIR-SPANDAR en- counters were on day 45. An encounter on day one would not be available until 19.45 hours into the mission. A non-separable payload would not survive to make a link with SPANDAR. A separable payload was dictated.

Table 6: Comparisons of Launch Vehicle Constraints

| Item | Delta II | Shuttle |
|---|---|---|
| Mass | **28.75 kg** | 68 kg |
| Nominal Dimensions | **31 cm cube** | 34 cm x 34 cm x 52 cm box |
| Max c.g. Location from Separation Plane | **10.2 cm** | 26 cm |
| Ejection Velocity | **0.6-2.4 m/s** | 0.6 - 1.2 m/s |
| Rotational Impulse | TBD | TBD |
| Minimum Resonant Frequency | 35 Hz | TBD |
| Preferred Resonant Frequency | 50 Hz | 50 Hz/**100 Hz** |
| Yield Safety Factor | 1.65 | **2.0** |
| Ultimate Safety Factor | 2.0 | **2.6** |
| Combined Limit Loads | +/- 10 G all axes | **+/- 11 G all axes** |
| Small Component Limit Loads | 20 G | **22 G** |
| Sinusoid Vibration Levels | **1.4 Thrust** 1.0 Radial/ Tangential four octaves per minute | **1.25** **four octaves per minute** |
| Overall Random Vibration | **12.9 $G_{rms}$** | **12.9 $G_{rms}$** |
| Overall Acoustic Levels | **146.2 dB Levels greater than Shuttle above 200 Hz.** 30 second test | 142 dB **Levels greater than Delta II below 200 Hz.** **60 second test** |
| Shock Spectrum | **Delta greater at lower and higher frequencies. Shuttle greater from 500 to 4000 Hz.** | |
| Pressure Venting and Venting Rate | Comparable. **Delta II's minimum boundary values are lower.** | |
| Design Envelopes | Protuberances accommodatable are comparable for both. Shuttle bulk diameter and length are greater. | |

Configuration

Use of off-the-shelf hardware was an engineering choice favored for reducing costs. Several commercial buses were considered. Most had greater capability than required for an SP or were too costly. The

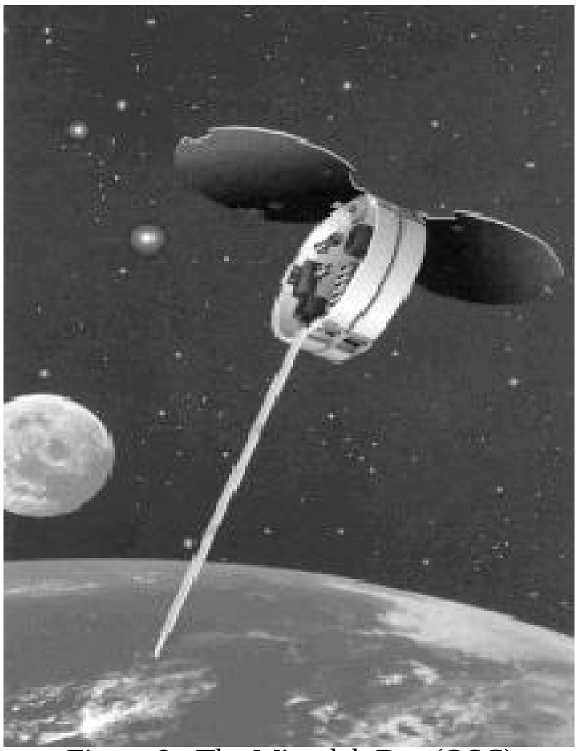


Figure 9: The Microlab Bus (OSC)

OSC Microlab[23], used for the ORBCOMM system, was intriguing. It can be launched from Pegasus, and while the Pegasus orbits analyzed were not favored, other orbits could become available. Figure 9 is an artist's rendition of the spacecraft. Note the 96.5 cm diameter paddles which open on orbit to accommodate the solar panels. If equipped with a rectenna on the bottom, the receiver would be 50% larger than that available from the baselined 31 cm cube with deployable body-face paddles. Its approximately $10 million price tag removed it from consideration, but it would be a favorable configuration for a WPT demonstration.

FREJA-C[24], depicted in Figure 10, was also considered. It is shown inverted from its flight orientation. It is an open-architecture, aluminum-frame structure with four aluminum, honeycomb paddles for solar arrays. The paddles could also support rectennas. The mass of the satellite without payload is 18 kg. Costs, in 1994 dollars, are given in Table 7.

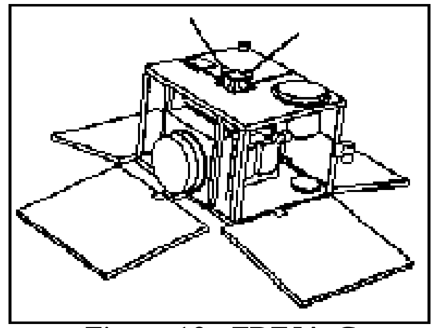


Figure 10: FREJA-C
(Swedish Space Corporation)

The cost for modifications deserves comment. Modifications would be required if a FREJA-C bus were to be purchased. For example, at 45 cm x 45 cm x 39 cm, it is too large to be a Delta II or HH secondary payload. Essentially the entire bus would have to be resized and reconfigured. The decision was made to use the cube-with-paddles configuration in a one-of-a-kind satellite developed at University of Illinois.

Table 7: FREJA-C Costs for Gravity-Gradient Stabilized Version

| | |
|---|---|
| FREJA-C Satellite Platform | $585,000 |
| Satellite Control and Data Reception Station | 70,000 |
| Modifications to Standard Design, per man hour | 100 |

The spacecraft configuration is shown in Figure 11. Like FREJA-C, it is a cubic satellite with four, deployable panels which are hinged at the satellite's bottom. Paddles are attached at the bottom, in contrast to FREJA-C, to prevent structural obstruction of the rectenna when it is linked with the radar station. Photovoltaic arrays are adhered to the upper paddle surface. The rectenna occupies the lower paddle surface and a portion of the bottom body panel. The interior cube, protected with foil during flight, is defined by four L-beams which separate the upper and lower body platforms which are constructed of honeycomb material. This arrangement allows easy access for assembly, test and maintenance.

The electronics shelf, 7.1 cm below the top panel, isolates the electronics from launch shocks and provides a thermal path for temperature control. Located on the shelf is the electronics box which houses the boards which process payload data, attitude control, telemetry, command and communications signals. The shelf also supports the control box for reaction wheels, and the secondary battery.

Two antennas are included on the bottom face of the satellite. A patch antenna is intended to provide data downlink and independent measurement of power densities received from the radar station. A canted, turnstile UHF antenna (not shown) is included for low-rate command uplinking.

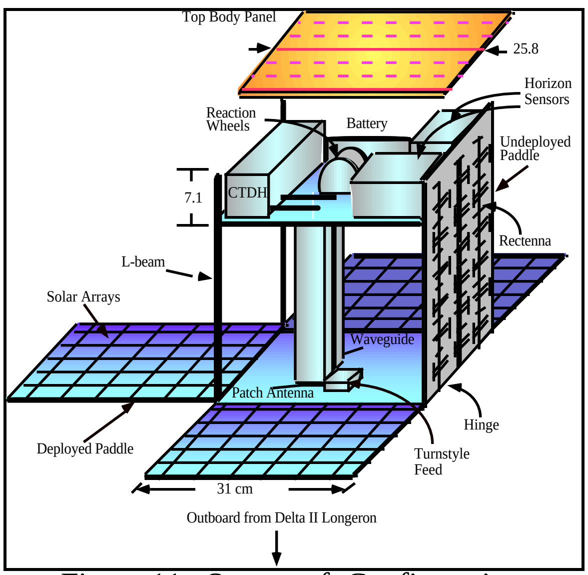



Figure 11: Spacecraft Configuration

Table 8: Mass and Power Budgets

| Subsystem | Mass (kg) | | | Power (W) |
|---|---|---|---|---|
| Structure | | AL 13.3 | Composite 6.6 | - |
| Payload | 1.1 | | | - |
| Attitude Determination and Control | | | | |
| Horizon Sensor Head (for two) | 0.7 | | | 0.6 |
| Horizon Sensor Electronics | 1.3 | | | 3.6 |
| Reaction Wheels (for three) | 2.5 | | | 5.0 |
| Reaction Wheel Control Box | 1.8 | | | 0.0 |
| Total | 6.3 | | | 9.2 |
| Communications | 0.7 | | | 4.0 |
| Command and Data Handling | 5.0 + sensors | | | 14.0 |
| Power | 2.7 | | | TBD (not considered significant) |
| Thermal Control | 0.2 | | | - |
| | Total | AL | Comp | Peak: 19.5 |
| | | 29.3 | 22.6 | Avg during encounters: 12.0 |
| | Limit | 28.8 | 28.8 | |
| | Margin | 1.7% | 21.5% | |

Table 8 gives the mass and power budgets for the spacecraft. Note that an all-aluminum structure resulted in a negative mass margin. It is necessary to use composite materials for the panels and shelf to maintain a 21.5% mass margin.

Subsystem Issues

An attempt was made to reduce satellite mass and power consumption by eliminating the need for on-board data storage. The ability to make simultaneous links to the Wallops Island radar station and the ground station on the university campus in Champaign, IL was investigated. Orbital geometry permitted this for only a portion of SPANDAR encounters. Figure 12 presents the geometry of the dual link. At the start of most encounters, the satellite is at point A so the two ground locations are in the same longitudinal direction. During the radar encounter, the geometry becomes that of B, with the satellite between the two stations. To uncouple rectenna and downlink antenna pointing, a second antenna, or a rotational maneuver would be required to maintain the simultaneous link. Each of these solutions lead to an increase in mass. Simultaneous downlink was disallowed.

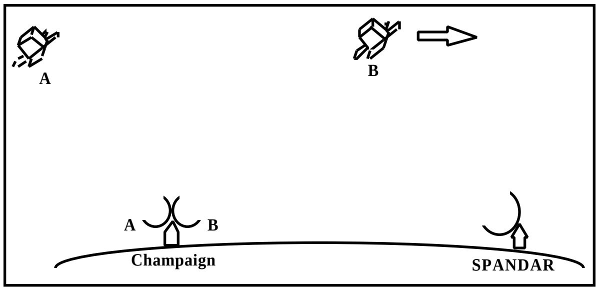



Figure 12: Communications Geometry

Requirements for an onboard computer were derived. Throughput was estimated at 233 KIPS. The memory required for processing is 130 kilobytes with an additional 60 kilobytes required for data storage. A small, low-mass unit was sought. Three alternatives were considered. The IBM-3 with 1750A processor recommended itself at 1.5 kg and 1475 $cm^3$. The cost of this unit is on the order of two million dollars[25]. A more cost-effective approach was sought.

A commercial laptop was considered. These can be obtained for less than $7,000. They are equipped with built-in modems which could be linked to the antenna system for up- and downlink Laptops weigh about three kilograms with the battery installed. A laptop would have to extend through the electronics shelf and into the honeycomb body panels to fit on the spacecraft. Removing the case would reduce the dimensions somewhat.

A VME bus with electronics box to house the processing boards was selected. The processors are easily produced within the university environment using commercial chips, PALs and EPROMS. Eight boards would be required for transmitter, receiver, RF carrier signal generation, processor and memory, interface, encryption, modem and attitude determination and control. The 5 kg unit was estimated to use 14 W of power.

The attitude control system represents the tall pole of the mission. Most small satellites employ the inexpensive and lightweight gravity-gradient or spin stabilization methods. In order to keep the rectenna normal to incoming microwave beam, off-nadir angles as great as 46 degrees must be accommodated. The satellite will have to slew as much as half a degree per second during four of the six SPANDAR encounters. This high slew rate has potentially detrimental effects on the rectenna paddles.

Until further work is done, placeholders were included to complete the initial configuration. Mass and power were allotted for three reaction wheel assemblies. These were scaled by total satellite mass from MSTI-2 units and then doubled to provide what are hoped to be conservative mass and power estimates. Because satellites of this class are not generally three-axis, high slew rate vehicles, obtaining off-the-shelf reaction wheels may be difficult, expensive, or impossible. The problem is challenging, but does not seem unreasonable to solve. Achieving the required torques for the high slew rates should not be out of reach for a satellite of this mass. The mass margin is great enough to accommodate a three-fold increase over what has been allocated for the system. The battery will necessarily increase as well and this will have to be included in system growth allowances.

Attitude determination is aided by the fact that mission-critical events are scheduled to take place when the satellite is sun lit. Sun sensors will be used during encounter. Two Barnes 13-477 horizon sensors have been included for roll and pitch determination during eclipse, but more work needs to be done in analyzing the eclipse condition.

A mission profile was developed to aid in power scheduling. Four distinct usage regimes were identified – encounter, data downlink, time between encounter, and a 47-day period between data taking. This last mode anticipates that a second data taking period will be funded to determine system performance over time. Only the processor board will receive power during 47-day hiatus. An on-board timer will turn the other subsystems back on for the second round of data gathering. An attempt was made to run the entire mission on batteries only. With the system requiring 670 W-hr, the batteries only solution became too massive. A combination of solar arrays and secondary batteries was required. Because the 1.2 kg arrays are mounted low on the spacecraft to prevent obscuration of the rectenna, a 25 percent reduction in illuminated area is expected. Paddle area was sufficient to accept this reduction and still power the 40 W load. Losses could rise to 50 percent without detrimental effects.

Secondary batteries were sized to provide power during the unilluminated part of the encounters. The rectenna will be facing into the sun during a portion of the encounter, so the arrays will be in shadow. The batteries need to provide 13.3 W-hr of power. A 12.7 cm diameter, 6.7 cm long, nickel hydrogen battery with specific energy density of 40 W-hr/kg added 0.8 kg to the mass total. A 0.7 kg allowance, 2.5 percent of the satellite's total available mass, was made for power conditioning.

The thermal control system design has yet to be accomplished.

## Cost Summary

Table 9 summarizes the costs for the preferred Delta II-SPANDAR mission. The costs for SPANDAR and the Delta II were obtained directly from NASA. Ground station cost, taken from the FREJA-C cost summary, assumes that one must be purchased. The price given is considered a rough order of magnitude since the requirements for the ground station and suitable commercial hardware were not compared.

Table 9: Mission Costs

| Item | Cost | Note |
|---|---|---|
| SPANDAR Radar Station | $ 7,000<br>17,000 | Government Sponsored<br>Unsponsored |
| DELTA II LV | 0 | University Mission |
| University-Run Ground Station | 50,000 | Swedish Space Corporation |
| Spacecraft | 1,170,000 | Analogous Costing Used |
| TOTAL | $ 1,227,000 | Government-Sponsored |

A ground-up estimate of the spacecraft's cost was not made. Instead, advantage was taken of its similarities to the FREJA-C bus in making a cost estimate. The FREJA-C cost was quoted in 1994 to be $585,000 including vibrational testing. This was given for a gravity-gradient stabilized spacecraft. The FREJA-C cost for a spin-stabilized version was only $525,000. The current mission will require a more complex attitude determination and control system. The cost of the payload, the measurement circuit and the rectenna hardware development, construction and integration, must also be accounted for. An additional consideration is the university environment in which the satellite is to be produced and the stipends to students who will do the manufacturing.

In light of these considerations, the FREJA-C cost was doubled. There is no quantitative basis upon which to do this, but it is felt to be a conservative estimate. Recall that the cost for spacecraft development was limited to two million dollars. FREJA-C costs could be increased by 341 percent before this restriction is violated. The requirement is, for the moment, considered to be satisfied.

In Table 10, the current mission's cost is compared with the two previous studies. The Alaskan cost is halved because it included a laser WPT portion as well as microwave. This does not reflect a fair comparison since that cost includes a $17 million, dedicated Pegasus launch. This launch cost was deducted to arrive at a $7 million cost. Roger's mission's was designed for an Ariane IV. Launch costs for that mission were estimated at $500,000.

Table 10: Cost Comparison With Previous Studies

| Study | $ Million | Notes |
|---|---|---|
| Rogers | 8-10 | |
| Univ. of AK | 24 | For Half |

| | | Mission |
|---|---|---|
| Univ. of AK | 7 | Half Mission Minus LV |
| Current Mission | 1-2 | |

## Conclusion

A realizable, low-cost, WPT mission has been described. It will not demonstrate power beaming at the levels of a full-scale SPS, but some of the issues involved in trans-atmospheric WPT will be investigated. An assessment of integral hardware performance can be made.

Government sponsorship is currently being sought in order to obtain the cost advantages described in this paper.

Once sponsorship is obtained, a revised target launch date will be set and secondary payload opportunities reanalyzed for that time period. Trends for commonly available orbits have been established, but specific contact times will have to be developed and concepts of operations firmed up so the radar time can be accurately assessed and cost estimates revised if necessary. Fidelity of all satellite subsystem designs must be increased.

The utility of this mission is the enhancement of empirical WPT design literature in a way that addresses the NRC concerns about cost and the all-or-nothing nature of SPS. This mission is viewed as part of a tiered program of developing the technologies which would lead to an operational SPS while permitting go or no-go decisions to be made between missions before a substantial sum need be invested.

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