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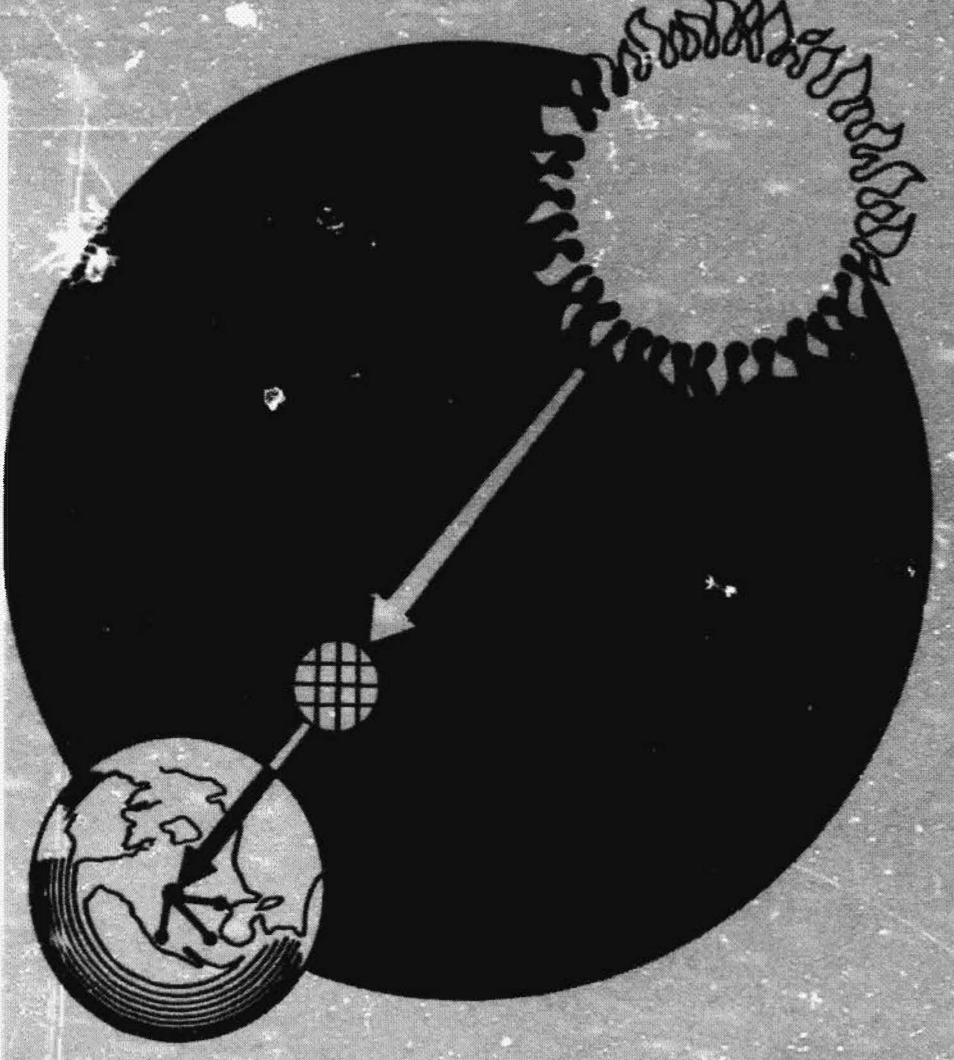

SATELLITE POWER SYSTEM

ENGINEERING AND ECONOMIC ANALYSIS

SUMMARY

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812
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ABSTRACT

A system engineering and economic analysis was conducted to establish typical reference baselines for the photovoltaic, solar thermal, and nuclear Satellite Power Systems. Tentative conclusions indicate that feasibility and economic viability are characteristic of the Satellite Power System and, therefore, deserve additional study. Many key issues exist but appear resolvable with the anticipated technological advances of the next decades.

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FOREWORD

The George C. Marshall Space Flight Center of the National Aeronautics and Space Administration in Huntsville, Alabama, for the past 5 years has investigated the applications of space technology and of space itself to the solution of specific terrestrial energy problems. Great progress has been made in this brief period. Our pioneer work in microwave power transmission and in solar heating and cooling of residential and commercial buildings has lead to further development and progress in these areas. As a natural extension, the collection of solar energy in space using a Satellite Power System, first postulated by Dr. P. E. Glaser, has also been under careful study and review at this Center. Substantial in-house efforts combined with numerous industrial contracts succeeded in defining critical program elements, overall system and subsystem requirements, and necessary technology advancement requirements.

The main thrust of MSFC's efforts is in the overall systems engineering and integration of Satellite Power Systems including supporting systems such as transportation, space construction base, and large space structures.

This summary report presents our findings to date pertaining to the unprecedented number of systems, subsystems, and operational elements with complex interrelations. The many elements of the program were broken down into manageable entities and their most sensitive parameters were defined. Numerous tradeoffs between options proceeded through increasing levels of depth in order to clearly show all areas that need concentrated technology advancement efforts.

A carefully structured cost and economic analysis was carried out concurrently with the generation of technical data. With the conservative assumptions concerning cost growth and findings of these efforts, construction of the SPS could be technically and economically possible toward the end of this century.

TECHNICAL MEMORANDUM X-73344

SATELLITE POWER SYSTEMS
AN ENGINEERING AND ECONOMIC ANALYSIS SUMMARY

1.0 INTRODUCTION

1.1 BACKGROUND

Rapidly increasing rates of consumption of the Earth's available fossil and nuclear fuel stores are characteristic of this latter half of the 20th century. Global population is increasing, as is that fraction of the population which forms the energy consuming "middle class." This is true not only in the United States, Russia, Japan, etc., but also in what are termed emerging nations. As a consequence, we may expect existing global energy sources to last only to the following approximate dates: oil, 1995 to 2005; coal, 2030 to 2080; and uranium (without breeder reactors), 2020 to 2050. As these energy sources are consumed, four additional factors emerge. First, their cost steadily increases as remaining quantities become more difficult to obtain (e.g., coal veins become thinner). Second, their consumption releases additional pollutants to the biosphere (e.g., CO_2 removed from the atmosphere over thousands of years by plants, which formed coal, is now being returned). Third, since energy sources are geographically concentrated (e.g., most coal reserves are in the United States, and most oil resources are in the Middle East), a potential for great international tension and possibly war may be created as reserves dwindle. Fourth, nuclear fission involves byproduct materials that may be used for weapon production by either governments or outlaws.

Thus, some attention is now turning to "renewable" or "nondepletable" energy sources. Primary candidates for electric power appear to be nuclear breeder reactors, nuclear fusion, and solar energy. These are characterized by varying degrees of complexity, technical risk, pollution, cost, etc. Each could reduce our dependence on imports and, if adopted by other nations, serve to reduce international tensions.

Solar power may be used directly for heating and cooling; it may also be used for the production of electricity. Primary concepts for electric power production on Earth are photovoltaic (solar cell arrays or "farms") and the thermal engine "tower top." In the tower top concept a field of steerable mirrors (heliostats) focuses energy onto a tower-mounted heat absorber. This heat can provide steam or some other fluid to turn turbogenerators.

Solar power plants on Earth suffer from the diffuse nature of solar radiation (insolation), reduction in insolation from clouds, haze, etc., the varying angle of the Sun's rays, and, of course, nightfall. A power plant located in space can receive nearly direct, unfiltered sunshine almost without interruption. For a given reception area, a space system will receive six times more energy per year than will the areas receiving the most sunlight on Earth and approximately 15 times more energy than a United States location with average weather.

In geosynchronous orbit 35 786 km above the Equator, a satellite has an orbital period of 24 h and remains in constant line of sight to stations on the ground. Solar power satellites in such orbits would generate electric power that would be converted to microwaves and beamed to receiving stations for distribution to consumers as conventional electric power. Figure 1-1 shows how receiving stations in various parts of the United States could be associated with a number of satellites in orbit.

Thus, satellite systems can provide high availability "baseload" power without the energy storage or backup facilities that greatly impact the cost and operational flexibility of terrestrial solar power stations. Space also offers these advantages:

- Thermal pollution from the power generation process is released in space rather than to the biosphere.
- The low gravity potential permits low-mass construction of the large areas necessary to intercept the solar energy. Consequently, the total amount of resources used is less than that for ground solar stations.
- There is no oxidation or corrosion.
- There are no tidal waves, earthquakes, etc.
- The satellite systems are far removed from demonstrators, terrorists, etc.

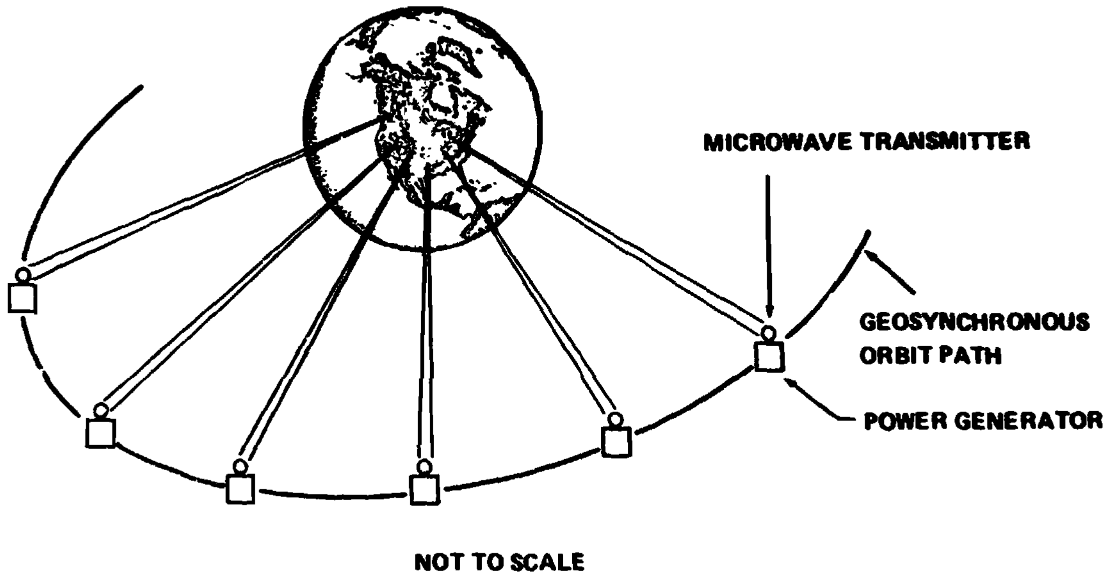


Figure 1-1. Solar power satellites.

1.2 POTENTIAL SYSTEMS

1.2.1 POWER TRANSMISSION

All solar power satellites will require a microwave power transmission system (MPTS). A circular transmitter that is nominally 1 km in diameter would mount the "tubes" which convert electric current to microwaves. This "phased array" uses a combination of mechanical and electronic steering to direct the beam to the receiving station. Phase control is implemented by a pilot beam transmitted from the receiver. Safety interlocking could be used, whereby turning off the pilot causes the power beam to lose coherence so that it is harmlessly dispersed. The receiving antenna has a nominal width of approximately 8 km. It consists of an array of small dipole antennas with integral power rectifiers and filters. The output is processed as required for compatibility with local power grids. It may prove possible to conduct farming beneath these antennas, since only a fraction of the sunlight would be intercepted.

1.2.2 PHOTOVOLTAIC POWER GENERATION

In this potential system solar cell arrays are connected in series and in parallel as required to provide the high voltage necessary for the transmitter. Several types of solar cells may be considered as candidates: silicon (used on current spacecraft), gallium arsenide heterojunction, and several thin film types such as cadmium sulphide.

Solar cells require several advances to provide maximum benefit for power satellite use. These advances are increased efficiency of conversion of sunlight to electricity, lighter weight (i.e., thinner), increased resistance to space radiation, and, of great significance, lower cost.

One approach to cost reduction is the use of solar concentrators. Mirrors of thin metal or metallized plastic film would shine additional solar energy onto the cells. This causes increased cell output, so that fewer cells per satellite are required. Concentration ratio selection must, however, take into account the heating of the cells which tends to occur, since heating reduces efficiency. Cell cooling, possibly with metal fins, is an additional possibility.

A lightweight structure of limited flexibility is required to support the cells. This structure will probably also be called upon to act as a power distribution system, i.e., carry electric power from the cells to the transmitter.

An attitude control system is also required to align the solar arrays to the sunlight, providing a "base" from which to point the transmitter antenna. Since the arrays face the Sun and the transmitter must face the Earth, a rotating joint must be provided. The electric power must cross this joint on its way to the transmitter.

1.2.3 THERMAL ENGINE POWER GENERATION

In this concept reflecting mirrors, probably in the form of a paraboloid, focus solar energy into a cavity absorber, i.e., the rays enter a hole in a sphere. This insulated sphere contains a heat exchanger assembly composed of an array of tubes. Gas (probably helium) flowing through these tubes picks up thermal energy. The gas flow expands through turbines which turn the electric generators. The turbines also turn compressors which route the gas through the remainder of the system and back to the cavity absorber to collect more energy.

This system is referred to as the "closed Brayton cycle" and has been demonstrated for Earth-based use in sizes up to 50 MW. In a power satellite a number of engines (e.g., 50) may be used to promote redundancy. The major moving assembly of the engines, the turbocompressor, is supported on gas bearings for long life.

The second law of thermodynamics permits useful work to be removed only from a heat engine which has a temperature differential across it. Thus, a cooling system is required in the form of a radiator to reject heat to space. This cooling system and a heat exchanger (called a recuperator) are also part of the gas circuit. The radiator utilizes pumps which circulate a liquid metal that has picked up heat from the helium flow. The liquid metal passes through panels composed of tubes and fins that dissipate the heat. Meteoroid puncture of these tubes is a potential problem; however, it is possible to align the panels "edge on" to the prevalent meteoroid direction so that the puncture rate is acceptable. An alternative radiator panel would be one composed of heat pipes in which each has its own inventory of metallic working fluid.

As with the photovoltaic concept, structural and attitude control systems are required, along with the transmitter and its rotary joint.

1.2.4 ADDITIONAL POWER GENERATION ALTERNATIVES

The thermionic converter is a potential alternative to thermal engines or solar cells. These passive devices use high temperature thermal energy to produce direct current electricity. High solar concentration ratios (over 1000) are required to achieve the necessary temperatures. Because of the high temperatures, the necessary cooling can be accomplished by fins attached to the thermionic diodes.

Finally, instead of a solar energy source, it may be possible to use nuclear reactors. These could energize turbomachines to produce electric power for the microwave transmitter. Breeder reactors could be used to extend our uranium resources; "bomb grade" fuels bred in these reactors would remain in geosynchronous orbit.

1.2.5 AUXILIARY SYSTEMS

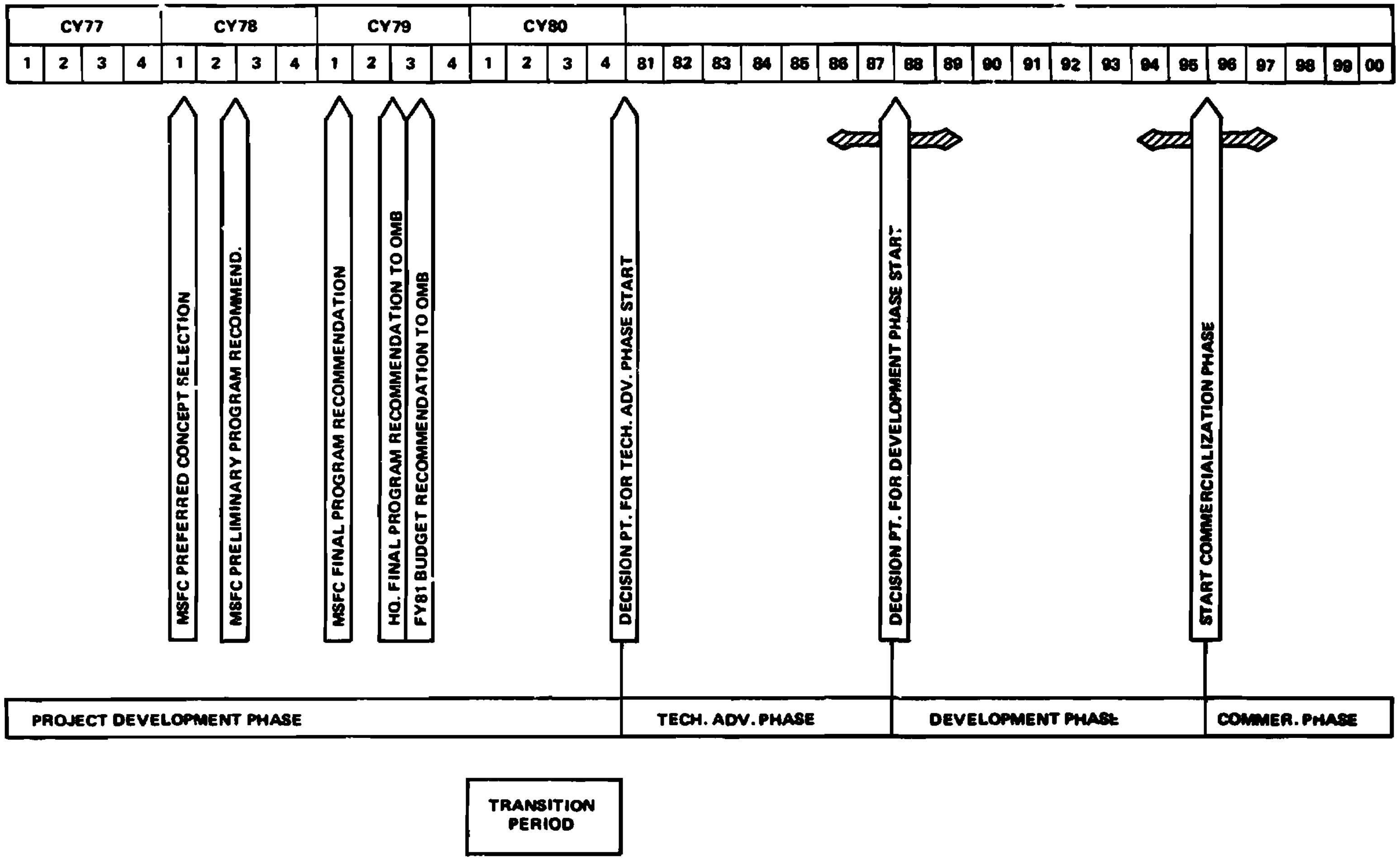
The space transportation of power satellites is generally considered to take place in two stages. Large reusable "space freighters" would be used to reach low Earth orbit. Another orbit transfer system would be used on the way to geosynchronous orbit.

1.3 PAST WORK

The possibility of satellite power systems (SPS) as a potential new source of terrestrial energy was first postulated, analyzed, and published by Dr. P. E. Giaser in 1968 and patented in 1973 [1-3]. The George C. Marshall Space Flight Center (MSFC) followed the evolution of Dr. Glaser's concept with great interest and proposed a major systems definition study in 1971. Consequently, MSFC, upon request by the NASA Office of Applications (OA), developed a "Solar Power Utilization Plan" in early 1972, which was followed by a "Twenty-Year Solar Power Development Plan" in the same year. Major systems studies were begun in 1974 and 1975, both under contract to industry and in-house by NASA [4-19].

This document is a status report of recent MSFC in-house study activities. These study efforts are a part of and follow an overall SPS program schedule, as shown in Table 1-1.

TABLE 1-1. SPS PROGRAM SCHEDULE



2.0 STUDY OBJECTIVES AND GUIDELINES

The satellite power system is envisioned as a supplemental source of energy for terrestrial applications late in the 20th century. This study has postulated program options and spacecraft designs that can lead to contributions of 10 to 30 percent of the total electric energy needs of the United States beginning in the 1990's. While considering the application of the SPS for the continental United States, some consideration has been given also to how the SPS might be utilized as an exportable resource. Future studies should explore more detailed implications of the SPS as a national exportable resource.

Early study planning indicated a need to establish reference baseline designs to provide appropriate departure points for system sensitivity studies. These reference baselines included the photovoltaic, solar thermal, and nuclear concepts with major study emphasis being directed to the photovoltaic and solar thermal conversion concepts.

The summary guidelines for the study were:

- 10 GW power output at rectenna/utility power interface*
- 20 mW/cm^2 maximum power density at rectenna center
- 30 year lifetime for system operation with a reasonable repair/refurbishment/maintenance philosophy
- One additional SPS brought into operation each year*
- Each satellite to be assembled in low Earth orbit and transferred to geostationary orbit*
- Assumption that a space station will be available in low Earth orbit to support assembly of the SPS and that some form of space station will be available in geostationary orbit to support operations and maintenance.

After the preliminary study investigation, trade studies suggested that some of the initial guidelines (denoted by asterisks) be considered for change:

- Basic design parameters indicated that 10 GW satellites would nominally have two or more microwave antennas, with some trade studies indicating that smaller satellites with one microwave antenna might be more economical.

One satellite per year may not be commensurate with developing a large contribution to energy needs late in the century.

- Although the initial engineering position is based on fabrication and assembly in low Earth orbit, a better understanding of fabrication and assembly in geostationary orbit will be most desirable.

3.0 DEFINITION OF SPS REQUIREMENTS

The concept of a system of space-based solar power satellites does not yield or derive its being from a conventional requirement or set of requirements. In general, the SPS is proposed as a supplemental source of electrical energy for terrestrial applications. The need is evident, but broad acceptance of SPS to satisfy the need involves the resolution of many complex issues such as economic competitiveness and environmental concerns. The total commitment to SPS is dependent upon a long term "learn as you go" process leading to a potential application during the last decade of this century.

4.0 DETERMINATION OF SPS PROGRAM ELEMENTS

An SPS responsive to the overall system requirements encompasses a large number of interrelated program elements and subelements. An approximate total of 600 of these have been recognized to date. Each one has to be considered while defining candidate SPS concepts, and each one is a part of the engineering and economic analyses.

The major SPS elements discussed in the following paragraphs form the building blocks of the overall study plan and are summarized in Figure 4-1.

4.1 SATELLITE POWER STATION

This element encompasses the power conversion options being studied. In addition to the photovoltaic and thermal solar energy conversion options, contractor studies are being performed on nuclear energy conversion systems (Appendix B). Each conversion option uses a rather similar microwave power conversion and transmission system as part of the satellite power station.

4.2 GROUND RECEIVING AND DISTRIBUTION SITE

This element includes the receiving antenna for the microwave beam, the utility interface with the related electric ground distribution system, the safety system related to microwave exposure protection and to the safety of maintenance and service activities, and the maintenance and service system.

4.3 MANUFACTURING, CONSTRUCTION, AND MAINTENANCE OPERATIONS

This element includes ground and orbital operations and their respective systems that support the required manufacturing, construction, assembly, and maintenance activities. A special operations management activity ties together equipment and manned operations and transportation and logistics requirements.

4.4 SPACE TRANSPORTATION

This element consists of five transportation systems necessary to provide operational satellite power systems: the heavy lift launch vehicle (HLLV), the personnel launch vehicle (PLV), the cargo orbital transfer vehicle (COTV) system, the personnel orbital transfer vehicle (POTV) system, and local space transportation vehicle (LSTV) systems.

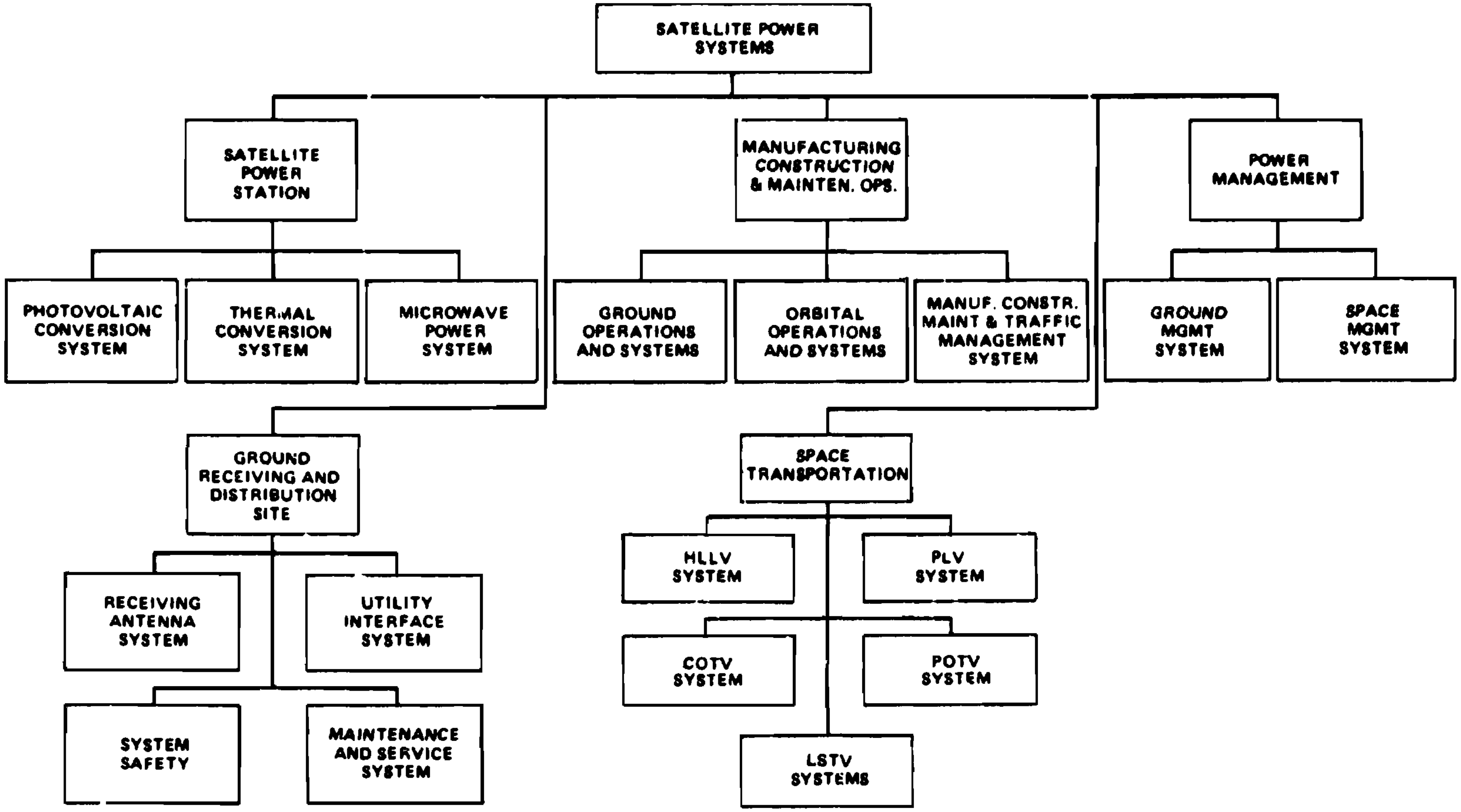


Figure 4-1. SPS program elements.

5.0 DEFINITION OF SPS CONCEPTS

5.1 SOLAR PHOTOVOLTAIC

The initial photovoltaic concept developed for study of the SPS is illustrated in Figure 5-1. Overall dimensions for this configuration, designed to supply 10 GW of power to the utility interface, are a maximum length of 21.05 km, a width of 9.53 km, and a structural depth of 0.215 km. The antenna is located at the center of the configuration to minimize distribution losses. With the selected 20 kV dc distribution voltage, these losses are a greater mass factor than the nonmetallic carry-through structure at the center of the antenna. The significant mass influence of power distribution at 20 kV can be effectively eliminated by distributing power at 40 kV. The nonmetallic structure could possibly be eliminated by using the power distribution mast as the central load carrying structure.

The solar array of Figure 5-1 is sized for a perpendicular attitude with respect to the Sun. This minimizes the solar array area and, in turn, the mass of the array. However, the mass of the attitude control propellant, the complexity of the rotary joint to correct for the rotation (approximately 23°) of the antenna with respect to the rectenna, and the complexity of the attitude control system to maintain solar pointing are increased. Early study results indicated that comparison of the solar perpendicular attitude to a perpendicular to orbit plane attitude results in comparable overall system masses; however, the lower complexity of the perpendicular to orbit plane attitude and later study mass estimates suggest a trend toward this attitude for future configurations.

The solar array consists of trapezoidal shaped modules that are 493 m square at the top and 215 m deep. The sides of the trapezoid are typically aluminized film reflectors that provide a concentration ratio of two for the solar cells located at the base of the trapezoid. The solar cells are passively cooled.

The planform shape of the photovoltaic configuration is elliptical. This shape is an efficient structural configuration with regard to the distribution of large quantities of current to the geometric center and, when analyzed for attitude control propellant consumption, results in a slightly lower mass system in comparison to an optimized rectangular configuration, because the elliptic shape is widest at the center where the structural bending moment is greatest. Pertinent descriptive data for this configuration are shown in Table 5-1 for a two antenna configuration.

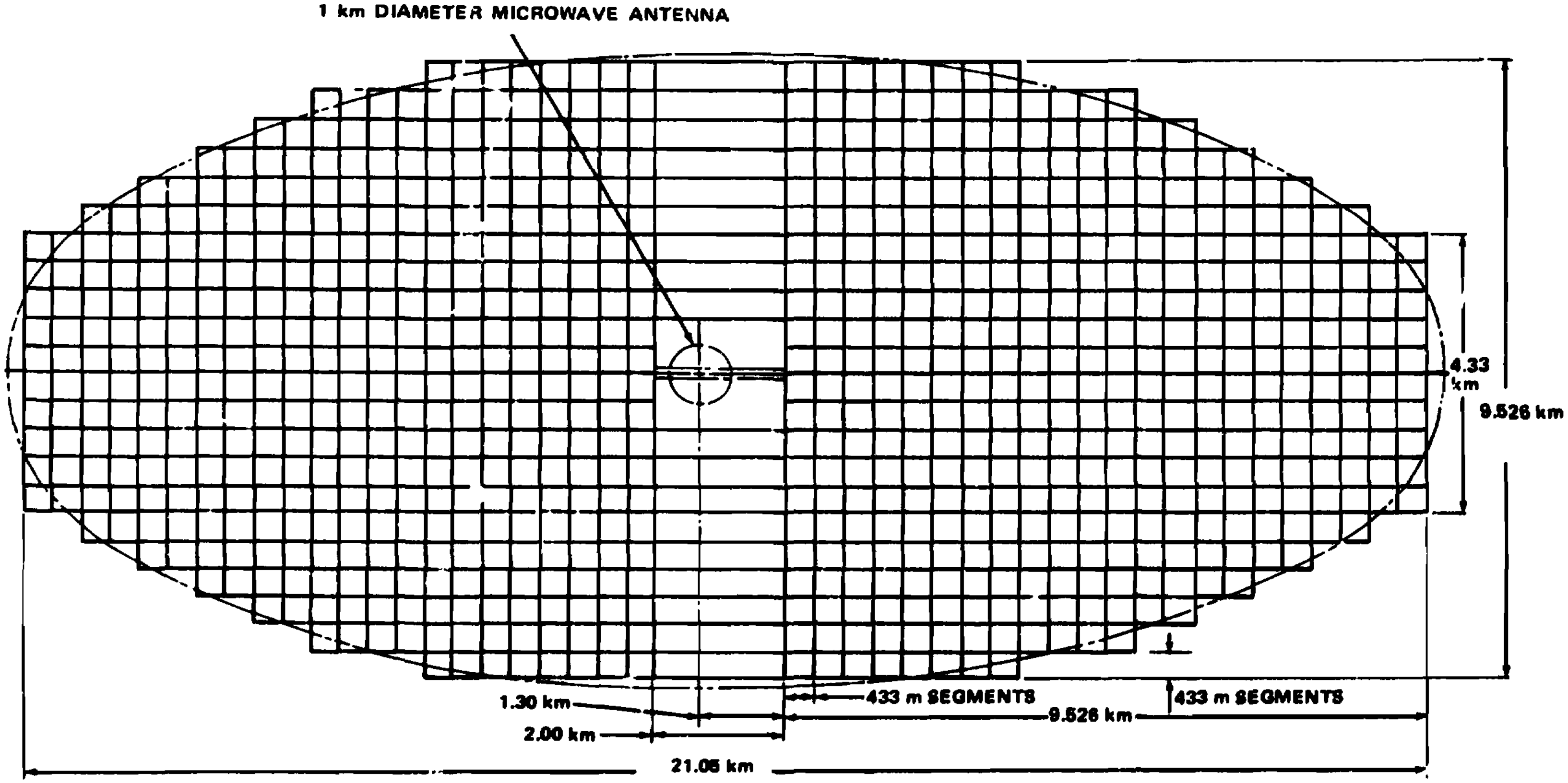


Figure 5-1. Solar photovoltaic SPS configuration.

TABLE 5-1. SUMMARY CHARACTERISTICS OF THE PHOTOVOLTAIC SYSTEM

Basic Characteristics		Subsystem Weights		
Subsystem	Current Concept Characteristics	Subsystem/Component	Weight (kg × 10^6)	
Solar Array		Solar Array		73.01
Concentration Ratio	2 	Blankets	46.48	
Total Array Area (Planform)	191.53 km^2	Concentrators	4.17	
Solar Blanket Area	88.04 km^2	Tension Mechanism and Hardware	1.01	
Cell Weight/Area	0.513 kg/m^2	Nonconducting Structure	7.34	
Watts Output/Module Area	237 W/m^2	Buses, Switches	9.54	
		Mast	4.46	
Microwave Antenna (Two)		Microwave Antennas (Two)		14.48
Diameter	1 km	Amplitrons	4.76	
DC-RF Conversion	Amplitron	Waveguides	6.84	
Configuration		IRF Amplifier	1.36	
Shape	Elliptical	Phase Control Electronics	0.08	
		Power Distribution	0.80	
Attitude Orientation	Perpendicular to Sun	Contour Control	0.24	
		Structure	0.40	
		Rotary Joints (Two)		0.40
		Mechanism	0.18	
		Structure	0.22	
		Control Systems		1.73
		Actuators	0.87	
		Propellant/Year	0.86	
		Subtotal		89.91
		Contingency (30%)		26.97
		Total System		117.09

The solar array area is 88 km^2, the antenna diameter is 1 km, and the system mass is approximately 117×10^6 kg, which includes a contingency of 30 percent.

Alternate photovoltaic configuration concepts are shown in Figure 5-2. These concepts feature two antennas, one on each side of the SPS or at each end, and in each option the antennas are nominally 5 GW each. A side location minimizes distribution losses but causes an attitude control propellant increase, because this offsets the advantage of having the major axis of the elliptical shaped SPS perpendicular to the orbit plane for minimum attitude control propellant.

Two antennas, when used to transmit 5 GW each of power to the ground, will reduce the power density to one-half as compared to transmission of this power by one antenna of the same size. This is potentially a desirable configuration change, since transmission of 10 GW of power with a 1 km diameter antenna from a geosynchronous Earth orbit at a frequency of 2.45 GHz results in a peak power density at the center of the rectenna above 20 mW/cm^2.

For the transmission of 10 GW of power, two complete 5 GW power systems are lower in weight than a single power system with two antennas because of the large distribution losses for the larger solar array when distributing power at 20 kV. This conclusion could change if the distribution voltage was raised to 40 kV. In any configuration, each transmitting antenna has a companion ground rectenna.

An end location of the antennas is shown by the configuration at the bottom of Figure 5-2. This antenna location does effect minimum attitude control propellant, but distribution voltage must be high to reduce distribution power loss.

5.2 SOLAR THERMAL CONCENTRATOR

A typical solar thermal concentrator configuration is shown in Figure 5-3. This configuration consists of 544 independent modules sized to minimize demonstration costs by the delivery of individual components to Earth orbit by the shuttle. The number of independent modules for solar thermal concepts which have been studied ranges from 4 to approximately 550. Recent optimization activities indicate the appropriate number of independent modules would be nearer the small end of the spectrum. AC power is distributed from the modules of Figure 5-3 to antenna locations, where the power is converted to dc and transmitted to the ground. Each power module consists of a concentrator, with a concentration ratio of approximately 2000; a "light pipe" absorber

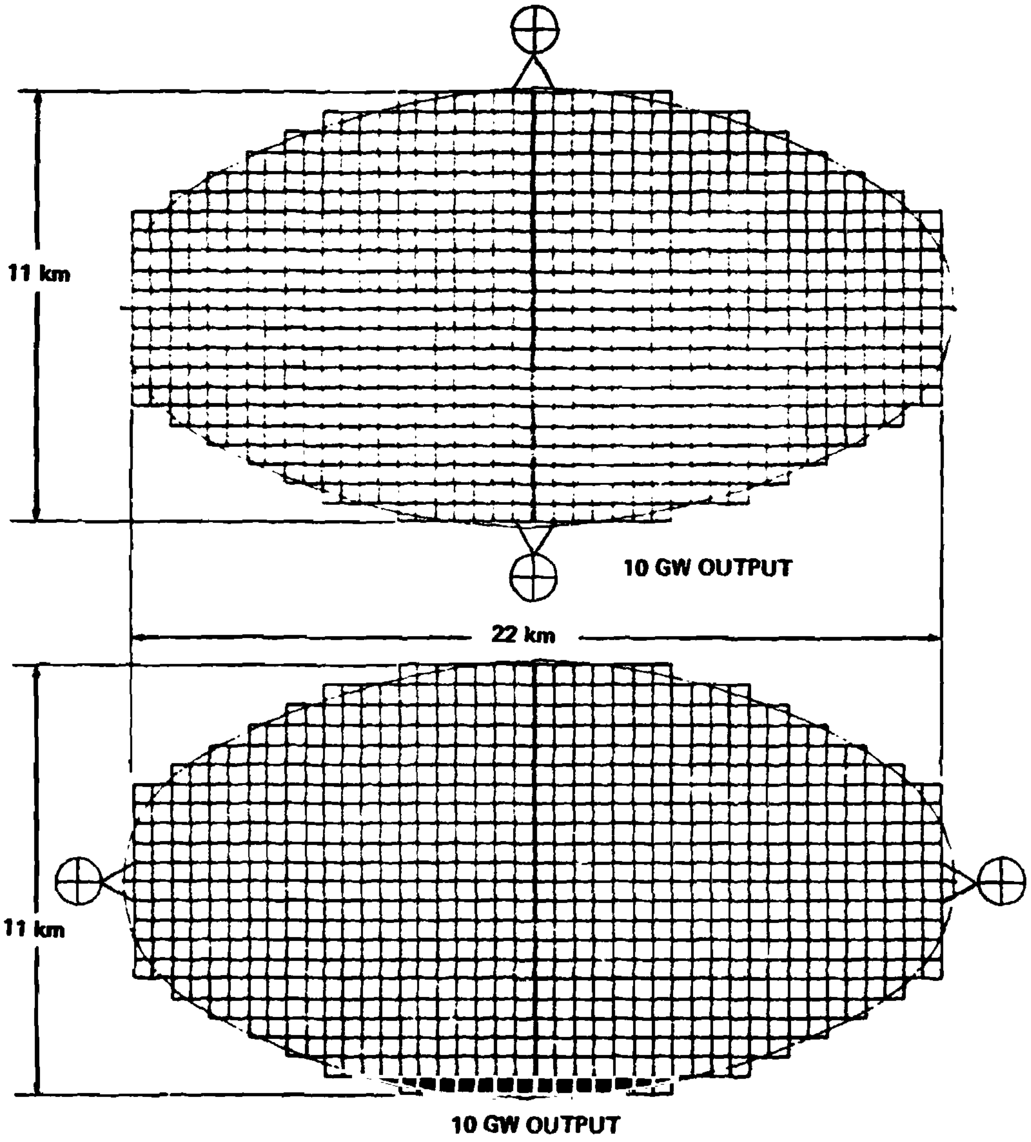


Figure 5-2. Solar photovoltaic SPS configuration options.

that minimizes thermal losses and focuses incoming solar rays to appropriate surfaces for absorption; and a thermionic conversion system that operates at high temperature (approximately 1815°C) in combination with a Brayton cycle conversion system that operates at low temperature (approximately 1038°C). Overall dimensions of this configuration for production of 10 GW of power are

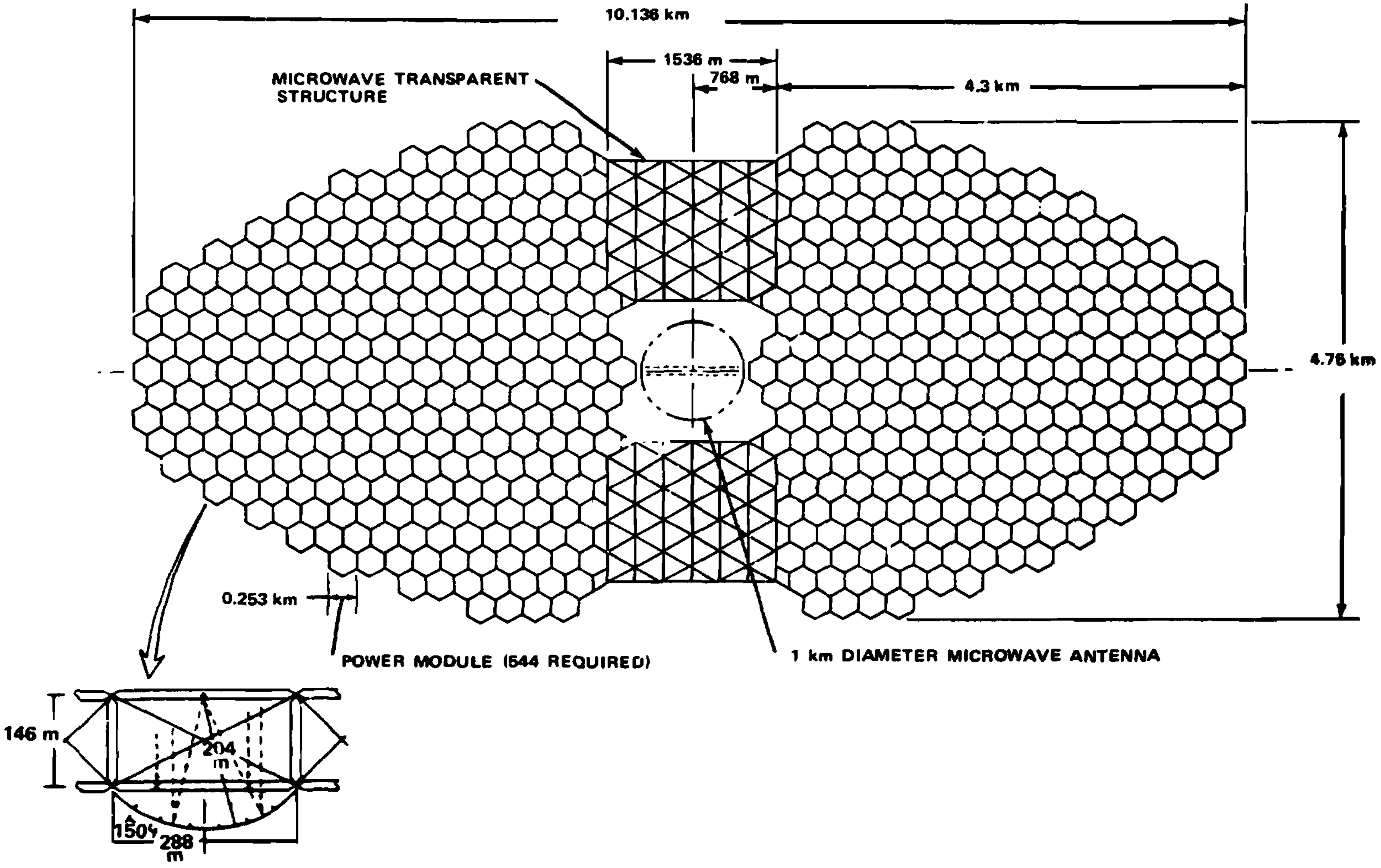


Figure 5-3. Solar thermal concentrator concept assembly.

approximately 5 × 10 km for an elliptical arrangement of the power modules and a center location of the single antenna. The profile of an individual concentrator module is shown as a detail of Figure 5-3. The total area of the concentrators is 322 km^2, the output per module is 540 W/m^2, and the total estimated mass is 221 × 10^6 kg. The orientation of the configuration is perpendicular to the Sun. Pertinent descriptive data for this configuration are presented in Table 5-2.

An alternate configuration concept, which features end locations for the antennas, is shown in Figure 5-4. A trade study has been conducted to identify the configuration with an optimum number of modules for minimum mass and to determine the cost of the operational system. The results of this study show the optimum number of modules to be approximately 40.

5.3 NUCLEAR

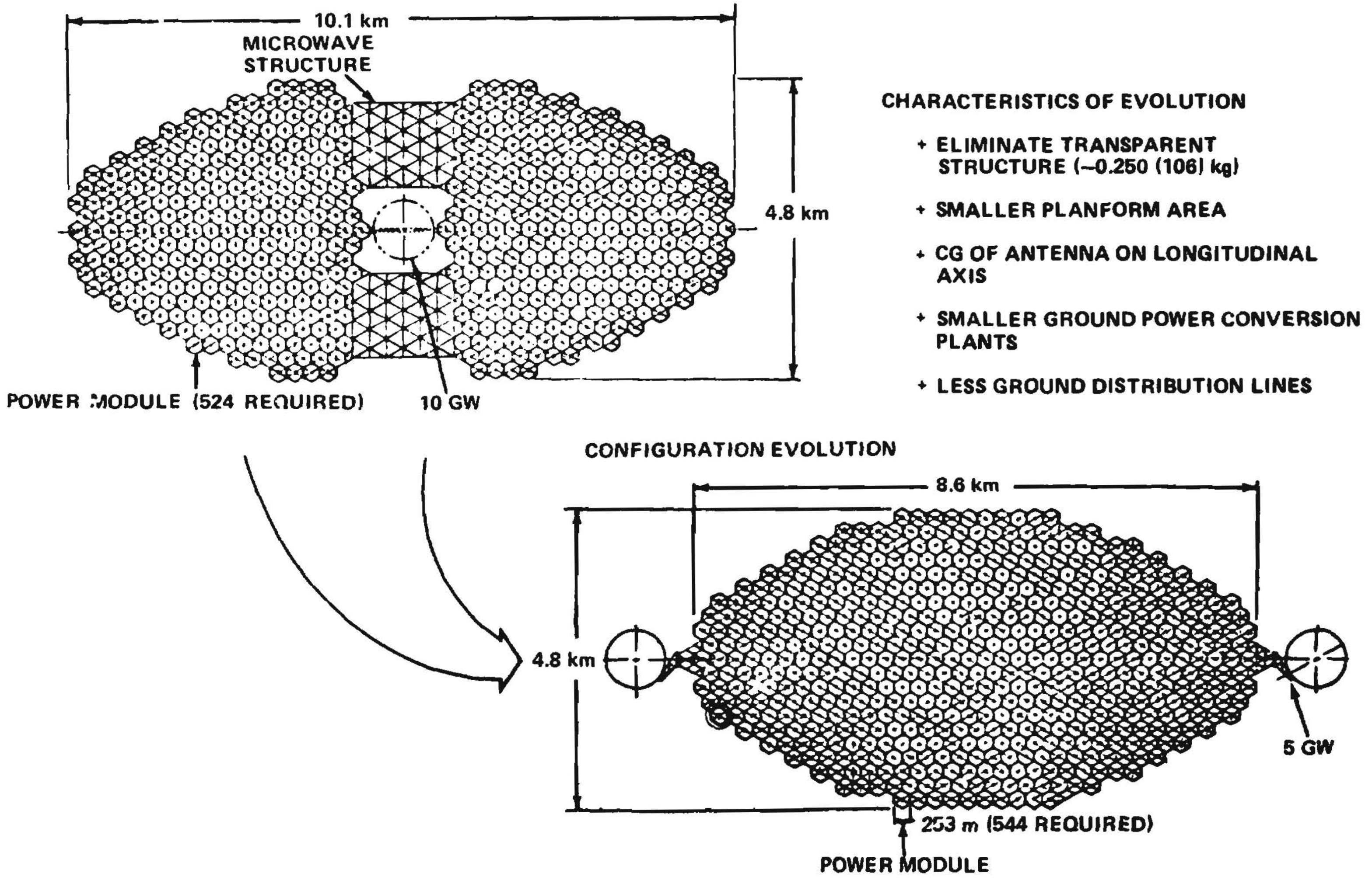
A typical concept of the nuclear Brayton configuration is shown in Figure 5-5. In this configuration an antenna 1 km in diameter is located between the nuclear modules and their required radiator area. A simplifying feature of the nuclear system is that occultations in orbit do not interrupt system operation. However, radiation problems make interfaces with this system complex. Also, an estimate of the mass of the nuclear system shows it to be quite heavy, approximately 300 × 10^6 kg. A schematic of the nuclear Brayton cycle system is shown in Figure 5-6. The concept shown utilizes a molten salt breeder reactor with continuous fuel reprocessing. The temperature limit of the molten salt limits the turbine inlet temperature and the operating efficiency of the Brayton cycle, which, in turn, affects the overall system size. The current trend is, therefore, to use a system with a gaseous fuel reprocessing system to relieve the above temperature limit constraint.

5.4 CONCEPT COMPARISON

Data comparing the photovoltaic, thermal concentrator, and nuclear concepts for satellite power are shown in Table 5-3. The nuclear system, having the smaller dimensions, is the more compact of the three systems. The mass estimate for the photovoltaic system, however, shows this system to be the lighter of the three.

TABLE 5-2. SUMMARY CHARACTERISTICS OF THE SOLAR THERMAL CONCENTRATOR SPS

Basic Characteristics		Subsystem Weights		
Subsystem	Current Concept Characteristics	Subsystem/Component	Mass (kg × 10^6)	
Energy Collection and Conversion		Energy Collection and Conversion		144.30
Concentration Ratio	2000	Concentrators	3.09	
Concentrator Total Area	32.2 km^2	Support Structure	8.90	
Concentrator/Radiator (Ratio of Area)	3.7/1	Thermionic Diodes	10.85	
Watts Output/Module Area	540 W/m^2	Turbomachinery/Generator	51.88	
		Absorber	2.17	
Power Distribution	AC			
		Radiators	60.70	
Microwave Antenna (Two)		Lines (Fluid)	2.77	
Diameter	1 km	Fluid	3.45	
Taper	9 dB	Motor Pumps (Fluid)	0.49	
DC-RF Conversion	Amplitron			
		Power Transmission		10.41
Configuration				
		Microwave Antenna/Subsystem		14.77
Shape	Elliptical			
Length/Width	2.0	Attitude Control		0.78
Microwave Antenna Location	End	Actuators	0.31	
Number of Modules	544	Propellants/Year	0.47	
Structure		Subtotal		170.26
Collector Module	Hexagonal	Contingency (30%)		51.08
Antenna	Hexagonal			
		Total System		221.34
Attitude Orientation	Perpendicular to Sun			



CHARACTERISTICS OF EVOLUTION

- ELIMINATE TRANSPARENT STRUCTURE (~0.250 (106) kg)
- SMALLER PLANFORM AREA
- CG OF ANTENNA ON LONGITUDINAL AXIS
- SMALLER GROUND POWER CONVERSION PLANTS
- LESS GROUND DISTRIBUTION LINES

Figure 5-4. Solar concentrator concept evolution.

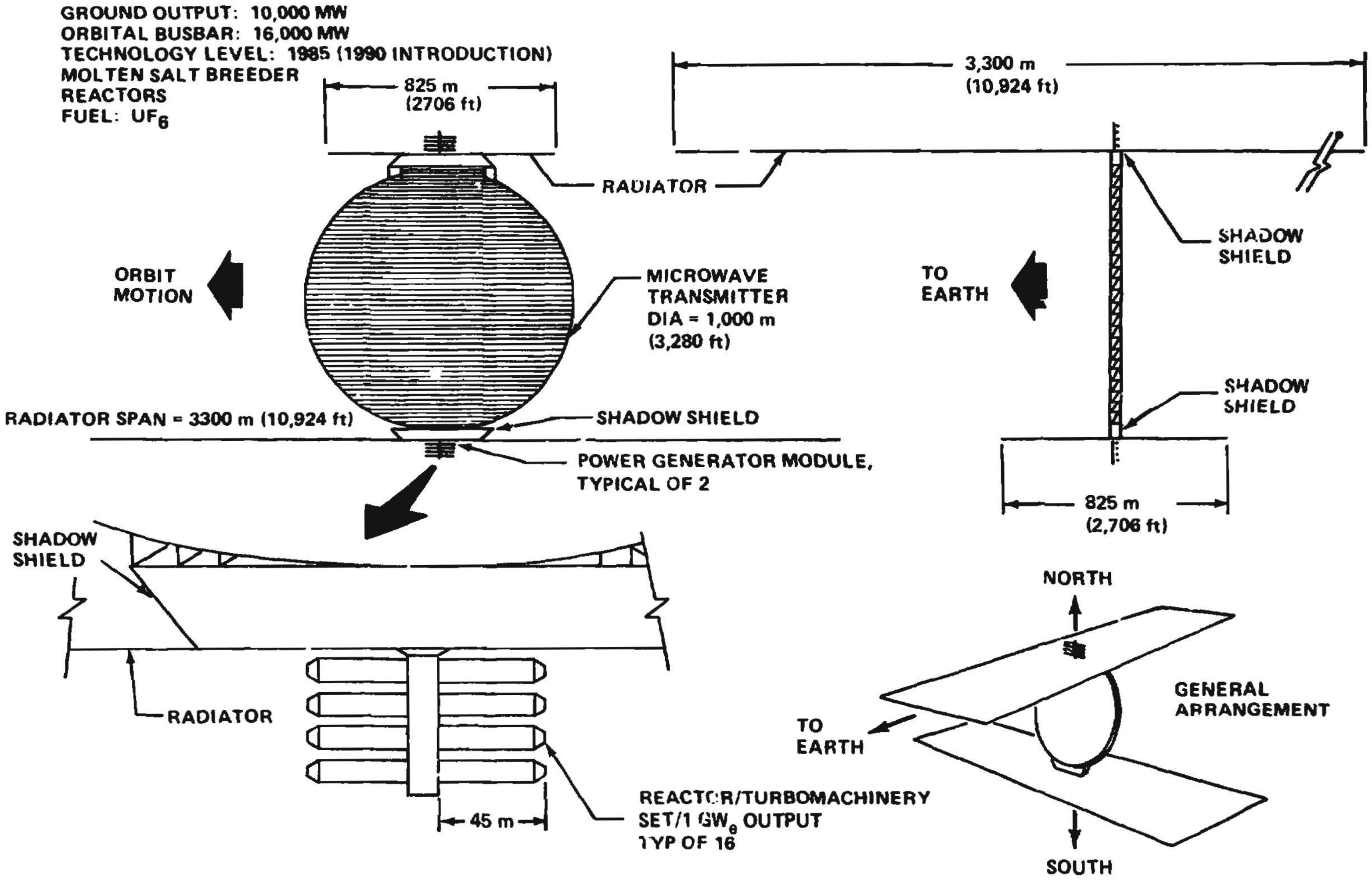


Figure 5-5. Nuclear thermal SPS configuration.

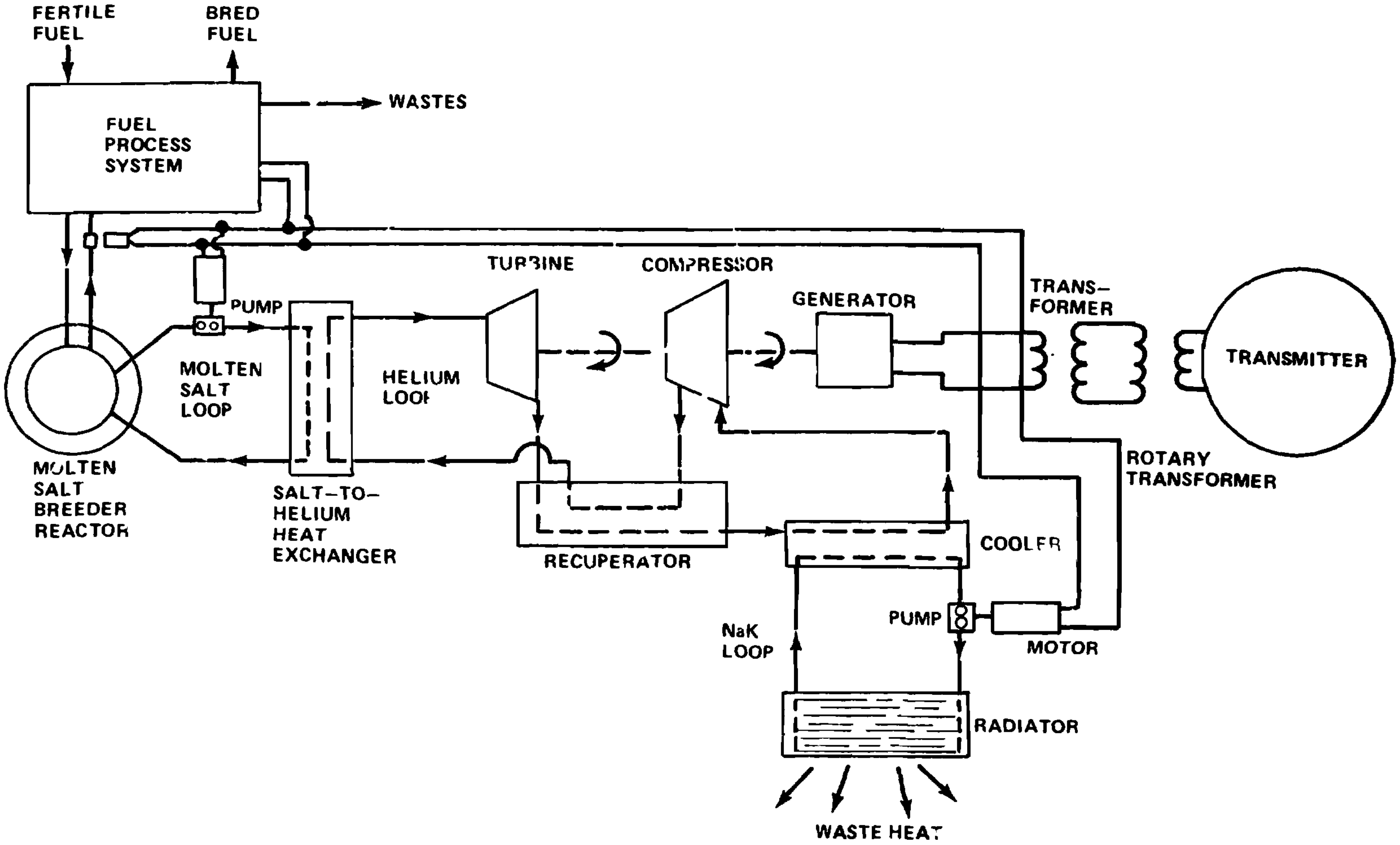


Figure 5-6. Nuclear Brayton cycle SPS schematic.

TABLE 5-3. SUMMARY OF REFERENCE BASELINES

	Photovoltaic	Solar Thermal	Nuclear
Configuration	Elliptical	Elliptical	Cross
Dimensions	~ 11 × 22 km	~ 5 × 9 km	~ 6 × 6 km
Size	~ 120 × 10^6 kg	~ 220 × 10^6 kg	~ 300 × 10^6 kg
Efficiency			
Microwave Antenna (Two)			
Diameter	1 km	1 km	1 km
Material Structure	Aluminum Composite	Aluminum Composite	Aluminum Composite
DC-RF Conversion	Amplitron	Amplitron	Amplitron
Location	End (Minor Axis)	End (Major Axis)	Center of Gravity
Attitude Orientation	Perpendicular to Sun	Perpendicular to Sun	MW Antenna and Meteoroid Avoidance
Concentration Ratio	2	2000	—
Power Distribution	DC	AC	AC
Cooling	Passive	Active	Active

6.0 DEVELOPMENT OF SPS TRADEOFF AND EVALUATION CRITERIA AND SIMULATION PROGRAMS

6.1 SPS TRADEOFF AND EVALUATION CRITERIA

Most of the trade studies that have been conducted to date were performed to obtain an understanding of the advantages and disadvantages of various schemes rather than to eliminate options. The results of many of these studies have indicated very slight differences in one approach versus another, and with further study and increased understanding, some conclusions could actually change. In most cases where one scheme appears better, the less favorable options were not eliminated from consideration but less effort may have been devoted to their further study.

Most tradeoffs were accomplished to minimize the mass and cost of the SPS program. However, there are some limiting constraints that are not as tangible as mass and cost that had to be considered. Many environmental issues had to be considered, such as operational environment, microwave beam, vehicle emissions, and other terrestrial impacts resulting from the nature and magnitude of the SPS program. For example, the power density level at which microwave beam-ionosphere interactions are expected to occur was used as an upper limit with a resulting effect on the size of the microwave transmission system.

Material properties, such as thermal limitations on the microwave structure were considered. The lifetime and reliability of materials and equipment were important parameters that had to be studied. Other factors such as the outage time for maintenance on turbomachines were considered in the tradeoffs. Factors such as program risks because of uncertainties in technological forecast were examined. The availability of resources was considered also. Tungsten was eliminated as a choice for radiators in the solar thermal concept because of the limited reserves of tungsten. Safety was also an important item for consideration.

Finally, comparisons were made of the various SPS concepts such as photovoltaic, thermal, and nuclear; and, of course, the SPS program itself was compared to alternate conventional and future power systems.

6.2 SIMULATION PROGRAMS

6.2.1 SYSTEMS DESIGN ANALYSIS MODELS

System design analysis models have been developed for both the photovoltaic and thermal SPS concepts for use in trade studies, sensitivity analyses, and to assure consistent and compatible designs. This has been accomplished through the use of integrated computer models that simulate not only the key design parameters but also their interactions.

To date most of the effort for the photovoltaic SPS has been devoted to the solar array portion, since this is the most massive and costly element of the SPS. The subsystems and related elements simulated within the model are the solar array including structures and power distribution, the microwave antenna, the rotary joint, attitude control, the rectenna, and space transportation. Figure 6-1 gives an overview of elements included in the model.

The primary outputs of the photovoltaic SPS model are total mass and unit cost. Secondary outputs include detailed mass and cost statements; blanket, reflector, and planform areas; number of HLLV launches; etc. Some of the design parameters included are: concentration ratio, temperature effects including passive radiator concepts, cell and reflector characteristics, and power distribution mast/feeder line mass and efficiency losses. Since all the major elements of the SPS are included, a more optimum overall SPS design can be established. For example, from an attitude control standpoint, it is desirable for the SPS to be long and slender when oriented perpendicular to the orbital plane. However, from a power distribution standpoint, a circular concept is preferred. With this model a concept can be chosen that gives the minimum total mass or cost. This model allows the study of variations in solar cell and reflector characteristics, concentration concepts, planform configuration shapes (e.g , rectangular, diamond, or elliptical), the impact of center-mounted versus end-mounted antennas, the impact on the number of HLLV launches based on the mass of the different segments of the SPS, payload density, mass and volumetric efficiency, HLLV payload capability, shroud size, and many other elements. Trade studies that have been accomplished to date include solar array orientation trades (perpendicular to orbit plane versus perpendicular to Sun), concentration ratios, power distribution efficiency, and planform length/width trades. Sensitivity studies have been conducted which include variations in the SPS efficiency chain from the solar cell to the ground power interface, power distribution voltage and efficiency, variations in the solar cell, and reflector characteristics. Results of these and other trade and sensitivity studies are contained within this report and were used in deriving some of the specific observations included in Section 11.0, Summary Baseline Definition.

Planned future studies include the investigation of alternate concentration techniques, variations in planform configurations, variations in SPS output power levels, different power distribution and structural schemes, the impact of space fabricated versus collapsible structures, etc.

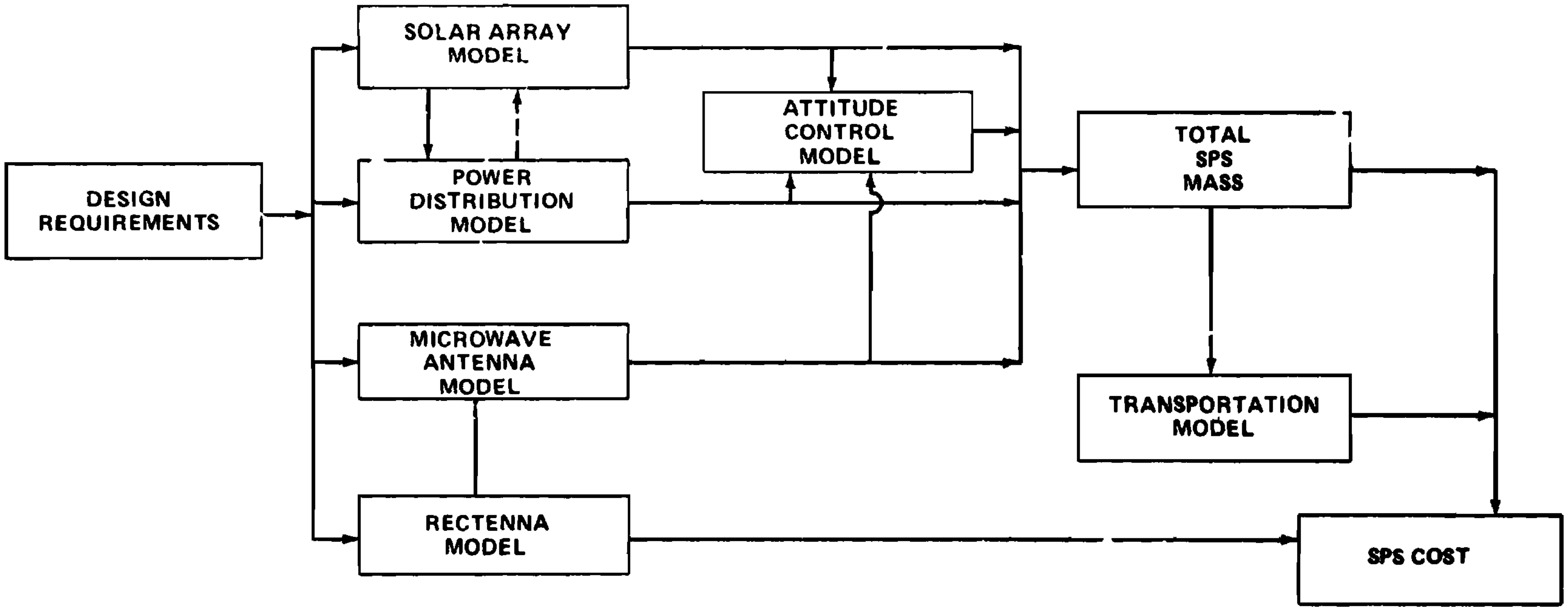


Figure 6-1. Systems design analysis model.

To date the thermal model is not as highly developed as the photovoltaic model but has been used for preliminary analysis in determining the optimum number of concentrator modules.

The development of these models began in-house in April 1976 and modifications and improvements have continued since. As promising ideas appear they are incorporated into the models so that their overall impact on the SPS can be evaluated. Current plans are to continue in this activity so that better SPS designs can evolve.

6.2.2 TRANSPORTATION MODEL

The transportation model computes flight rates, inventories, and engine and vehicle buy dates per year for the support vehicles of a given SPS with a designated assembly location.

If the SPS is assembled in low Earth orbit (LEO), the COTV only transfers the logistics (attitude control propellant and spare parts) needed to resupply the SP [illegible] are on-line. If the SPS is assembled in geosynchronous Earth orbit (G [illegible] h logistics and construction materials must be transferred. The model comp [illegible] the COTV construction and logistics flights per year based on the quantity of material to be transferred and the payload of the vehicle. This flight rate combined with the propellant per flight, engine life, time to change engine, checkout time, and trip time yields the propellant, engine buy, and inventory required per year to support the transfer of cargo from LEO to GEO.

The POTV flights per year, engine life, and propellant per flight are inputs to the model and yield POTV engine buys and propellant required per year to support the transfer of personnel from LEO to GEO.

The HLLV flights can be divided into five categories, i.e., construction, logistics, orbital transfer vehicle (OTV) engine, OTV propellant, and OTV delivery flights. Construction and logistics flights per year are based on the HLLV payload and the construction or logistics material required. The OTV engine flights per year are based on the COTV and POTV engine buys for that year along with the mass of each and the HLLV payload. The OTV propellant flights per year are based on the number of COTV and POTV flights for that year and the propellant required for each along with the HLLV payload. The OTV delivery flights per year are based on the required COTV and POTV inventories for that year along with the engine and stage weights for each and the HLLV payload. The HLLV inventory is based on the total flights per year and the average turnaround time between flights. Vehicles expire after 10 years or 500 flights, whichever comes first, and buy dates are computed on this basis.

The shuttle flights per year required for crew rotation and the vehicle inventory are inputs to the model. These inputs together with a 10 year, or 500 flight, life are used to compute the shuttle buys per year.

An information flow diagram of the model is given in Figure 6-2. Sample results for the photovoltaic and thermal SPS's assembled in LEO or GEO are given in subsection 9.2.

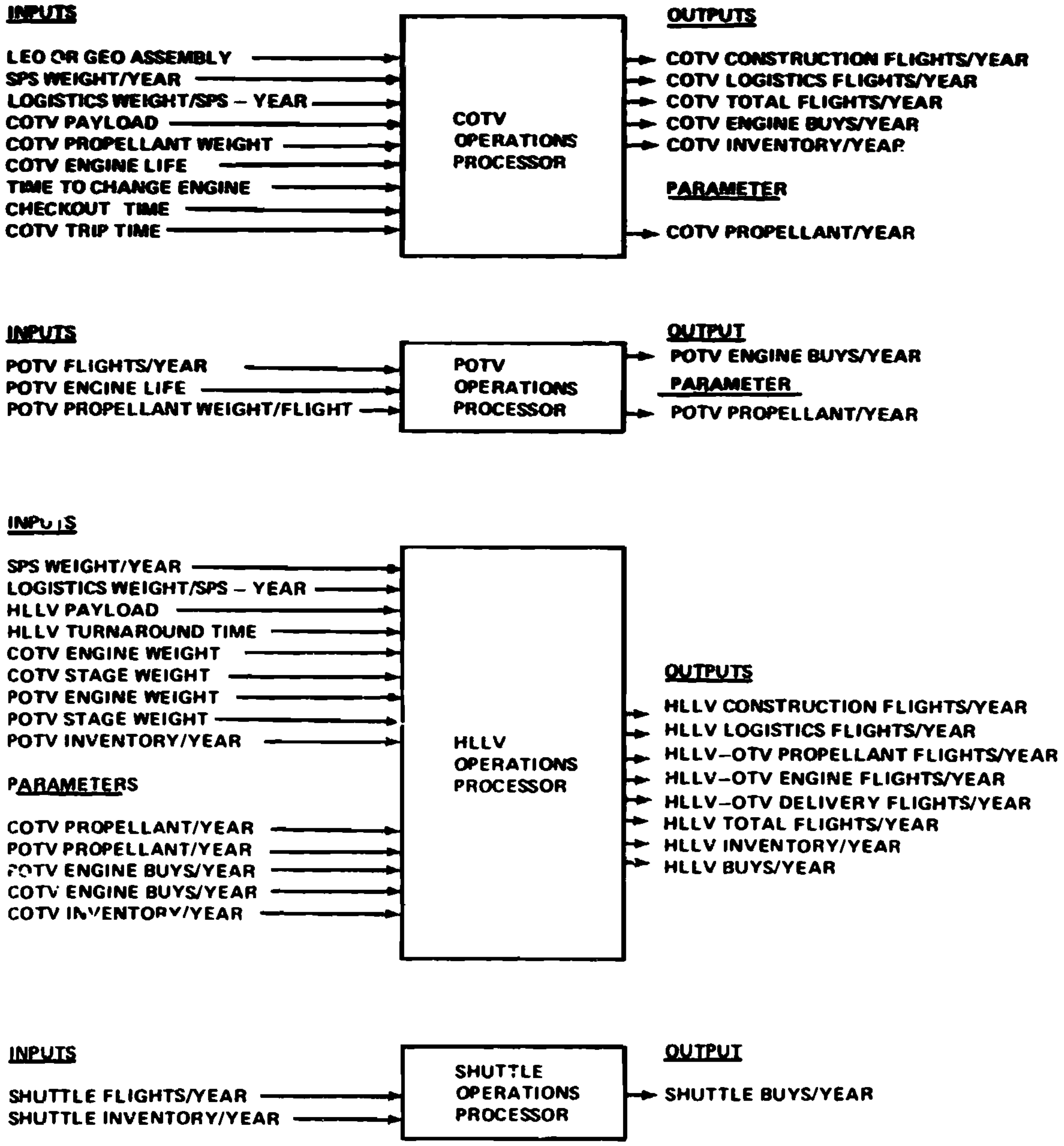


Figure 6-2. SPS transportation model.

7.0 SATELLITE POWER STATION

7.1 PHOTOVOLTAIC POWER CONVERSION SYSTEM

7.1.1 REQUIREMENTS AND ANALYSIS

The SPS solar array is required to produce sufficient power output to deliver 10 GW to a ground station designated to receive, condition, and transmit the power to the utility interface. This photovoltaic system design allows for typical solar array degradation and will provide the required output power without maintenance for 5 years. After 5 years, the array output will be maintained by repair and maintenance operations as the need arises. The system shall be capable of satisfactory performance for at least 30 years.

The overall efficiency of the photovoltaic concept, excluding solar cells, is currently calculated to be 52 percent; this efficiency chain is shown in Figure 7-1. The solar cell output power of 19.4 GW encounters losses in the electrical connections, the power conductors, the rotary joints, and in the operating power to the SPS subsystems. The microwave system receives 17.3 GW for conversion and transmission. After reception and dc-RF conversion, 10 GW at low voltage is available from the dc power grid at the ground site. The dashed block of Figure 7-1 shows unconfirmed ground system losses estimated as discussed in section 8.1. The low conversion efficiency of photovoltaics makes the solar array size highly sensitive to all system losses.

Distribution of large quantities of power across the spacecraft is inherent to the SPS and indicates that high voltage distribution is desirable to minimize losses in conductors. Selection of distribution voltage has little effect on array configuration, although losses cause increases to the solar array area required. Power distribution and control are discussed in subsection 7.1.4.

To evolve the appropriate geometric solar array configuration, an early decision must be made on the concentration ratio of the solar energy collection scheme. Integral to the concentration ratio selection process is the consideration of how structural weight, solar blanket weight, and, ultimately, the cost can be minimized.

Although the results are not conclusive for all concentration ratio geometries, some trades have been completed (Fig. 7-2) that indicate the concentration ratio should be near two for the geometry considered in the study thus far.

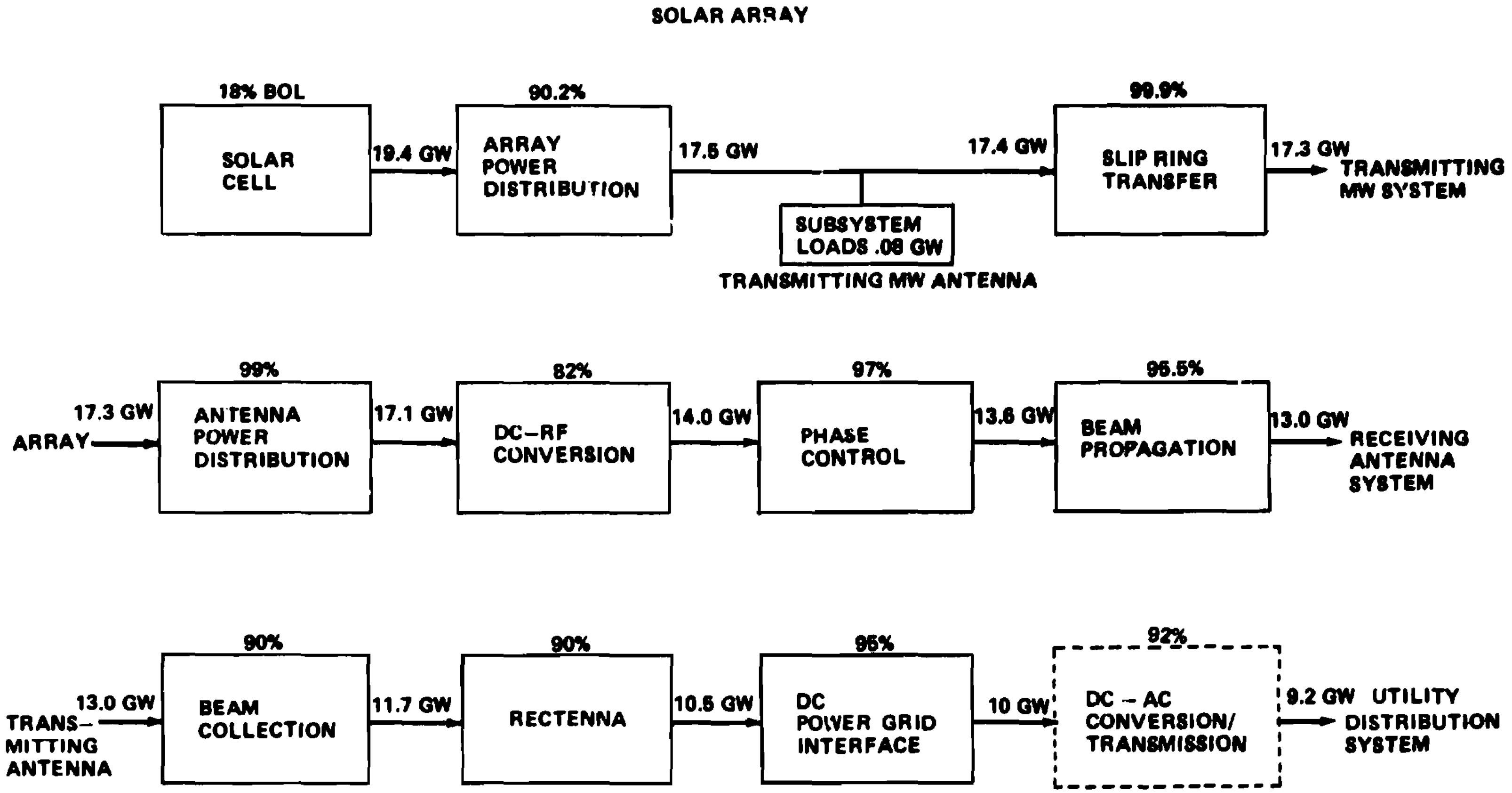


Figure 7-1. Electrical efficiency chain.

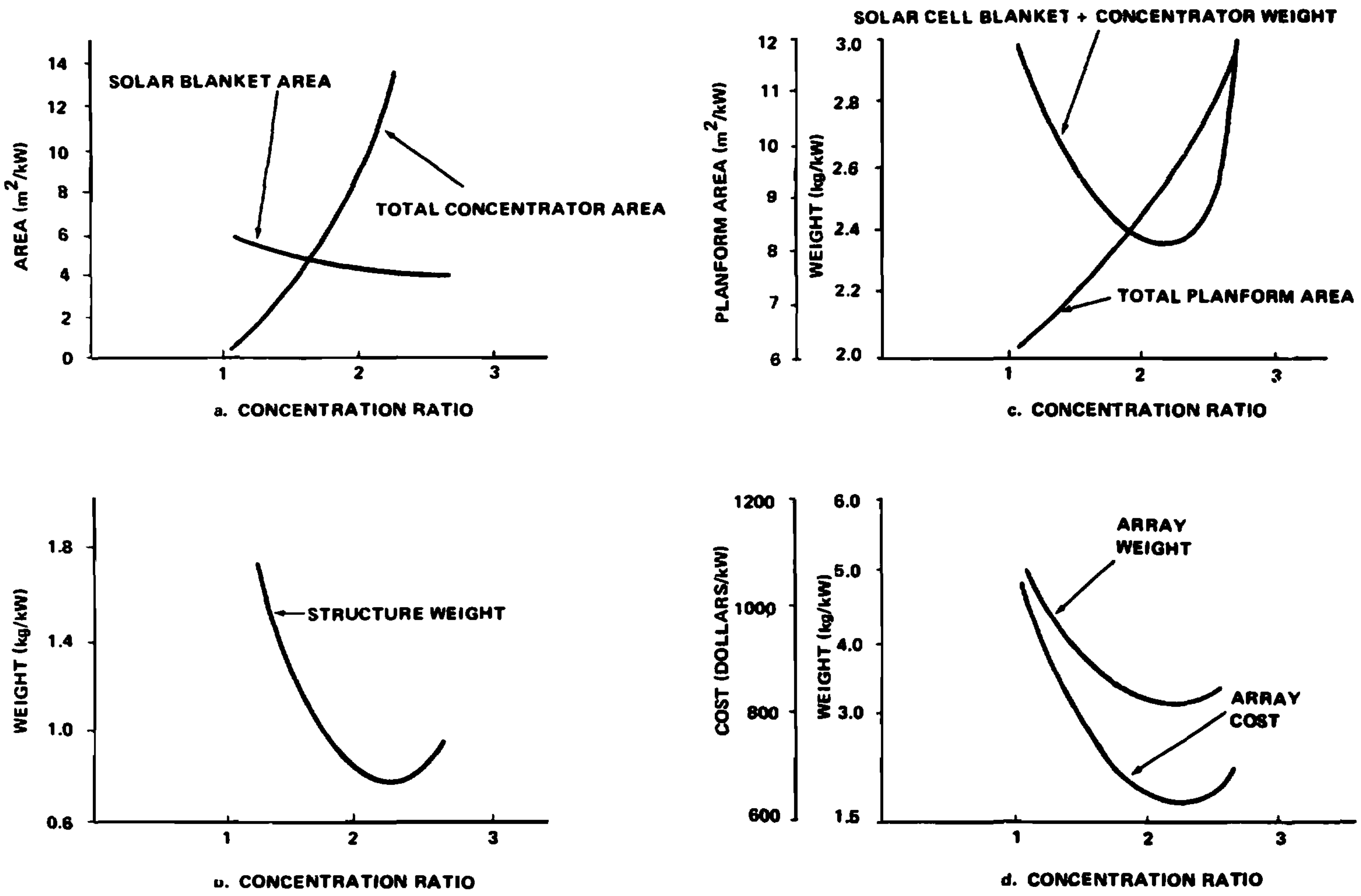


Figure 7-2. Concentration ratio trades.

Characteristics of various photovoltaic systems were studied to select a solar cell technology suitable for the SPS. The rapid progress in this technology makes assessment of practical performance goals difficult. The work reported here has assumed the availability and utilization of high efficiency low cost silicon solar cells, although competing technologies show promise. The rationale for this choice is the availability of user experience, the broad industrial basis for semiconductor silicon, and the known availability of silicon. Table 7-1 summarizes candidate photovoltaic device progress up to mid-1976.

The design and application of low mass solar arrays has been a continuous problem in spacecraft design. The most common solutions have been the mounting of solar cells on the spacecraft or on special rigid panels, and these represent methods that are probably not feasible for the SPS. Recent requirements for large lightweight solar arrays for space station and the solar electric propulsion stage (SEPS) have resulted in a substantial body of useful lightweight array engineering data. The use of plastic film materials has received a great deal of effort, and products of this technology have been flown in small solar arrays. The largest, highest performance array presently under construction is that for the SEPS, rated at 25 kW, which is the latest power collection technology from which to evolve the SPS. The construction of the SEPS array has been used extensively to derive the SPS design reported herein.

The potential for increasing the output of solar cells by sunlight concentration is subject to solar cell performance constraints, e.g., the loss of conversion efficiency at elevated temperatures and illumination levels. The aspects of maintaining reasonable cell temperatures at low levels with moderate concentration (less than 10 Suns) were investigated. The literature concerning photovoltaic devices under increased illumination was reviewed, and it was concluded that silicon technology was satisfactory for the less than 10 Sun level. Several concepts for cooling the solar cells were studied, and one of the more promising concepts is depicted in Figure 7-3. This concept may be capable of a savings of 34 percent in mass and 22 percent in planform area, and a 40°C temperature reduction at the solar cell as compared to the original configuration utilized in the SPS study. These types of configuration changes will be factored into the next phase of study activity. The integrated reflector/radiator concentrates sunlight on the thin narrow solar cell assembly, which is thermally coupled to the reflector radiator.

7.1.2 SOLAR ARRAY SUBSYSTEM

The SPS solar cell blanket/reflector concept used for this study is illustrated in Figure 7-4(a) and considers the structure as an integrated part of

TABLE 7-1. EFFICIENCIES OF PRESENT SOLAR CELLS

Cell Type	Efficiency (%)	Comments
Single Crystal Silicon	13 to 15	Commercially available; experimental devices reported with higher efficiency; estimated practical efficiency is less than 21%; 17 to 19% is probably more realistic; has history of reliability and is in wide use.
Polycrystalline Silicon		Laboratory devices only.
Large Grain Size	7 to 8	3 to 4 mm diameter grains.
Small Grain Size	4	3×10^{-4} to 5×10^{-4} mm diameter grains.
Copper Sulfide/Cadmium Sulfide	6 to 7	Commercially available; polycrystalline; some estimates indicate 10% is the practical maximum but others have higher hopes; commercial units are 3 to 4%, and French workers report 7%; reliability is reported to be low (probably due to humidity and process).
Gallium Arsenide		Laboratory devices only at present.
Single Crystal	14 to 15	Maximum practical efficiency approximately 21%.
Polycrystalline	—	No data available.
Cadmium Telluride	5 to 6	Laboratory devices only; mostly European work (France and Germany).
Cadmium Sulfide/ Indium Phosphide	12	Laboratory devices only; single crystal.
Schottky Barrier Types		Laboratory devices.
Metal Oxide/GaAs	13 to 10	
Other Types		

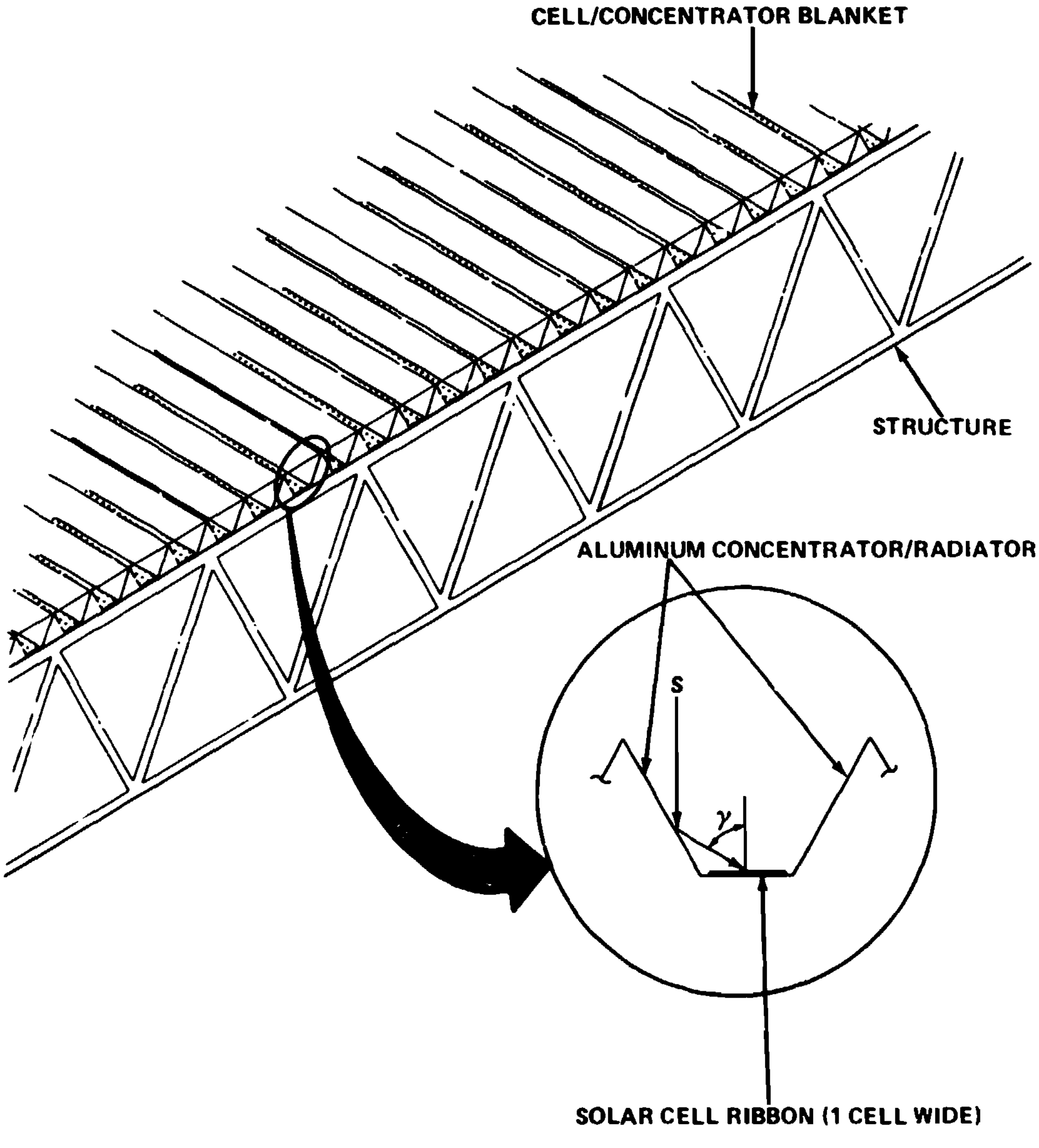
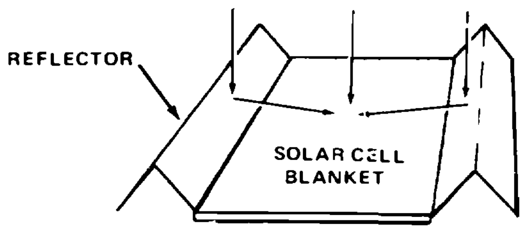
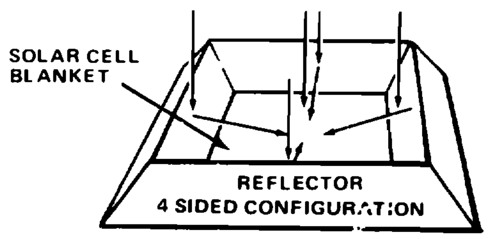


Figure 7-3. Trough configuration with concentrator radiators.

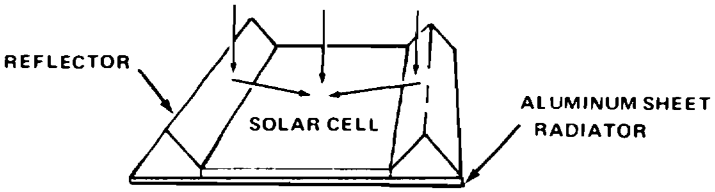
the concept to develop the concentration ratio. It is a simple flexible solar cell blanket composed of solar cells mounted on thin plastic film stretched across a lightweight structure. The structure also supports a prismatic surface of metalized plastic film to serve as a concentrator (concentration ratio of two) for the solar cell blanket.



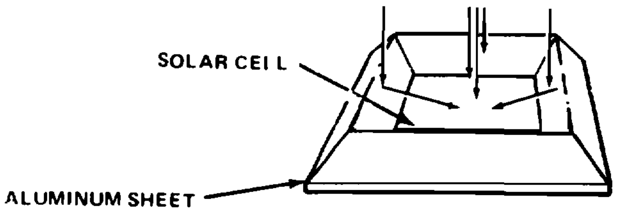
a CONCENTRATION RATIO ~ 2–3



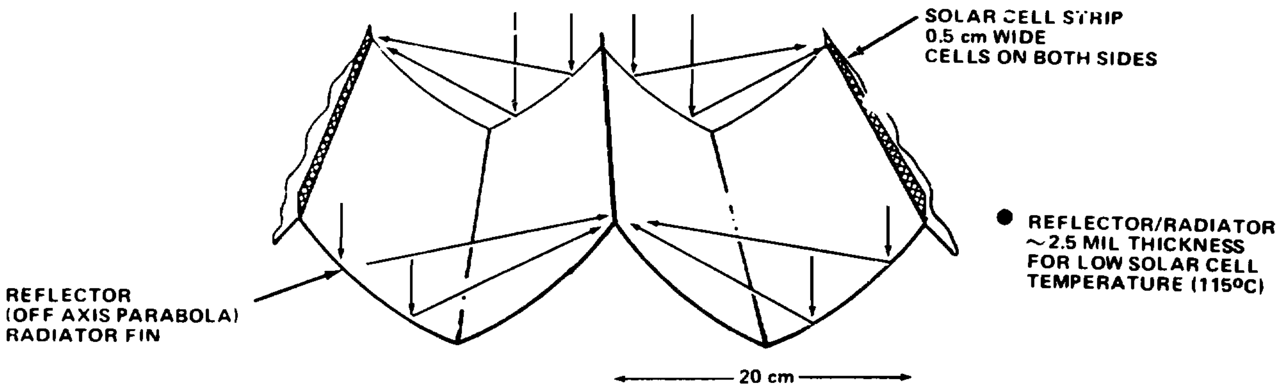
b. CONCENTRATION RATIO ~ 2–3



c. CONCENTRATION RATIO ~ 2–5



d. CONCENTRATION RATIO ~ 3–4



e. CONCENTRATION RATIO ~ 10:1

Figure 7-4. Reflector/solar cell configuration options.

Some of the more prominent features of the solar cells used in this study are: 18 percent efficiency cells, and very thin cell covers and substrates. More detailed features of the SEPS and SPS solar array characteristics are given in Table 7-2. The selected solar cells are similar to those in present use upgraded to higher performance. The characteristics assumed for the solar cells seem reasonable for advanced silicon solar cells.

While the concept of Figure 7-4(a) represents a conservative adequate design, other concepts (Fig. 7-4, b through d) are postulated that have inherent advantages of higher concentration ratios and efficiency as a function of better temperature control characteristics.

The performance features presented in Table 7-2 can be obtained by minor technology improvements — decreasing assembly losses, reducing the unused planform area, and improving thermal characteristics of the solar cells. Low radiation degradation in geosynchronous orbit and low thermal cycle degradation (because of the relatively small number of eclipses) help maintain the array performance. These performance figures take into account the effects of off-normal reflectance and absorption of sunlight, optical characteristics of the reflector and solar cells, variation in solar insolation due to Earth orbital eccentricity, and minimal solar radiation. Geosynchronous micrometeoroid damage is small. No allowance has been made for energy storage for eclipse operations.

Table 7-2 compares the characteristics of a typical present lightweight array design with those projected for the SPS, based on SEPS technology. The improvements lie almost entirely in the area of solar cell performance and the handling of the large solar cell blanket/concentrators, including the automated manufacture of these assemblies.

7.1.2.1 HIGH EFFICIENCY SOLAR CELLS

Most of the SPS study activity to date has concentrated on silicon cells of moderate performance, but a key to the success of SPS is the selection and development of a high efficiency solar cell technology (silicon or other). Many new types of cells are postulated, and some of these are shown in Figure 7-5. The vertical multijunction (VMJ) cell in Figure 7-5(a) supplies power at a high voltage dependent upon the number of junctions used in its manufacture. The use of high voltage power transmission is desirable to reduce conductor losses, but many problems remain before large quantities of VMJ cells of high performance can be assured. Thin film photovoltaics, typified by the cadmium sulfide (CdS) system, are shown in Figure 7-5(b). Thin film photovoltaics are

TABLE 7-2. SOLAR ARRAY CHARACTERISTICS

Characteristic	1976 SEPS	Projected for SPS
Solar Cell	N/P Silicon	N/P Silicon
Cell Size	2 × 4 × 0.02 cm	2 × 4 × 0.01 cm
Cell Cover	0.015 cm Silica	0.007 cm Silica
Cell Efficiency	11.4%	18%
Cell Effective Area	92%	97%
Assembly Losses	7.5%	5%
Temperature Derating	83.8% @ 55°C	91% @ 52°C
Cell Efficiency at Beginning of Life (BOL)	8.6%	15.1%
Cell Stacking on Array	79.4%	91.2%
Array Substrate	Kapton/FEP	Kapton/FEP
Substrate Thickness	0.005 cm (2 mil)	0.003 cm (1 mil)
Ultraviolet and Particulate Degradation	25%	10%
Array Weight	0.969 kg/m^2	0.513 kg/m^2
Array Power	74.2 W/m^2	167.6 W/m^2
Specific Power	76.6 W/m^2	326.5 W/m^2

Note: Tensioning structure and attachment hardware not included.

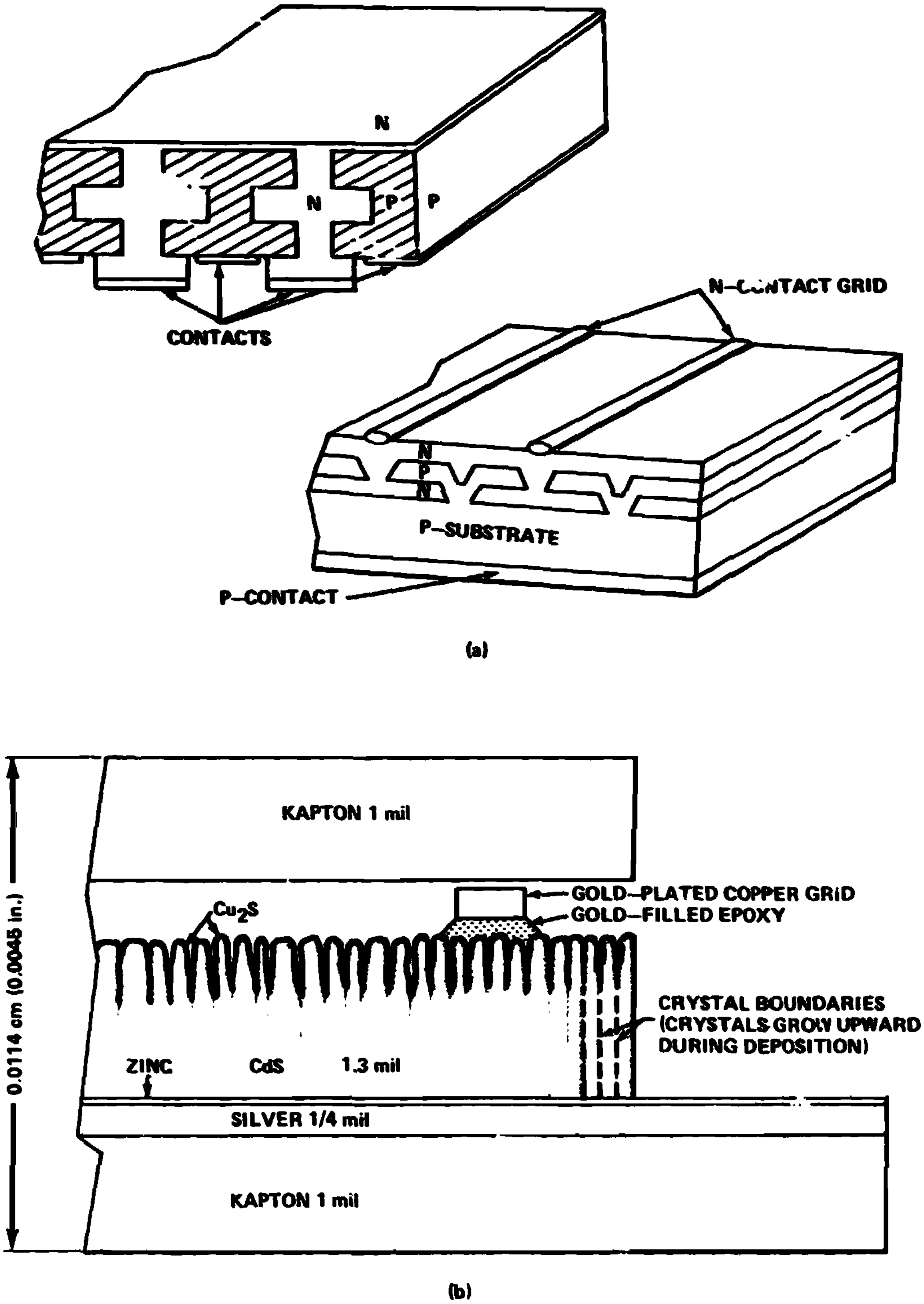
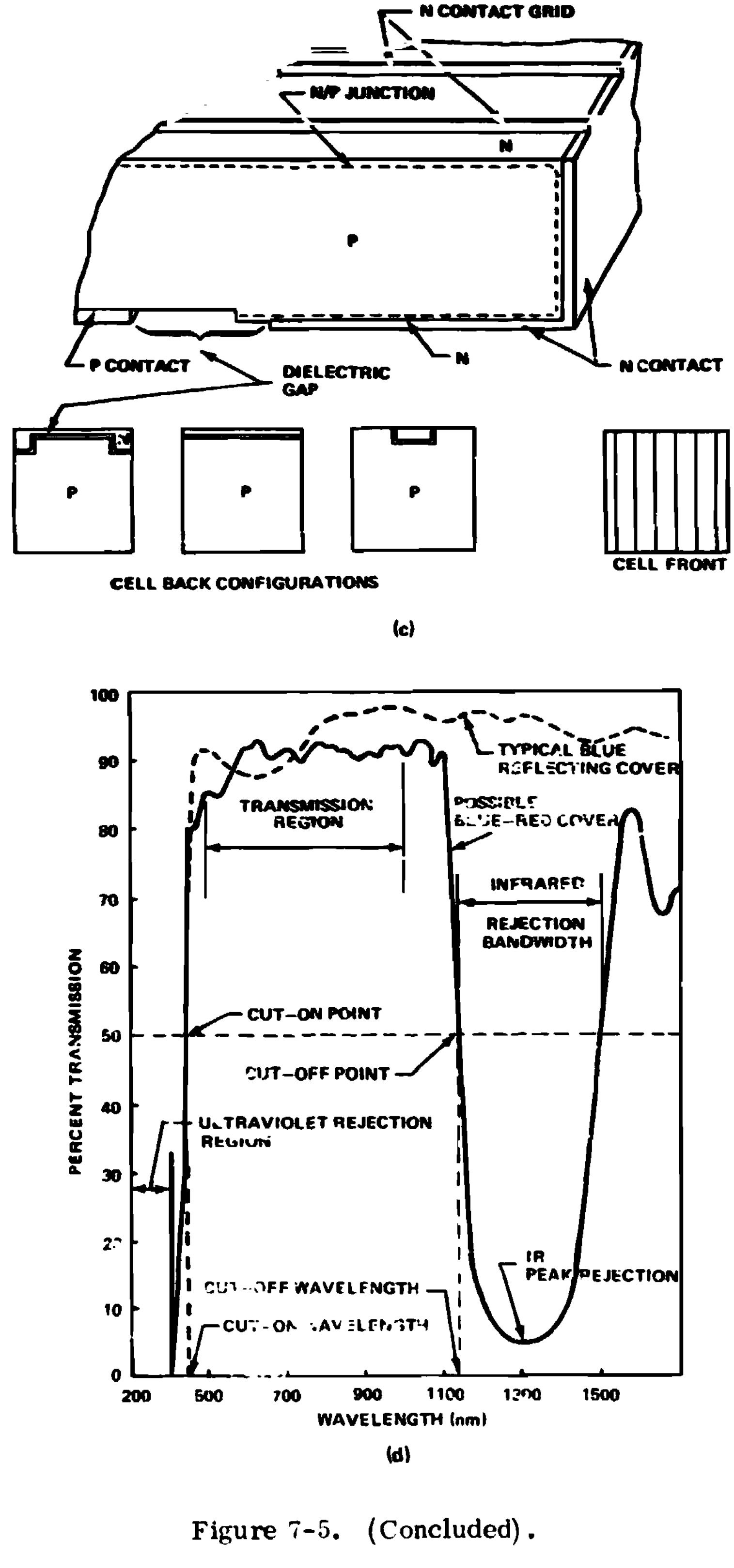


Figure 7-5. Solar cell description.



(c)

(d)

Figure 7-5. (Concluded).

typically manufactured by vacuum vapor deposition onto thin plastic films, but characteristically these obtain only low efficiency (up to perhaps 10 percent). Further study of various thin film systems may indicate advantages for their use on SPS due to the potential low cost, low weight, and simple construction.

Figure 7-5(c) depicts the solar cell selected for this study — a thin silicon cell with wraparound contacts, low area surface contacts, and some form of textured surface to improve optical performance. Cells typical of this design (constructed in COMSAT Corp. laboratories and called "nonreflective" or "black" cells), have recently shown conversion efficiencies in the 15 percent range. Further advances are expected in the future.

Solar cell performance is adversely affected by increased cell temperature. It is desirable to prevent that part of the solar spectrum unusable by solar cells from being absorbed in the cell and heating it. Figure 7-5(d) is a plot of the characteristics of an "optical bandpass" filter that performs this function. Only the spectral region capable of exciting the photovoltaic process is passed; ultraviolet and infrared radiation are absorbed or reflected by the filter. The filter may be integrated into the reflector surface or the solar cell and cover, or may be on all surfaces. Such filters are composed of fractional wavelength thick layers of various material applied by vacuum vapor deposition. An optical filter is sensitive to the direction of impinging light, but the SPS array pointing is adequate to prevent detuning of the filters. The manufacturing technology for optical filters is well known, being applied to virtually all optical surfaces on a commercial basis, but application to the large SPS surfaces will require study and development of automated (possibly in space) manufacturing techniques.

7.1.2.2 SOLAR ARRAY FLEXIBLE SUBSTRATES

The plastic composite substrate selected for the baseline SPS solar array is composed of Kapton, a polyimide film, and FEP fluoroethylene, such as Teflon. These materials have received extensive study for space applications, and studies of space station and electric propulsion arrays have selected them also. Present technology enables fabrication of printed circuitry on such substrates, much like the circuits needed for solar cell interconnection. Small arrays using this construction have been operated. For the SPS, a light low cost interconnection method is desirable. This indicates that welding methods may be preferable to the soldering methods most commonly used in the past. The HELIOS solar probe used an array with automatically welded interconnections, and other projects have used semiautomated or fully automated interconnections on a small scale. These techniques form a basis for the study and development of an SPS oriented assembly process.

Handling of the large, flimsy solar cell blanket and concentrator films is a problem that will require further study. These handling techniques are probably impractical under the Earth's gravity; therefore, they may require development and verification in Earth orbit.

7.1.2.3 KEY ISSUES

The particular photovoltaic system selected impacts the amount of solar concentration, the planform area, the structural design, the thermal design, and the manufacture and application of the solar array. Final selection of an SPS photovoltaic system will be dependent upon the results of research and development of the competing alternate photovoltaic systems and systems studies of the integration of these technologies into the SPS project. Table 7-3 summarizes some of the key issues in the selection and conceptual design of an SPS photovoltaic system.

SPS power distribution is a critical performance area. The power lost in transmission from photovoltaic device to the microwave conversion system is a substantial fraction of the total generated power, resulting in undesired and costly oversizing of the solar array. Transmission losses may be minimized by using high voltage transmission, but this poses inherent technology hurdles in handling and controlling the high voltage.

Other key issues, which need further study, relate to the power needed to support the upkeep and maintenance of the SPS; the power supply during eclipse of the SPS (not necessarily to maintain power transmission but to support station activity); and the power needed to maintain SPS attitude control, communication, and control within the SPS.

7.1.3 SOLAR ARRAY STRUCTURE SUBSYSTEM

7.1.3.1 PRIMARY STRUCTURE

The primary solar array structure consists of current carrying structure, non-current carrying structure, and dielectric structure. The current carrying structure utilizes the lateral beam trusses by proper beam element sizing and by electrically isolating the conducting lateral beams from the nonconducting longitudinal beams. Approximately 45 percent of the lateral beams must be increased in cross sectional area above that required for structural purposes to meet the current carrying requirements. This is accomplished by increasing the material thickness of the basic structural elements. The non-current carrying structure consists of longitudinal beam truss members and tensioning wires. The support structure between the arrays is referred to as the dielectric structure and is assumed to be fabricated from a fiberglass type material, such as an S-glass/epoxy composite, so that it is transparent to the microwave beam from the transmitting antenna.

TABLE 7-3. KEY ISSUES FOR SPS POWER GENERATION AND DISTRIBUTION

- Array Performance is Highly Sensitive to Many Parameters
  - Solar cell thermal coefficients and thermal properties
  - Solar cell cover or surface optical properties
  - Reflector (concentrator) physical properties
  - Radiation degradation performance of all components
  - Other environmental effects on all components
  - Thickness and size of basic cells
  - Output voltage and current (and possible requirement for voltage regulation)
  - Interconnection requirements and characteristics
  - Substrate characteristics
  - Off-Sun pointing (especially for concentrator versions)
  - Transportation and assembly of very fragile arrays
- SPS Power Distribution can be a Critical Performance Area
  - Power loss is substantial (ECON at 8 percent loss gives up 1.4 GW) which reflects into larger impacts on array and/or concentrator
  - High voltage distribution introduces additional penalties along with benefits
  - Long life of conductor structure at elevated temperatures
  - Rotary joints or contacts for power transfer from array to microwave system
- Power Required for the Control and Maintenance of the Station
  - Includes thrusters, communications, upkeep of systems, local transportation
- Ground Power Conversion, Conditioning, and Distribution
  - Rectenna output may require change to be compatible with distribution networks
  - Small changes on the ground can reflect into large changes on the SPS
  - If terrestrial energy storage is required, technology solutions must be considered

The structural cross section geometry utilized for the baseline solar cell array structure is shown in Figure 7-6 and is based upon the structural concept developed by the Grumman Aerospace Corp. The lateral truss geometry conforms to the required geometry of the reflectors and solar cell blanket, eliminating the need for secondary structural members. The length of the individual members provides adequate structural depth for bending stiffness. These characteristics are necessary to achieve very low structural planform area densities of 0.05 to 0.15 kg/m^2. This structural concept was incorporated into an elliptical planform to achieve minimum overall SPS weights and to accommodate power distribution through some structural members.

A listing of the primary characteristics and requirements of the solar array structural configuration is presented in Table 7-4. These characteristics and requirements are the primary drivers in determining and defining major structural parameters such as beam sizing, tensioning cable sizing, materials selection, etc. Structural assembly considerations such as beam end joining concepts and tensioning wire attachment concepts were excluded, since their influence on the size and mass of the total configuration is considered to be minimal.

A comparison of rectangular and elliptical planform shapes indicates a structural mass advantage for the elliptical shape. The longeron cross sectional area is an indicator of the size of the structural elements used to construct all major structural members and, hence, is an indicator of total structural mass. A comparison of the candidate shapes based on the first mode bending period is shown in Figure 7-7. For given stiffness sizing criteria, the comparison of structural mass of an elliptical planform to a rectangular planform, either of equal length or equal length-to-width ratio, is indicated. A systems comparison of the weight differences must also consider other mass drivers, such as shear and bending moment loads occurring during orbital transfer maneuvering and attitude control propellant requirements. Current comparisons indicate propellant mass advantages for the elliptical planform shape.

Further comparisons were made between the elliptical and rectangular planform configurations by evaluating the effects of orbital transfer propulsion systems in different locations. The location options considered in the study are:

1. A 66 720 N thrust transfer propulsion system located on the microwave antenna

2. Two 33 360 N thrust transfer propulsion systems located on the ends of the configuration

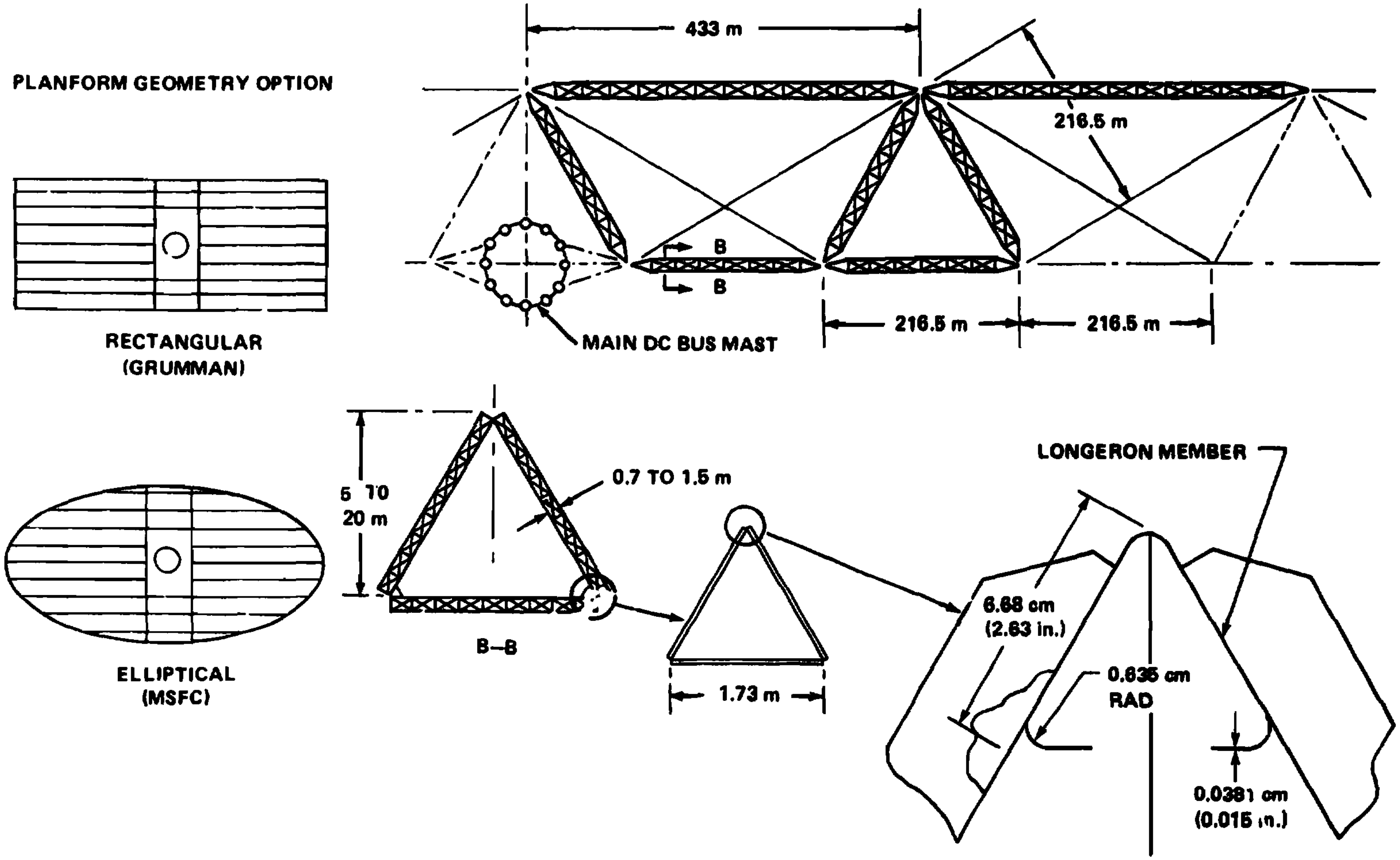


Figure 7-6. Solar array structural configuration.

TABLE 7-4. SOLAR ARRAY STRUCTURAL CONFIGURATION CHARACTERISTICS AND REQUIREMENTS

Minimum Structure Area Density 7 to 21 kg/m^2	● Widely Spaced Truss Members ● Minimum Secondary Structure for Reflector and Solar Blanket Support
High Bending Stiffness (First Mode Bending Frequency ≤ 2 cycles/h)	● Eliminate Excitation of Structural Modes from Attitude Control ● Minimize Degradation of Solar Orientation Because of Deflection
High Bending Moment Capability	● Longitudinal Member Strength Consistent with: — Gravity Gradient/Attitude Control Torques — Orbit Transfer Thrust Loads
Structural Member Utilization for Power Distribution Buses	● Lateral Member Minimum Cross Sectional Area Sized for Power Distribution Through Structure to Central Power Mast

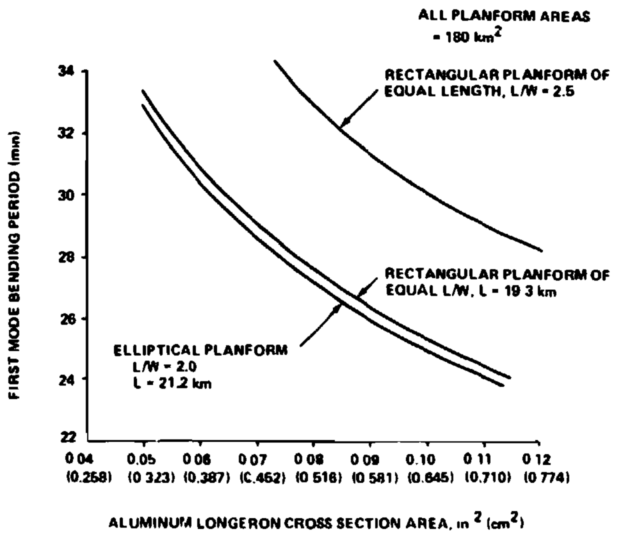


Figure 7-7. Comparison of rectangular and elliptical planform geometry.

3. Two 33 360 N thrust systems extended beyond the photovoltaic structure, a distance assumed to alleviate plume impingement complications.

Deflection and slope curve plots for the two planform configurations during orbital transfer thrusting for the three different propulsion concepts show the elliptical configuration to have the advantage over the rectangular configuration with the transfer thrusters antenna mounted. With the transfer thrusters end mounted, deflection and curve plots favor the rectangular configuration. However, as will be shown later, end mounted thrusters induce a much higher moment on a photovoltaic structure than do antenna mounted thrusters.

Sizing of shear stabilization cables was conducted by determining the change in structural stiffness, as indicated by the first mode bending period, with the change in cable cross sectional area, assuming the cable material to be steel. With increasing cross sectional area, the first mode bending period is asymptotic to a value where shear stiffness is infinite (Fig. 7-8). As indicated, this is achieved with very small cable sizes. A cable size of 0.065 cm^2 was used in all subsequent analyses.

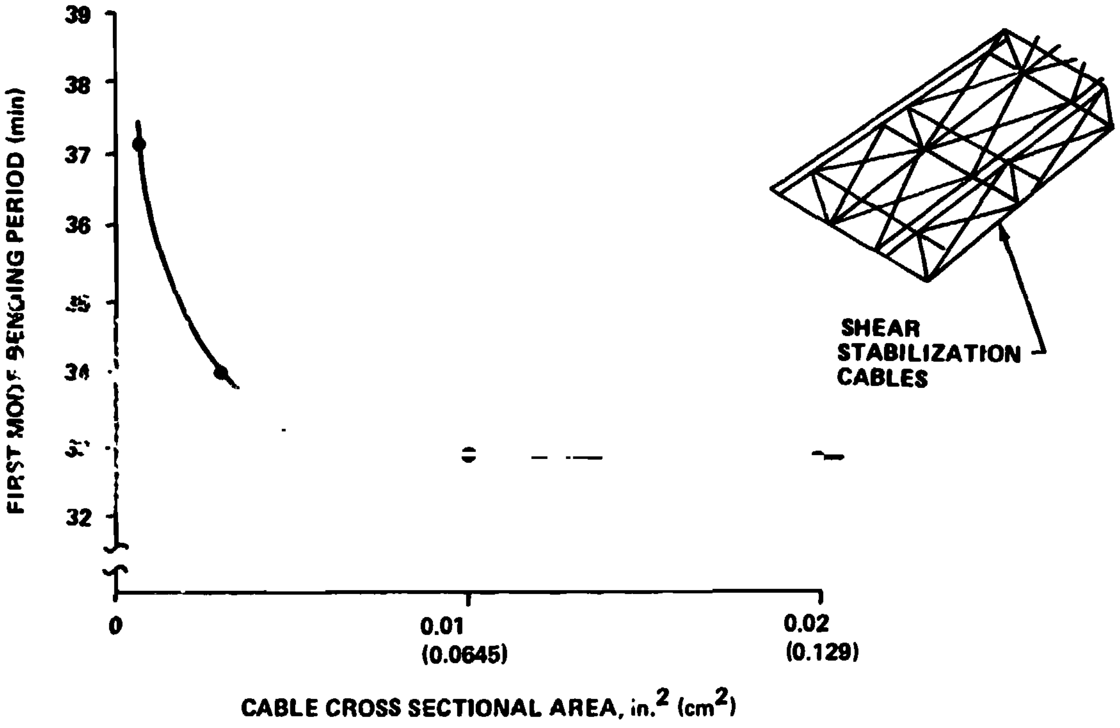


Figure 7-8. Effect of shear stabilization cable size on bending stiffness.

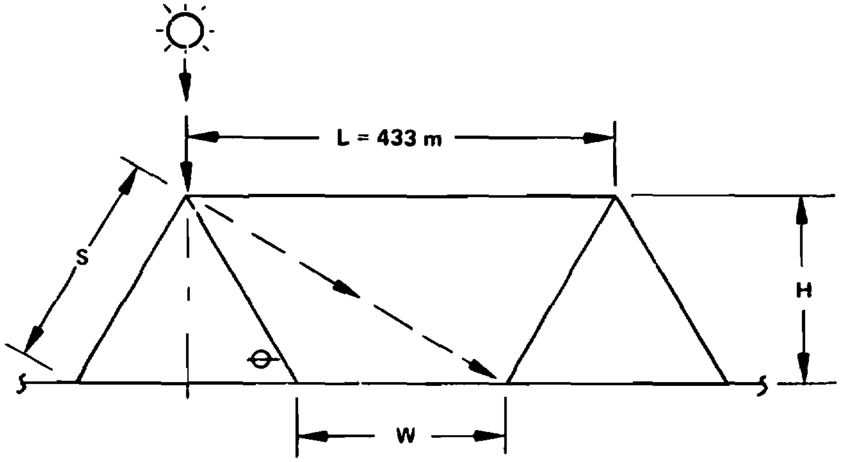
For different values of solar concentration ratio, the structural cross section geometry must change, since the primary structural members align with the reflectors and solar cell blanket. The geometrical variations indicated here are based upon maintaining a constant spacing between the upper longitudinal members. With increasing concentration ratio, the height of the structure, H, in Figure 7-9 increases, thus increasing structural moment of inertia and improving structural efficiency. However, with increasing concentration ratio, the total planform area must increase for a given power output to account for the decreasing solar cell efficiency that occurs as a result of increased temperature. It should not be concluded that options do not exist or cannot be developed that allow increased concentration ratio while retaining relatively high efficiency (Fig. 7-4, e).

For the structural cross sectional geometrics indicated previously, the change in first mode bending period was determined as a function of the structural member size, indicated by longeron cross sectional area. As can be seen in Figure 7-10, the structural member size increases with decreasing concentration ratio for given stiffness criteria. A longeron area of 0.605 cm^2 with an overall structural height of 214 m results in a first mode bending frequency of less than 25 min, which satisfies the 30 min upper limit.

The basic structure which was shown earlier for the photovoltaic power system, is a beam truss. This beam truss is triangular in cross section and consists of nine V-hat cross section longerons. These longerons are packaged into three small beams, nominally 1 m in height, located at the vertices of the truss beam.

The truss beam longitudinal loads are carried through the V-hat section longerons which are sized at 0.038 cm thickness and are shaped as shown in Figure 7-6. These longerons are fabricated from 2219-T62 aluminum and fail by local crippling at a compression load of approximately 4000 N. This failure mode is assumed independent of length, and the value is plotted as a horizontal dashed line in Figure 7-11. The Euler buckling failure mode is length dependent as shown on the same plot. The two curves intersect at a point corresponding to a longeron unsupported length of approximately 1.8 m. This point can be described as a design point where longeron failure could be expected to occur in crippling and/or buckling under a 4000 N compressive load. To increase the longeron unsupported length beyond 1.8 m would lower the load bearing capability of the longeron, according to the Euler buckling curve.

A family of curves representing Euler buckling loads for the small triangular beams located at the vertices of the basic truss beam and consisting of three longerons is presented in Figure 7-12 for beam heights ranging from



θ (degrees)	CONCENTRATION RATIO (AREA)	CONCENTRATION RATIO (EFFECTIVE)	H (m)	W (m)	S (m)	PLANFORM AREA (km^2)
68	2.44	2.10	316.06	177.61	340.88	193
65	2.29	2.00	261.15	189.45	288.15	182
62	2.11	1.89	214.36	205.05	242.78	177
59	1.94	1.76	174.60	223.18	203.69	168
56	1.75	1.62	137.46	247.56	165.81	159

Figure 7-9. Effect of solar concentration ratio on structural cross section geometry.

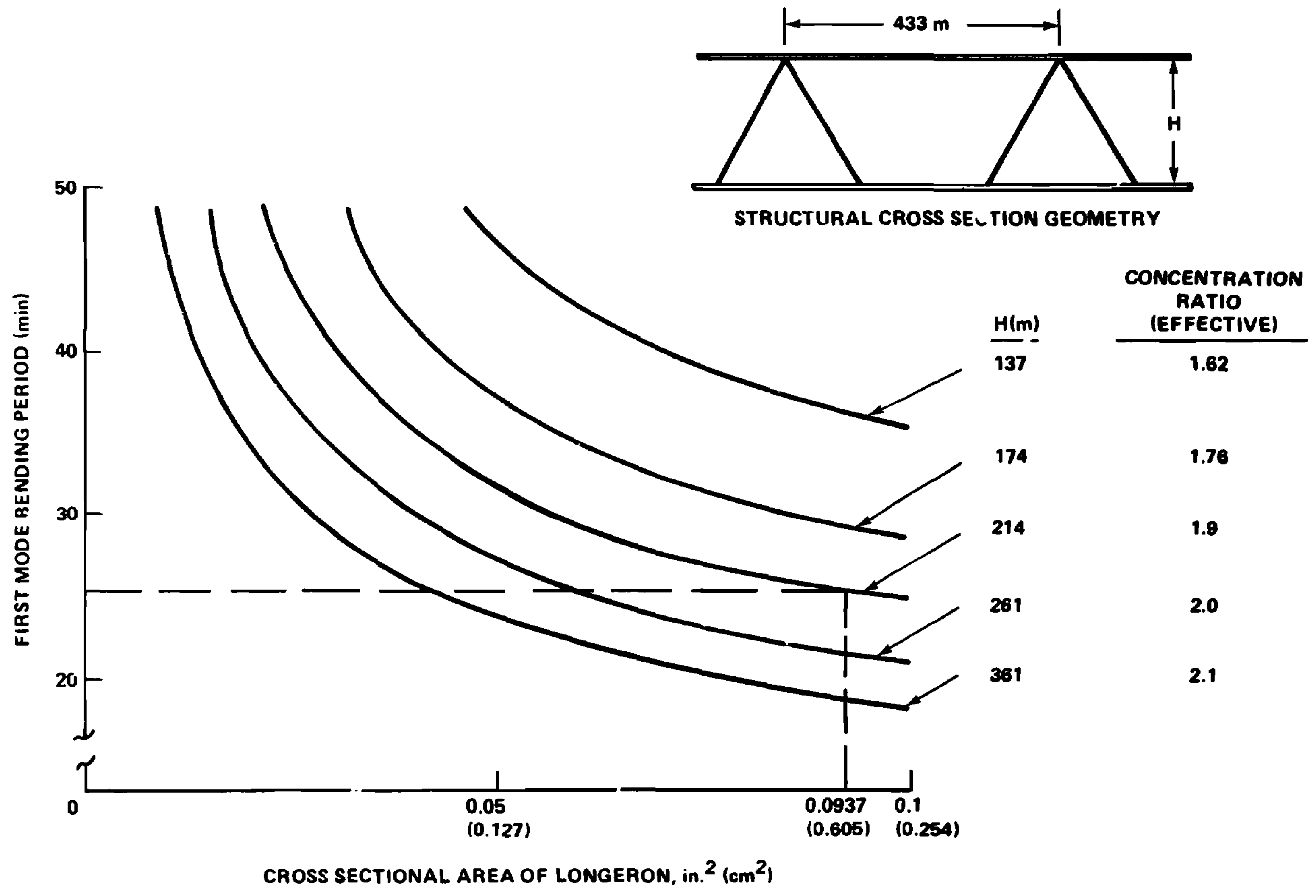


Figure 7-10. Effect of structural geometry and longeron area on structural stiffness.

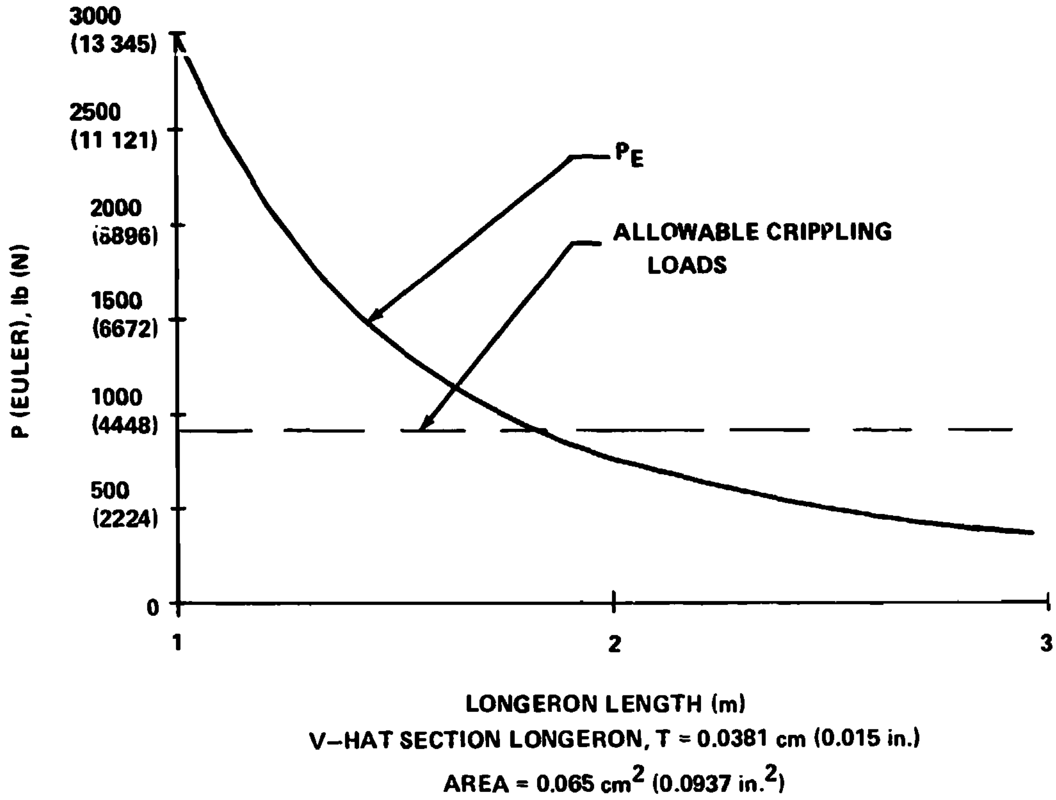


Figure 7-11. Euler buckling loads versus longeron lengths.

0.35 to 1.5 m. Also shown in Figure 7-12, by a horizontal dashed line, is the allowable crippling load for this three-longeron triangular beam. The allowable crippling load value was determined by assuming that all three longerons are equally loaded in compression and fail simultaneously. Thus, the crippling load plotted on the chart is the maximum allowable compressive load. Curves in this figure may be interpreted the same as for the longeron curve plot of Figure 7-11. For instance, the maximum practical distance between supports for a 1 m beam would be approximately 43 m.

Figure 7-13 is a plot of Euler buckling loads and a crippling load for a large beam truss. Data are shown for beam truss heights ranging up to 20 m. Since the large beam truss carries axial compressive loads through nine V-hat section longerons, the axial crippling load is nine times that of a single longeron. The curve plot is interpreted the same as for Figures 7-11 and 7-12.

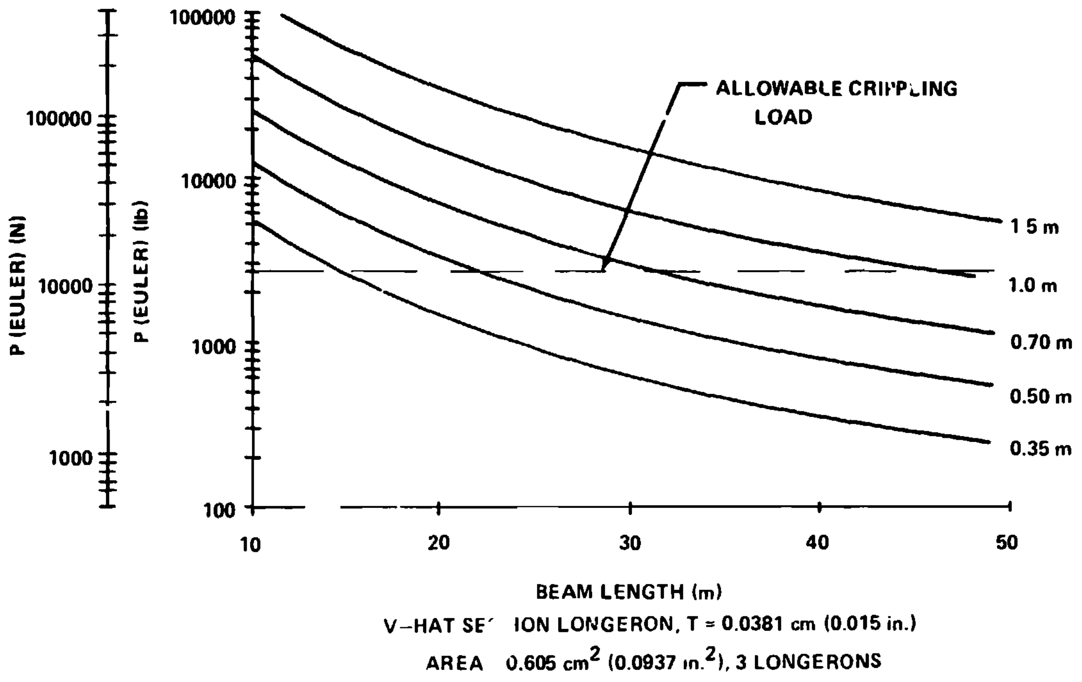


Figure 7-12. Euler buckling loads versus beam length for selected heights.

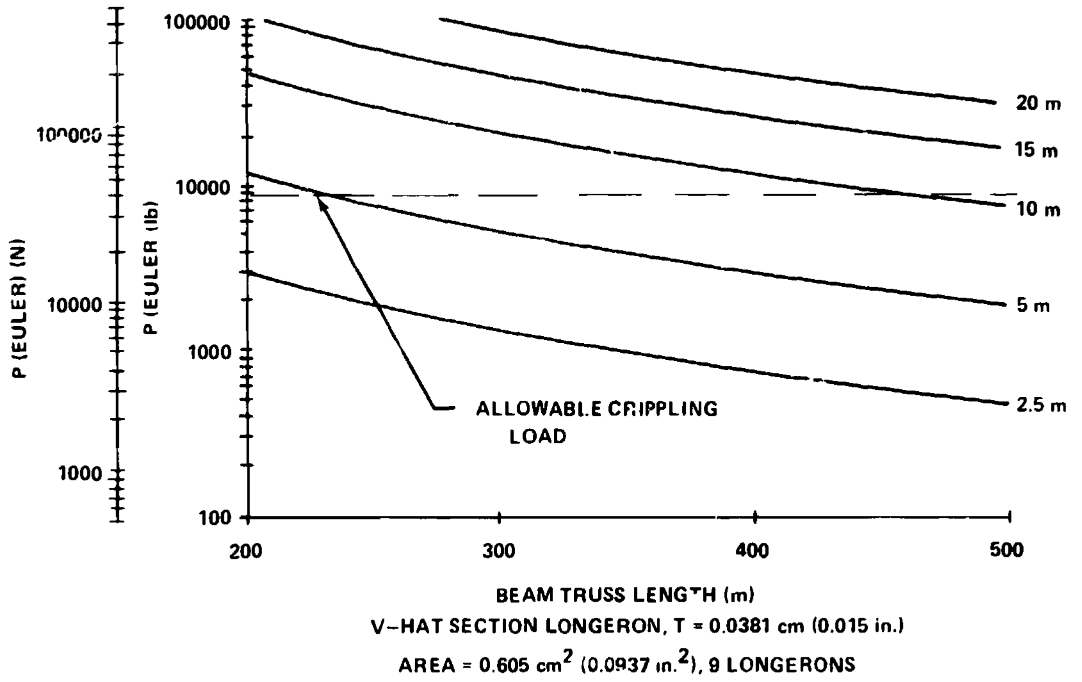


Figure 7-13. Euler buckling loads versus beam truss length for selected heights.

Longitudinal girder loads were computed and plotted for orbital transfer and attitude control thrust conditions. Figure 7-14 shows the results plotted against longitudinal distance from center. Also shown is the limit load capability of a girder sized for stiffness. When utilizing the thrusters for the orbital transfer (LEO to GEO) propulsion, it now becomes evident that location of these thrusters at the extreme geometric distances from the center of gravity of the satellite will produce the worst case structure design. The preferred location of the orbital transfer propulsion would be near the center of the configuration. The attitude control girder load shown occurs when opposing end-mounted attitude control thruster groups fire in opposite directions. All loads shown are assumed for static load conditions and are averaged across the structure width.

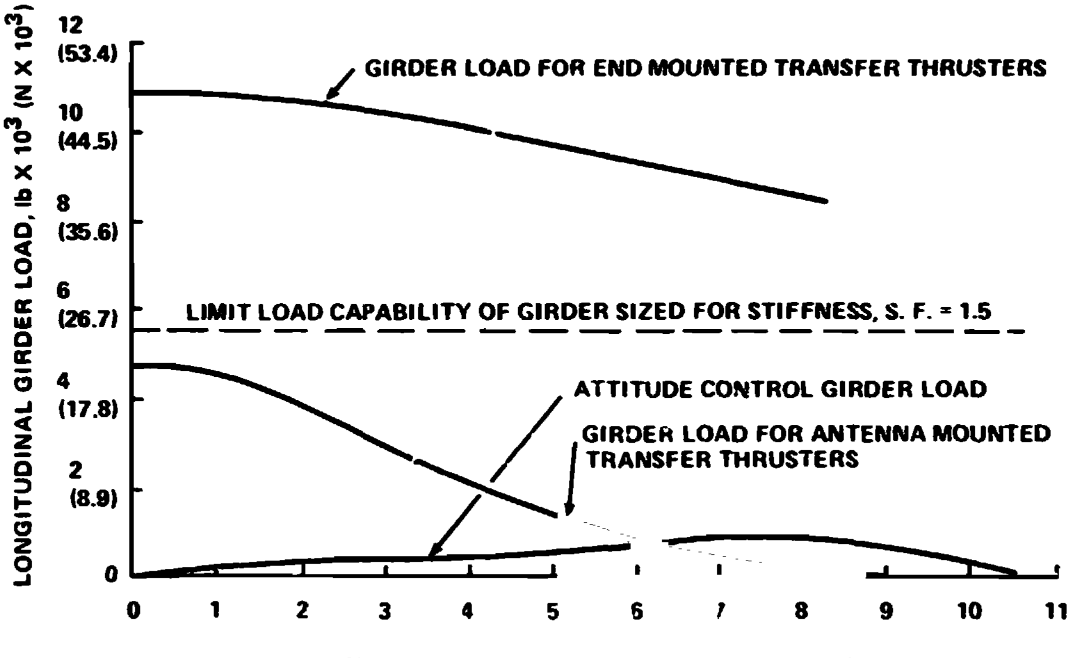


Figure 7-14. Longitudinal girder load distribution for orbit transfer and attitude control thrust.

In summary, it has been shown that the photovoltaic structural requirements can be met with a basic solar array structure consisting of large triangular truss girders ranging in length to 433 m with shear stiffeners provided by cross-bracing cables. The truss girders are fabricated from smaller triangular beams which have V-hat section longerons and channel section lateral members. Truss girders are also stiffened by cross-bracing cables. Height considerations for the smaller triangular beams range from 0.7 m to more than 1 m. Height considerations for the large truss girders range from 5 to 20 m. Nominal sizing of all members will require further iteration and refinement. Beam side loads imposed by the tensioning requirements of the reflectors and solar cell blankets must be addressed. Effects of thermal distortion and manufacturing anomalies must also be considered. Final beam truss sizing will depend on these additional considerations, plus any changes in orbit transfer thrust loads. Beam truss girder height requirements can be reduced considerably by reducing the beam length. In high bending stress areas, additional intermediate lateral members could be installed to decrease the effective length of the longitudinal beam truss girders. This would not require a change in the overall bay size nor would it affect the reflector panels and blankets.

7.1.4 POWER DISTRIBUTION AND CONTROL SUBSYSTEM

A simplified diagram of the selected power distribution system is shown in Figure 7-15. The solar array is divided into 792 modules producing 22 MW each. The solar array modules develop 20 kV which permits connection directly to the amplitrons without complex power conditioning equipment. Circuit protection and voltage regulation are provided at each module. The very large conductors required to transfer power to the amplitrons are the most significant consideration to the power distribution system. Methods were developed to minimize the mass required for power conductors, and criteria were established for combining the functions of power conductors with structural support.

7.1.4.1 POWER CONDUCTORS

The selected voltage of 20 kV is relatively low when compared to terrestrial transmission lines operating at 382 kV ac and 750 Vdc. The low voltage was selected to reduce power conditioning requirements, but it created the necessity to transmit a large current. The task for the SPS was to transmit 10^6 A over a distance of 20 mi. This obviously required large conductors (or structures). Compared to terrestrial transmission lines, the SPS task was easier because of the freedom from corona effects but more difficult from the standpoint of heat dissipation.

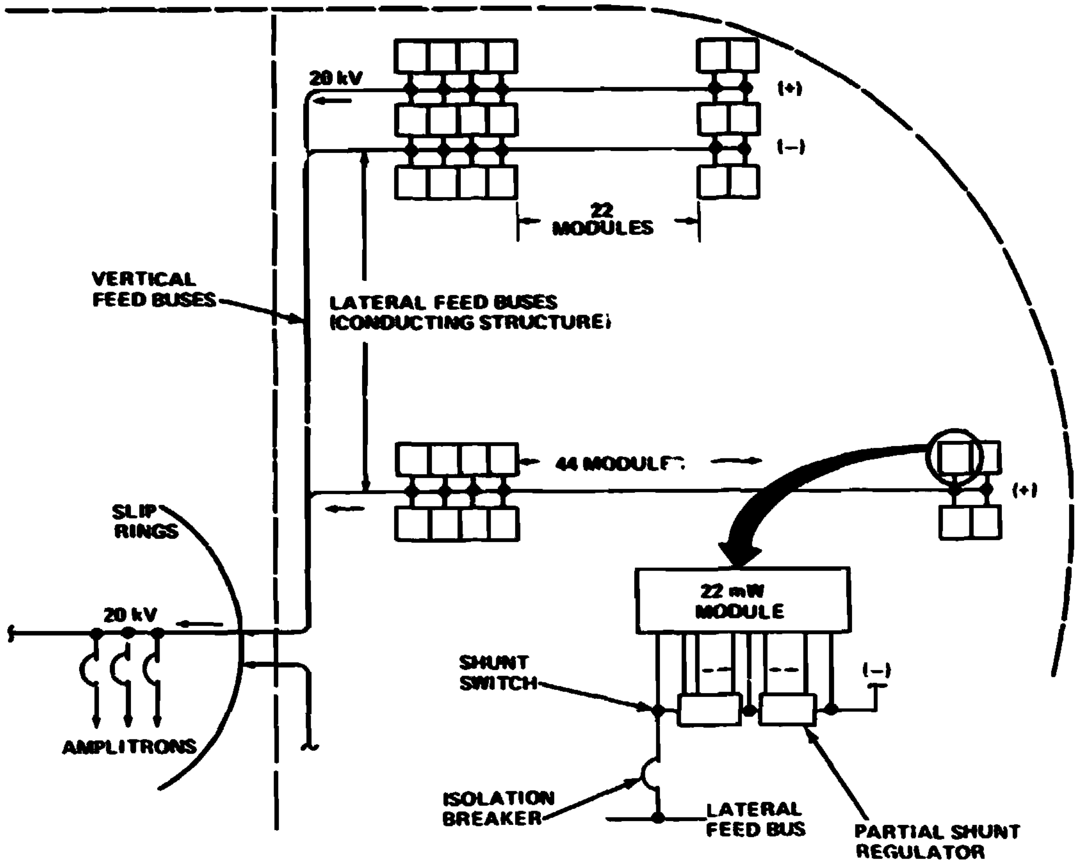


Figure 7-15. Electrical power distribution.

Characteristics of power conductors were driven from the requirement that a conductor must have sufficient surface area to dissipate the I^2R power loss for a given surface temperature. Conductor weight can be decreased without increasing temperature if a larger surface area is provided. Therefore, the power conductors for the SPS were in the form of aluminum tubes on sheets. The penalty for reducing conductor mass is an increase in power loss, as shown in Figure 7-16. This power loss must be compensated for by increasing the size of the solar array and corresponding nonconductive structure. The resulting increase in array weight is included on the same figure. The curves represent one-half of the 792 module solar array.

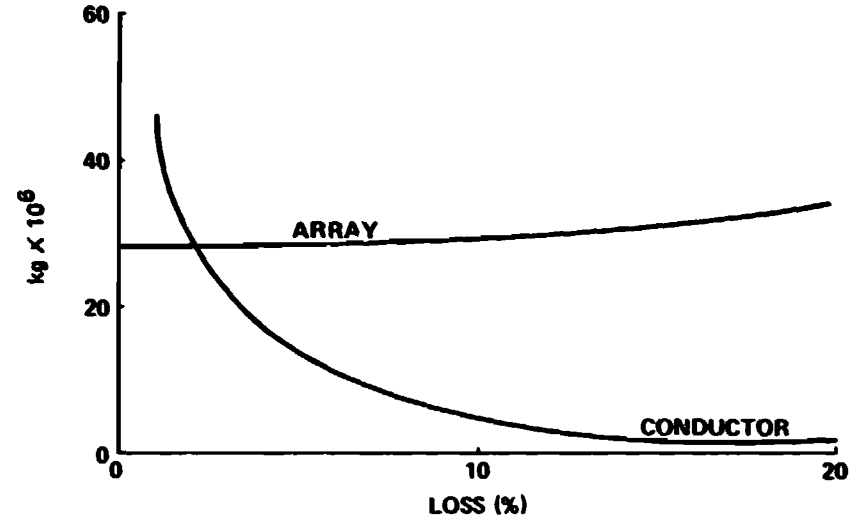


Figure 7-16. Individual solar array and conductor masses versus distribution losses.

A minimum system weig[illegible]urs when the summation of the solar array and power conductor masses co[illegible]o a minimum value. A surprisingly high power loss of 13.5 percent was fou[illegible]o yield the minimum system mass as shown in Figure 7-17.

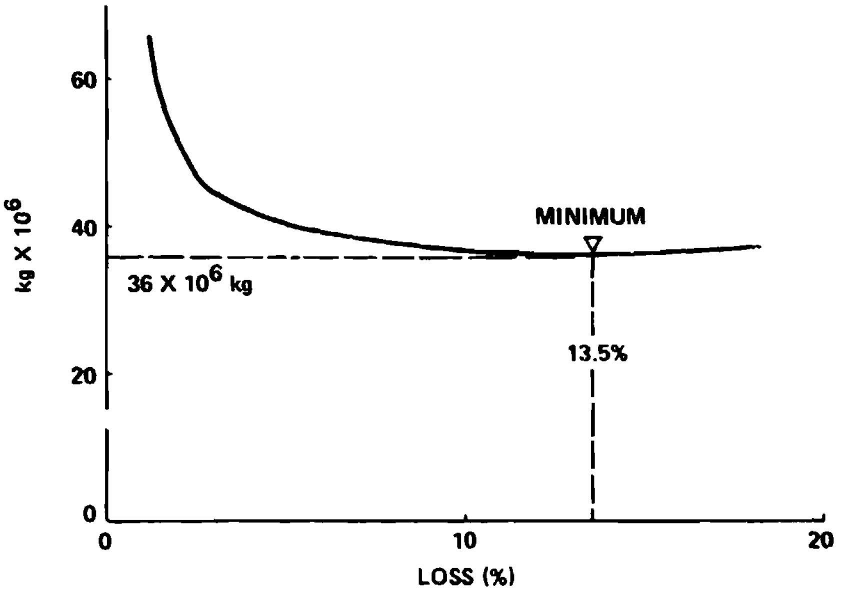


Figure 7-17. Solar array plus conductor mass.

Conducting structure was utilized for the lateral feed buses shown in Figure 7-15. A limited analysis to verify compatibility with structural requirements showed that the surface area provided by the baseline structure was adequate if thickness was increased. Thickness had to be increased by steps as modules were progressively connected along the length of the beam. This increased the mass of the structure by a ratio of 8 to 13 for the particular beam analyzed. Overall, the combined structure/power conductor weight was reduced by a ratio of 17 to 13 as a result of eliminating separate requirements for power conduction and structural support.

7.1.4.2 HIGH VOLTAGE POWER TRANSMISSION

While the total mass of the SPS can be reduced by transmitting at increased voltages (Fig. 7-18), the possible mass must be weighed against increased high voltage solar array/transmitter tube problems and the complexity of power conditioners. The relationship between mass reduction of power conductors and technology risk can be analyzed by considering the following power system options:

1. Operate the solar array, power line, and transmitters at 20 kV: 13.6×10^6 kg conductor mass.

2. Increase the operating voltage of the transmitters, power line, and solar array to 40 kV: 6.8×10^6 kg conductor mass.

3. Increase the operating voltage of the solar array and power line to 400 kV, and provide power conditioners with a specific weight of 0.456 kg/kW at the transmitter tubes: 8×10^6 kg conductor and conditioning mass.

4. Operate the solar array at a low voltage, the power line at 400 kV, and provide power conditioning at each end terminal: 16×10^6 kg conductor and conditioning mass.

For all options except option 4, the solar array was operated at a relatively high voltage when compared to present satellite solar arrays. This creates a considerable technology risk to the solar array development program because of the lack of analytical and test data on high voltage effects. Studies and limited vacuum tests have been conducted on small solar arrays biased at 16 kV, but the test results were inconclusive. There has been no flight test verification, and the highest voltage being considered for an advanced technology development program is the 400 V SEPS array.

A solar array voltage of 20 kV was established for the baseline SPS design because this was the voltage selected for the transmitting tubes (see subsection 7.3.4.1), and it was assumed that power regulators could be avoided.

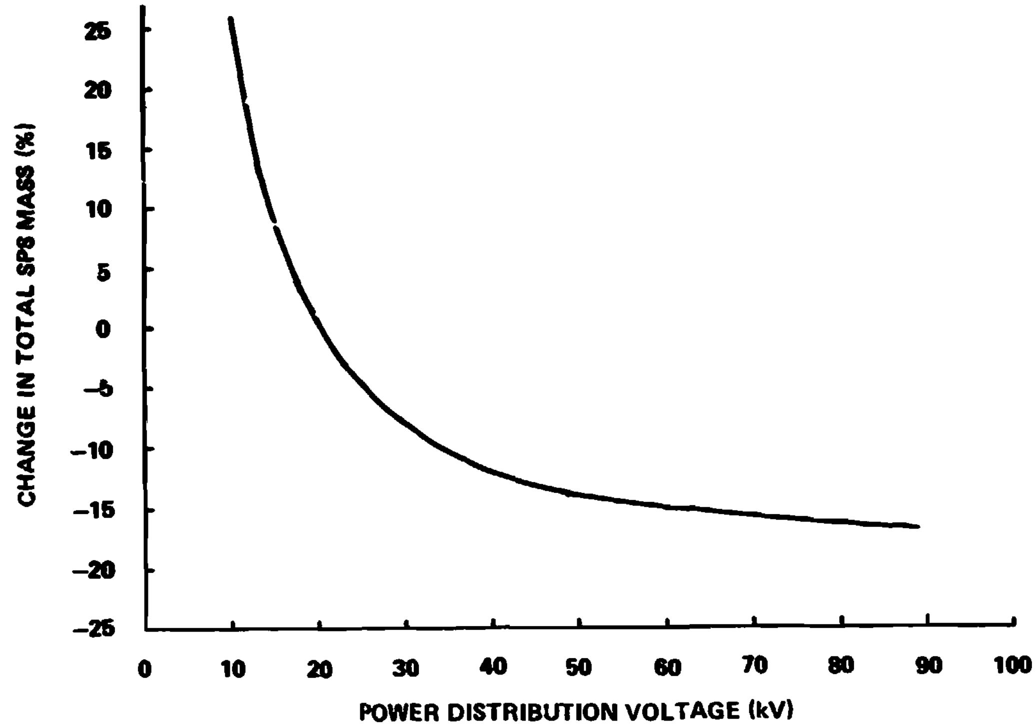


Figure 7-18. Power distribution voltage trade.

Consequently, there are two possible high voltage problems to be considered: (1) power loss due to interaction of the space plasma with the solar array and (2) "sparking." Based on preliminary investigation of solar array power loss caused by plasma interaction, a power loss of 0.02 percent was predicted for operation at synchronous orbit. It should be cautioned that this is a preliminary estimate based on tentative analytical math models and that flight testing should be undertaken to substantiate power loss prediction methods. For the second concern, solar array sparking, it was postulated by Lewis Research Center [14,15] that sparking might be controlled by insulating all conducting surfaces floating negative with respect to the space plasma. Provisions for insulation were not incorporated into the design of the solar array because of the lack of design data defining the sparking phenomena. Special emphasis studies have been initiated to analyze high voltage effects, but results are not expected for some time.

For the second option it was assumed that the operating voltage of the transmitter tubes was increased to 40 kV and the voltage of the solar array and power line was increased correspondingly to 40 kV. By a first order approximation, the mass of the conductors was reduced from 13.6×10^6 kg to 6.8×10^6 kg. The power loss due to plasma interaction was increased from 0.02 percent to 0.04 percent. The effect of sparking remained undetermined. Option 2 is the lowest mass option for power distribution that requires the development of a 40 kV transmitter tube and a higher voltage switch gear.

For those believing that voltage problems in space are minimal, the third option is attractive. The solar array and transmission line are operated at 400 kV, and power conditioners are provided to reduce the voltage for the transmitter tubes. Although this appears to be a high technology risk option, it cannot be rejected until more is known about high voltage effects and specific tube requirements.

The final option considered was similar to terrestrial transmission systems where power conditioners are provided at each end of a high voltage dc transmission line. This option allows independent selection of operating voltage for the transmitter tubes and the solar array as may be necessary depending on technology development. The mass required for power conditioning at each end of the transmission line makes this option the highest weight approach.

7.1.4.3 SWITCH GEAR

In-line circuit protection is required for each 22 MW module in the system, as shown previously in Figure 7-15, to prevent power feedback from the main lateral buses. Crossed-field (Penning) discharge circuit breakers rated at 20 kV, 1.1 kA were tentatively selected. One feature of the distribution system is that lateral feed buses are not tied together (multiple circuit slip rings are required); therefore, excessively large circuit breakers are not required for isolation of the vertical feed buses. Solid state shorting switches are provided on the solar array to short circuit array sections during maintenance.

7.1.4.4 POWER CONDITIONING

The system voltage is regulated to within 1 percent to satisfy requirements of the amplitrons. The voltage of a solar array varies as a function of connected load, solar array orientation, solar cell temperature, and aging. The method selected for voltage regulation is to sequentially short out sections of the array. The location of the partial shunt regulators is indicated in Figure 7-15. The effects of solar array faults need to be analyzed also.

7.1.4.5 MAGNETIC FORCES

A brief analysis was performed to estimate magnetic forces acting on power conductors. A model, consisting of a ring of four positive and four negative thin walled tubular conductors, was developed for the analysis. The total current was 10^5 A. Based on preliminary calculations, the forces appeared to be of a manageable level. The radial pressure tending to collapse each tubular conductor is 138 N/m^2. Differential forces are 2.2 N/m and 1.8 N/m. The outward radial force on the eight conductors comprising the ring was 4.4 N/m.

7.1.4.6 CONDUCTOR MASS FACTORS

There appear to be several promising design approaches for reducing the power conductor mass of the 20 kV system, although the system engineering required to evaluate the approaches was not attempted in most cases. Factors contributing to conductor mass can be listed as follows:

1. Array Shape — Conductor mass and power loss are functions of length. If length is doubled, both mass and power loss are doubled. The shortest conductor length is obtained from a split circular array with the antenna located at the center. The large reduction in mass and power loss possible for a circular array is shown in Table 7-5.

TABLE 7-5. CONDUCTOR MASS AND LOSS

SPS Configuration	Antenna Location	Number of Antennas	Mass (kg × 10^6)	Loss (%)
Circular	Center	1	5.[illegible]	9
Elliptical (Reference)	Center	1	8.6	13.5
Elliptical	Long Axis	2	10	15

2. Antenna Location — Power is collected from a distributed solar array source and concentrated at the transmitting antenna. Th[illegible], conductor size increases near the antenna. When the antenna is locat[illegible] the extremities of the solar array, the major portion of the conductor mass is likewise located at the extremities. The result is increased structure mass, less opportunity to utilize conductive structure, and increased conductor length. The increase in conductor mass and loss is shown in Table 7-5 for a case where two antennas are located on the long axis of an elliptical solar array.

3. Conductor Structure Axis — For the waffle type structure of the SPS, it is considered overly complex to route power in both X- and Y-directions. A parametric comparison of combined conductor and structural mass for several possible options is as follows: (a) no conducting structure, 34 × 10^6 kg, (b) conducting structure for X-axis only, 29 × 10^6 kg, (c) Y-axis only, 26 × 10^6 kg, and (d) X- and Y-axis, 21 × 10^6 kg. Option (c) was selected, although some studies have indicated that it is possible to approach the ideal condition of option (d) through a deliberate basic design approach with the goal of reducing conductor mass.

4. Bus Material — Aluminum reduces conductor mass by 57 percent as opposed to copper for a given temperature and has been selected for the structural material.

5. Bus Configuration — As discussed previously, thin conducting sheets or tubes with favorable thermal view factors are preferred for power conduction. However, consideration must be given to reducing power loss caused by interaction with the space plasma. Power loss is proportional to exposed conducting surface area and can be significant under certain plasma conditions.

7.1.4.7 REQUIRED TECHNOLOGY ADVANCEMENTS

The primary concern in the power distribution subsystem was the large structures required for power transmission. The major technology requirement lies with the assembly of large structures in space; however, the conduction of power across structural joints must be given attention. The most immediate technology concern relates to the interaction of the space plasma with high voltage solar arrays. Studies are underway to further analyze the phenomena, but additional effort is required to obtain experimental verification of power loss calculations and proposed solutions to reduce sparking. A considerable effort is in progress to study the other voltage problem — spacecraft charging. This activity should be continued.

Three types of in-line commercial circuit breakers have potential application for the SPS: (1) crossed field (Penning) discharge devices with interrupting capacity of 100 kV, 2 kA, (2) vacuum interrupters with capacities to 20 kV, 15 kA, and (3) modified ac circuit breakers as preferred by the Europeans. Studies should be undertaken to find the most suitable breakers.

The baseline SPS utilized a partial shunt switching regulator to control bus voltage, but there are several competitive system options that would require dc-to-dc or dc-to-ac and ac-to-dc power conditioners. The list of candidate active circuit components that should be evaluated includes hydrogen thyratrons with capacities to 50 kV, 10 kA and solid state thryistors with capacities to 2 kV, 1 kA.

7.1.5 FLIGHT MECHANICS

7.1.5.1 STATION KEEPING

Station keeping for the SPS is concerned primarily with two major phases: the assembly in low Earth orbit and the operational phase in geosynchronous equatorial orbit.

During assembly in low Earth orbit, the major perturbation that must be corrected is that resulting from aerodynamic forces acting on the system. The aerodynamic drag force acting on the system is

$$D \approx 4.4 \times 10^{-4} (A) \text{ N} \quad ,$$

where the projected area (A) is measured in square meters. The area is continually changing during the SPS assembly. Assuming the solar array is 25 percent deployed at the end of the assembly and the start of the transfer to geosynchronous orbit, the SPS will experience an aerodynamic deceleration of 2 × 10^{-5} g, which will require a propulsive force of 26 000 N to maintain altitude.

Station keeping the SPS at its operational geosynchronous orbit is complex in that stringent position control must be maintained in the presence of a number of perturbing forces that tend to cause the spacecraft to drift from its position. The perturbations that are encountered include those due to Earth oblateness, solar and lunar gravitational attractions, solar radiation pressure, and microwave pressure. The gravitational anomalies due to Sun, Moon, and Earth oblateness produce short period oscillations in all the orbital elements, but the major perturbation of concern is the nodal precession in the plane of the perturbing body. This nodal precession causes a long-period oscillation of the inclination of the spacecraft orbit to the Earth's equatorial plane. By remaining unchecked, these perturbations would cause the inclination of the SPS orbit to increase to a maximum of 15° in approximately 26.5 years. Table 7-6 shows typical motions that might be imparted to the operational SPS by gravitational anomalies.

TABLE 7-6. SYNCHRONOUS ORBIT PERTURBATIONS

Perturbation	Disturbing Force	Plane of Motion	Motion[a] (degrees/day)
Nodal Regression	Oblate Earth	Equatorial	0.0134
	Solar Gravity	Ecliptic	0.0018
	Lunar Gravity	Lunar	0.0040
	Combined	Equatorial	0.0188
Inclination	Combined		0.0023

a. Typical values expected; actual values dependent on launch date and time.

Because of the large area of the SPS, the dominant perturbing force in the operational orbit is due to solar radiation pressure. Assuming a solar radiation constant of 4.5 × 10^{-6} N/m^2 (no reflection) and an area of 242 × 10^6 m^2, the solar pressure exerts a force of approximately 1090 N on the SPS. This solar pressure produces short-period (24 h), long-period (approximately 1 year), and secular variations to the orbital elements of the SPS. If the SPS has the design capability to accommodate the orbit variations caused by the short-period perturbations, then the major impact of the solar radiation pressure is the change of the major axis of the orbit, which is cyclic over approximately 1 year and causes a change in the orbital mean motion with a resulting longitudinal drift of the SPS. Figure 7-19 shows the combined effects of the perturbations on the longitudinal position of the SPS. If station keeping maneuvers are conducted every 30 days, a velocity of approximately 3.7 m/s will be required to correct the inclination, and a velocity of 14 m/s will be required to correct altitude and ellipicity of the operational orbit.

7.1.5.2 LEO TO GEO ORBITAL TRANSFER-MISSION AND PERFORMANCE ANALYSIS

7.1.5.2.1 BACKGROUND

Previous studies at MSFC, both in-house and contractual, have shown that high specific impulse, low thrust electric propulsion provides a significant economic advantage over present and projected chemical propulsion

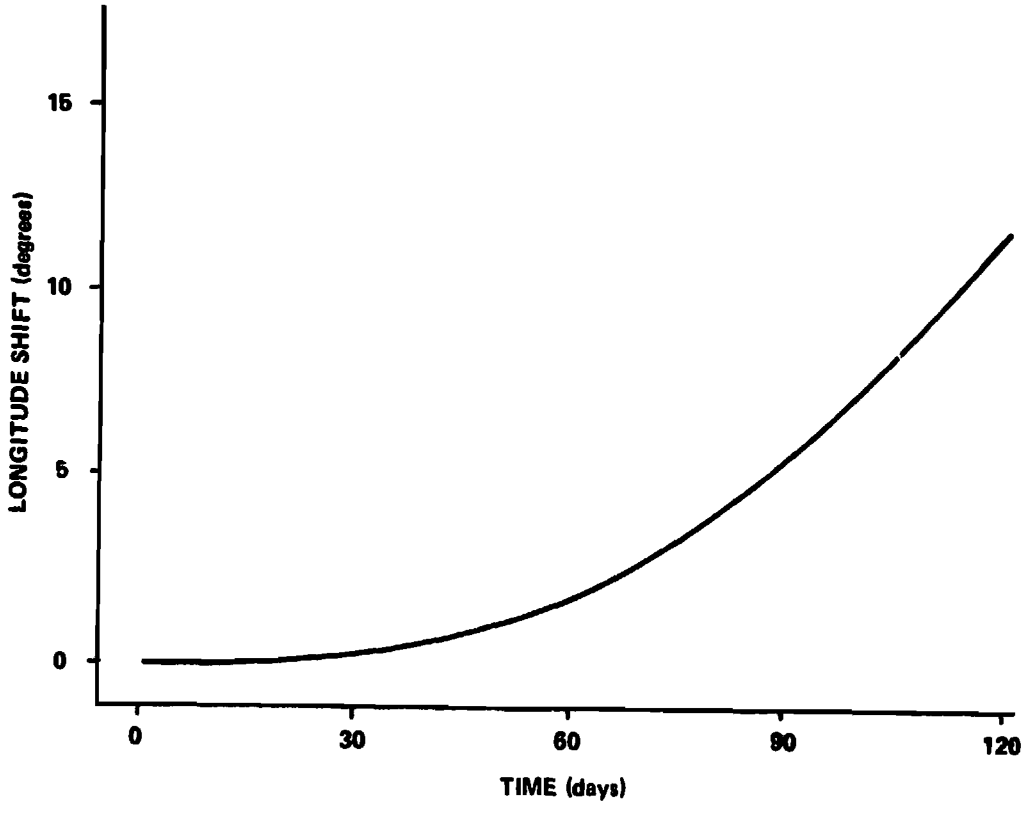


Figure 7-19. SPS longitude shift due to perturbations.

systems for the transporting of massive payloads from LEO to GEO or other intermediate orbital points. Consequently, low thrust electric propulsion becomes a prime candidate for the transporting of a self-powered solar power satellite from a low Earth assembly orbit (435 km) to a geosynchronous position.

The theoretical feasibility for implementation of this very complex problem has been demonstrated; however, the challenge is so great that the complexities and anomalies involved are unequaled by any space venture that has been attempted.

7.1.5.2.2 MISSION PROFILE

A projected mission profile for LEO to GEO orbital transfer will have to take into consideration a number of trades to arrive at the optimum set of initial orbital parameters that will in effect define the assembly orbit for the construction of the SPS. Outstanding among these trades are:

1. HLLV and electric propulsion performance penalities and limitations

2. Thermal cycling (loss of thrust due to shadowing)

3. Solar array (Van Allen Belt transit) degradation

4. Economics associated with long versus short trip times

5. Aerodynamic drag in LEO

6. Thrust load limitations on SPS structures.

Several iterations of these trades have been performed, and trends are developing such that a tentative or preliminary mission profile and associated assumptions may be stated at this time. A first significant assumption is that a nearly total sunlit orbit is assumed for the transfer phase of the mission.

Assumption of an SPS assembly orbit of 435 km and 55° inclination will eliminate the necessity and cost of a chemically augmented (hybrid) propulsion system while utilizing only low thrust propulsion for orbital transfer. Opportunities for sunlit orbit interception and SPS transfer occur at this point. However, an HLLV performance penalty of approximately 25 percent (with an analogous increase in launch cost) will be incurred for 55° inclination launches when compared to due East (28.5°) inclination launches. Some study data indicate that this penalty may be reduced to less than 10 percent for multistage HLLV's, but this must be further substantiated. If an SPS assembly orbit of

28.5° inclination is assumed, a chemically augmented propulsion system will be needed to provide an acceleration of approximately 10^{-3} g for approximately 21 h to boost the 1.4×10^8 kg SPS payload (chemical system mass not included) to an altitude of approximately 2037 km to intercept a totally sunlit orbit. Thus, the trade is between the cost and complexities of the chemically augmented propulsion system versus the increase cost because of performance penalties of HLLV launches to a 55° assembly orbit.

First iteration conclusions for a smaller scale SPS indicate that these relative costs nearly cancel each other. If, however, it can be demonstrated that HLLV launch penalties can be reduced to less than 10 percent, then an economic advantage is evident.

For this study, an SPS assembly orbit of 435 km and 55° inclination is assumed, utilizing low thrust electric propulsion for orbital transfer from LEO to GEO and taking from 60 to 100 days to arrive on station. A second important assumption for SPS LEO to GEO orbital transfer is that a low thrust sortie through the Van Allen radiation belt will cause an exposed (nominal cover-slide assumed) photovoltaic array to degrade to a significant degree.

Previous in-house studies have provided considerable insight into the problem of degradation by charged particle radiation with regard to design impacts and operating procedures for SPS transits through the "hot" portion of the radiation belts. Analyses have shown that at altitudes of 5556 to 7408 km, an equivalent intensity on the order of 10^{15} 1 MeV $\frac{\text{electrons}}{\text{cm}^2\ \text{day}}$ will penetrate a 3 mil cover-slide and reach the vulnerable portion of a solar cell. The fastest expected SPS transit time through this region using low thrust propulsion is on the order of 4 days, which will result in an almost instantaneous power degradation of exposed solar arrays on the order of 25 percent. Although this altitude range describes only the "hot" portion of the belt, significant array degradation occurs in an altitude range from as low as 1852 km to approximately 13 890 km.

The implications are apparent. Figure 7-20 shows that trip times on the order of 60 days from LEO to GEO require many thousands of magnetoplasmadynamic (MPD) thrusters requiring very large quantities of power. A significant portion of the total spacecraft power generation capability must be available for the electric thrusters during the orbital transfer phase. If solar cells are the energy collecting devices, then those exposed are subject to greater degrees of degradation.

Referring to Figure 7-20, extending the allowable trip time to 100 days significantly reduces the power requirement of the propulsion system and subsequent degradation; thus, extending the mission trip within the constraints

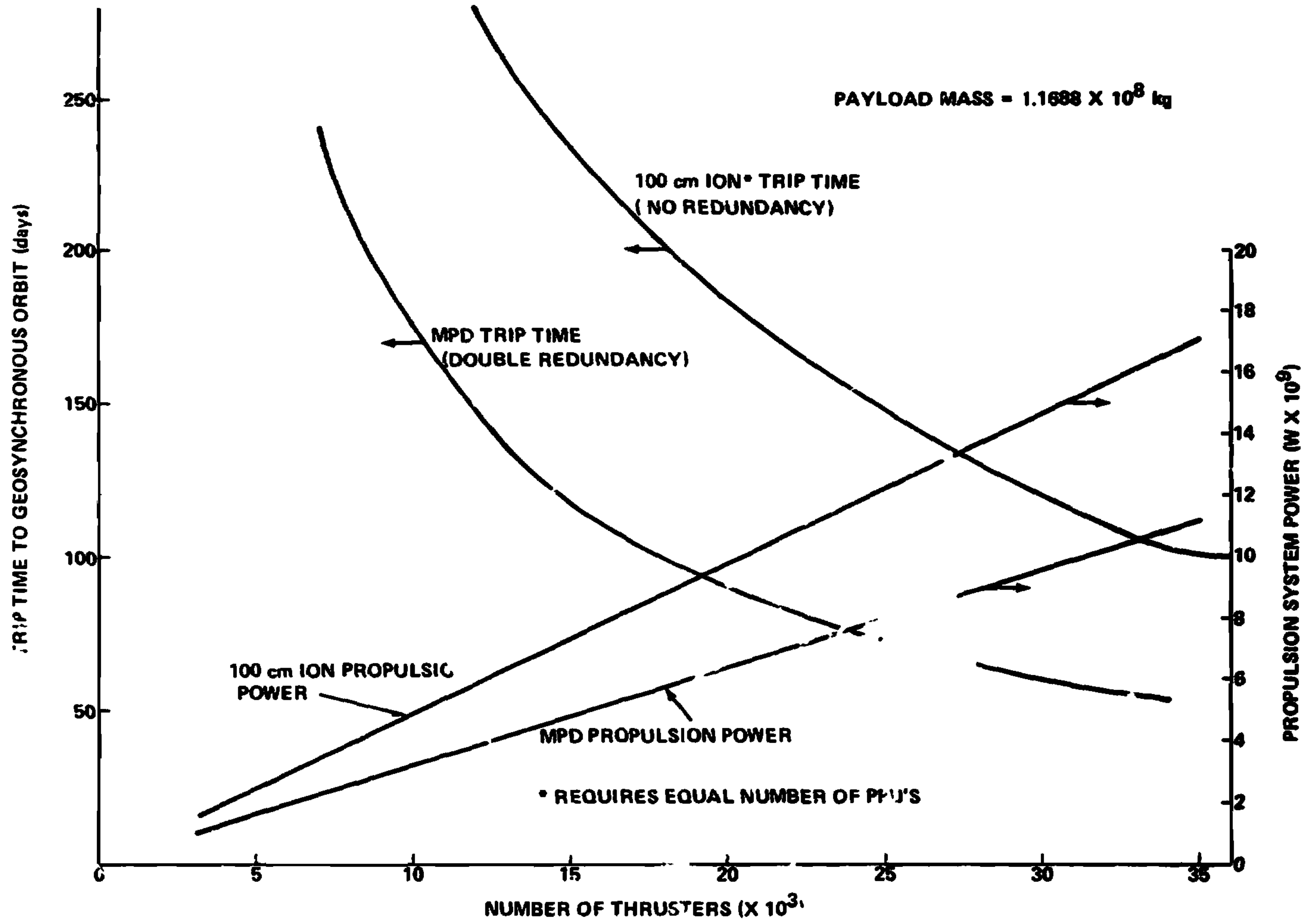


Figure 7-20. Electric propulsion performance and requirements.

of achieving a totally sunlit orbit would be cost effective. Other economic factors regarding allowable trip time will need to be factored into these studies.

7.1.5.2.3 ELECTRIC PROPULSION PERFORMANCE EVALUATION AND REQUIREMENTS

Two designs or concepts of electric ion thrusters received significant attention for application to the self-powered SPS orbital transfer capability. These are a 100 cm electric ion thruster and a 10 cm MPD thruster as defined in the midterm review of NASA Contract NAS8-31444, "Payload Utilization of SEPS (PLUS)," by The Boeing Company on March 2, 1976. These thrusters are only concepts and will require a substantial technology advancement effort to be available in the SPS projected operational time frame.

For reasons that will become apparent during this discussion, the MPD electric thruster concept appears to have significant performance advantages over other electric thruster concepts and is therefore a suggested baseline for SPS application.

Figure 7-21 shows relative performance expectation and system masses for both the 100 cm ion thruster and the MPD thruster propulsion systems. From Figure 7-21 it is seen that a short trip time (less than 100 days) produces an exponential rise in the propulsion system mass for the 100 cm ion

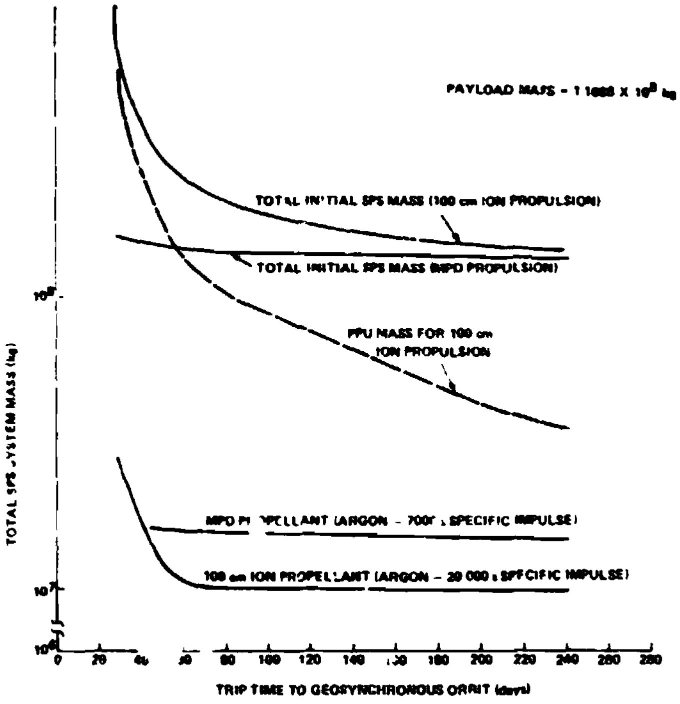


Figure 7-21. Relative SPS total mass versus LEO to GEO orbital transfer time.

thruster concept, holding the SPS payload mass constant. This is explained by the notations on Figure 7-20 that a number of power processing units (PPU's) are required equal to the number of 100 cm ion thrusters needed. The PPU's for this thruster concept are quite heavy, weighing nearly 1600 kg each, while the relative mass of the needed PPU's for the MPD thruster concept is negligible by comparison. A significant mass reduction in the PPU concept for the 100 cm ion thruster will reduce the demonstrated performance advantage of the MPD thruster concept.

This mass penalty places severe limitations on the performance of a 100 cm ion propulsion system. For example, Figure 7-20 shows that the propulsion power requirements for a 100 day trip time will equal the power producing capability of the entire SPS solar array. This means that the degradation factor of 50 percent will result in staggering and unacceptable power and economic losses. The conclusion is that the 100 cm electric ion thruster concept does not appear feasible for SPS application as presently envisioned.

Figure 7-21 also shows that the mass fraction ratio of total system mass to payload mass for the MPD 60 to 100 day trip times has a range of 1.23 to 1.197, respectively. The argon propellant requirement for this trip range is approximately 15.55×10^6 kg, assuming a specific impulse of 7000 s for the suggested baseline MPD thruster. The number of MPD thrusters required for a 60 day LEO to GEO orbital transfer is 30 000, each thruster having an assumed thrust level of 6.85 N.

Figure 7-22 shows the expected initial system acceleration as a function of the number of electric thrusters required, which can also be equated to trip time in days when used with Figure 7-20.

Earlier in-house structural analysis for the photovoltaic SPS concept indicated that a system acceleration level of 1.0×10^{-4} g for orbital transfer was a near limit for the specific structure design. This acceleration limit corresponds to approximately 20 000 baseline MPD thrusters and a trip time of 90 days. This specific acceleration limit is compatible with the present mission profile range of trip times.

The initial system acceleration requirements shown contain a small margin for steering losses. For a strictly orbit raising application, thrust vector pointing is nearly perpendicular to the radius vector for an initially circular orbit. Thrusting for inclination changes by SPS, however, requires considerable out-of-plane thruster steering and necessitates a considerable gimbaling capability for the orbital transfer thrusters. The mission profile, as

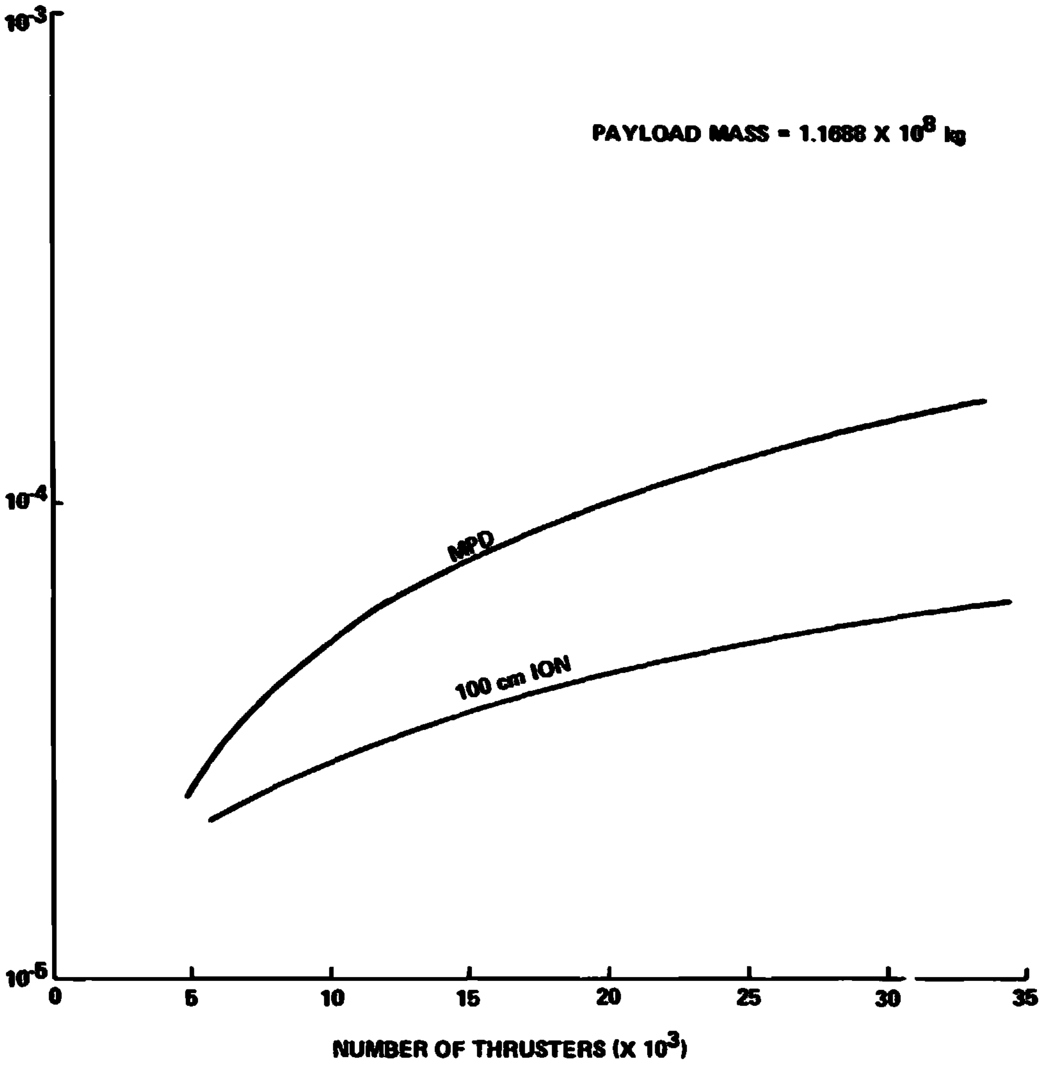


Figure 7-22. Total initial SPS system acceleration due to orbital transfer thruster firings.

set forth earlier, for maintaining a nearly total sunlit orbit will require simultaneous orbit raising and inclination changing (out-of-plane) steering to be successfully accomplished.

These analyses, although set forth for a photovoltaic SPS concept, should apply in a near proportional manner to a solar thermal SPS concept. An advantage of the solar thermal concept is the absence of a solar array degradation problem. However, some studies have indicated that the Van Allen radiation may cause mirror [illegible] other reflective surfaces to suffer degradation of original properties if exposed during LEO to GEO orbital transfer.

7.1.5.3 ORBITAL TRANSFER PROPULSION

Assuming the SPS is assembled in low Earth orbit (435 km) with subsequent transfer to its geosynchronous orbit, an integrated SPS propulsion system will be required to perform the transfer as well as maintain attitude control at all times. The SPS propulsion system will also have to provide orbital drag makeup during assembly in the low orbit as well as correct for orbit distortion because of solar pressure. Geosynchronous station keeping propulsion requirements include counteracting the effects of solar pressure, Sun and Moon induced inclination changes, Earth ellipticity induced drifts, and possibly microwave recoil. Attitude control propulsion requirements include counteracting the effects of gravity gradient torques (in both the LEO and the GEO) and, possibly, microwave recoil and antenna angular accelerations.

This subsection is devoted primarily to discussing the propulsion requirements for orbit transfer. The propulsion requirements for attitude control are discussed in subsection 7.1.6.2. However, the orbit transfer propulsion system(s) can be used (at a reduced allocation) to aid in station keeping and controlling the attitude of the SPS.

The major portion of the cost of the SPS propulsion system involves transporting the propellant required for the LEO to GEO transfer from the Earth's surface to LEO. To minimize this amount of propellant, and subsequently the cost, the SPS propulsion system must utilize an engine capable of producing a specific impulse much greater than conventional chemical engines. The high specific impulse achievable with electric propulsion is attractive, particularly when considering the availability of a large amount of onboard power. In designing a propulsion system to transfer a photovoltaic SPS from the assembly LEO to the operational GEO, numerous design variables must be considered, such as follows:

1. A low SPS acceleration level will be required (≤ 0.001 g).

2. Continuous tangential thrusting to maximize propulsion use efficiency is desirable.

3. Plume impingement avoidance seems necessary.

4. Earth occultation, if unavoidable, will allow electric propulsion only during the sunlit portion of each orbit, thereby requiring on-off operation of electric thrusters with subsequent thermal cycling; the effective shadow time is increased by any thruster start delays.

5. A chemical propulsion system will be required at least for attitude control to assure Sun orientation during the occultation.

6. If the numerous electric thruster start transients cannot be allowed, a chemical propulsion system will be needed to climb to a continuously sunlit orbit.

7. In addition to providing attitude control, propulsion (electric and/or chemical) will be needed for atmospheric drag makeup and solar pressure corrections.

8. Since the physical size of the SPS limits maneuverability, the transfer propulsion system (assuming a continuously sunlit orbit) will have to be integrated into the SPS structure such that continuous tangential thrusting can be allowed without plume impingement or on-off operation of electric thrusters.

Operating electric thrusters in an on-off manner as required by occultation of the SPS is undesirable because of thruster thermal cycling during on-off propulsion operations. Until further system definition indicates a more desirable optimization, a chemical propulsion system is assumed to boost the SPS to a sunlit altitude and inclination such that continuous electric propulsion can be expected. The exact orbital parameters are yet to be derived (considering low cost), consequently, the chemical propulsion system required to perform the transfer has not been sized. However, it will be desirable to begin the trip in advance of solstice to provide a transfer window and to maximize the available unocculted trip time. Should the geosynchronous transfer burn not be initiated during the launch window, the SPS may have to remain in LEO for another year until the Earth's shadow is again in the proper position with respect to the SPS orbit. The electric propulsion system required to perform the LEO to GEO transfer must be selected on the basis of low transfer cost (dollars/kilogram SPS). The various orbit parameters and mission requirements must be established. Then, for a given SPS (size, mass, and acceptable acceleration level), the characteristics (number of thrusters, masses, etc.) for different types of electric propulsion systems to satisfy the mission requirements are computed. The cost of the electric propulsion systems are computed based upon a credible cost model. The transfer cost based upon the inherent range of performance (specific impulse) for each type of electric propulsion system and the transfer cost versus varying propulsion times are calculated. Other parameters to be considered include launch cost and the economic factor of radiation degradation. The type, size, and required performance of the electric propulsion system to satisfy the mission requirements at the lowest cost can then be selected.

The chemical system selected to boost the SPS from LEO to a continuous sunlit orbit is O_2/H_2 using a high pressure, pump fed, tug type engine. This selection is preliminary and is based on the high specific impulse of O_2/H_2 (approximately 470 s) minimizing the launch cost because of less propellant needed in LEO. Integrating high thrust chemical engines into the structurally fragile SPS presents a problem. This problem is still under investigation. Also, the large initial mass of the SPS in conjunction with the ΔV that the chemical system is to satisfy requires a long burn time (possibly up to 24 h).

The types of electric thrusters considered in this study are the MPD, the arc jet, the ion, and the resistojet. The arc jet and the MPD thruster are considered as variations of the same plasma device and will both be categorized under the broad heading of MPD thruster. The resistojet thruster is eliminated because of its inherent low specific impulse characteristic. The ion thruster is well known, whereas the MPD thruster is not so well understood. Table 7-7 summarized the physical, electrical, and performance characteristics of an ion thruster and two MPD thruster concepts that are potential candidates for use on the SPS. The ion thruster and one of the MPD thrusters were proposed by The Boeing Company. The other MPD thruster is a Jet Propulsion Laboratory (JPL) proposal. Several versions of the ion thruster concept have been operated in space, and a 30 cm beam diameter design will be used on the SEPS. The 100 cm ion thruster presented in Table 7-7 was postulated and its characteristics were obtained by extrapolation from 30 cm thruster data. These data have high confidence. In contrast, the MPD data are speculative; only some general design-trend historical data exist on which to base its characteristics. This becomes apparent by contrasting the Boeing and JPL proposals. However, many MPD concepts have been designed and ground tested with appreciable success.

Although ion thrusters are essentially state-of-the-art and have the potential to deliver an attractively high specific impulse, MPD thrusters appear to offer many advantages when applied to the SPS. The most significant difference is the number of power supplies. A 30 cm ion thruster with a single cathode may require as many as 14 separate power supplies for operation. Each hypothetical 100 cm thruster with 10 cathodes may require as many as 31 power supplies, if the cathodes must be independently supplied. Ion thrusters cannot be ganged on power supplies because of transient interactions. Thus, each ion thruster must have its own power processing unit which, as shown in Table 7-7, can be quite massive. In contrast, MPD thrusters may be ganged in series or parallel onto a common power supply providing a vastly simpler propulsion system. Since both arcs and magnets tend naturally to operate at low voltage and high current and since there are few active components in an MPD thruster, the power supply required for operation can be very simple. MPD thrusters

TABLE 7-7. ELECTRIC THRUSTER CONSIDERATIONS[a]

Item	Boeing Ion	Boeing MPD[b]	JPL MPD
Propellant	Argon	Argon	Argon
Diameter (cm)	100	10	23
Specific Impulse (s)	18 000	10 000	2 551
Voltage (V)	10 000	300	300
Current (A)	39.8	1 958	25 000
Beam Current (A)	8.33	0	0
Thruster Power (kW)	398	587	7 500
Thruster Efficiency	0.837	0.820	0.50
Number of Cathodes	1	1	1
Number of Power Supplies	8	1	1
Thruster Mass (kg)	100	40	3 000
Power Processor Mass (kg)	199	0	0
Specific Mass (kg/kW)	3.66	1.9	1.35
Flow Rate (mg/s)	21.37	100	12 000
Thrust (N)	3.77	9.81	300

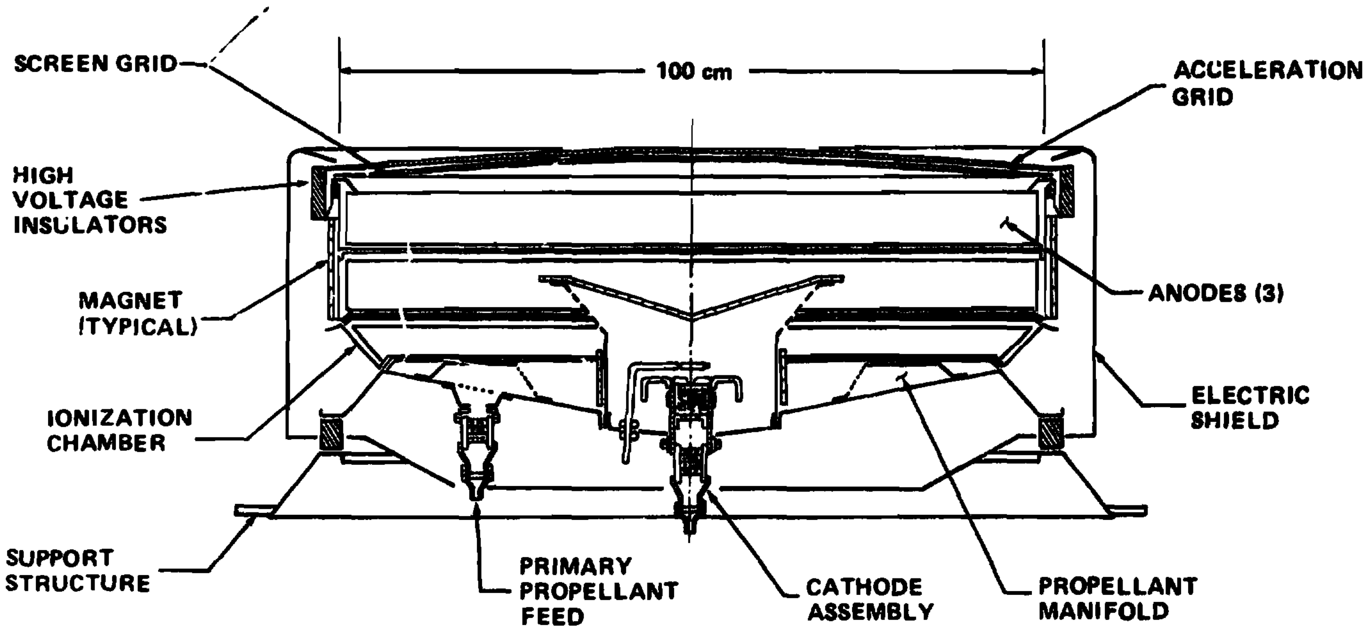
a. Data were taken from References 12 and 13.
b. Preliminary baseline selection.

can, therefore, use solar cell power directly with little power processing equipment. However, the operation of MPD thrusters at low voltages is not without penalties. High power operation requires high electrical current, which translates to heavy and/or cooled power cables. The payoff for MPD thrusters is that there is no space charge limitation as there is with ion thrusters, which means that a single MPD thruster may be able to produce orders-of-magnitude more thrust (high thrust per unit area) than any ion thruster. Of course, as more thrust is generated, the power requirement of the thruster will increase proportionally, so that cooling and associated mass penalties may be incurred. However, MPD thrusters are operable at very high temperatures, are capable of high specific impulse, and are structurally small and intrinsically simple devices that should effect a significant cost advantage. Hence, they are attractive for application to SPS.

It should be noted that several thousand electric thrusters may be required to transfer the SPS from LEO to its GEO station; therefore, in the interest of minimizing mass, and consequently cost, the propulsion system that requires the least total mass and cost would be preferred. A preliminary analysis indicates that the transfer propulsion cost is minimum when using an MPD propulsion system to transfer a photovoltaic satellite from LEO to GEO. Although selecting the optimum thruster size is not germane to this study, the Boeing MPD is selected as a preliminary baseline. Figure 7-23 presents the design concept of the 100 cm ion thruster. Figure 7-24 shows the MPD thruster design concept proposed by The Boeing Company. This thruster has an anode exit diameter of 10 cm. The outside diameter of the 40 000 A turn magnet, which can be operated in series with the cathode, is approximately 30 cm. When operating at an input power of 587 kW, this thruster does not require cooling equipment. Boeing proposed this thruster concept to serve only as a manufacturing vehicle for estimating cost. There is no technology precedent for this thruster. Figure 7-25 presents the MPD thruster design concept proposed by the JPL for comparison.

Propellants such as hydrogen, helium, ammonia, lithium, nitrogen, sodium, potassium, argon, or cesium may be used for MPD electric propulsion with some efficiency variation between each. However, environmental considerations dictate the use of argon as the propellant. Argon also has the property of good density but will have to be tanked under cryogenic conditions. The cryogenic properties of argon are similar to those of oxygen; therefore, the technology for long term storage as well as resupply of cryogenic argon will have to be established.

Several concepts for mounting the orbit transfer electric propulsion system on the SPS are under consideration. One concept involves the use of end mounted thruster modules as shown in Figure 7-26. Each module is expected to require 360° of gimbal motion per Earth orbit revolution, such that continuous tangential thrusting can be maintained. A preliminary structural analysis indicates, however, that the SPS bending loads allowed by this concept are

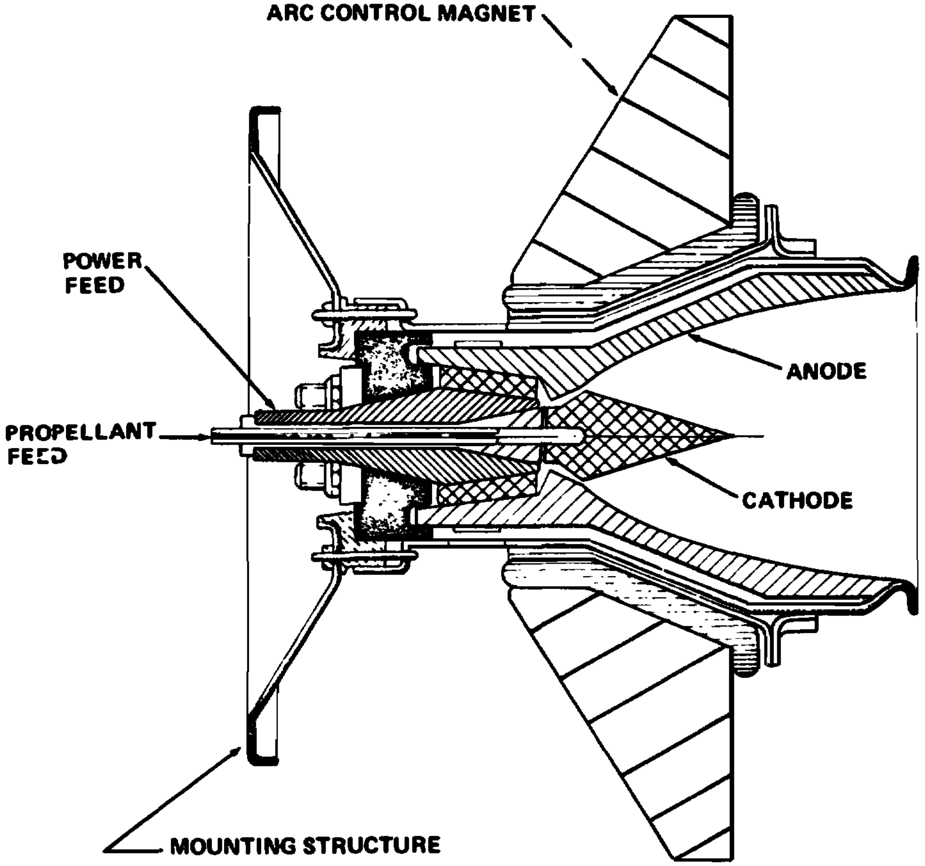


NOTES:

NEUTRALIZER(S) OMITTED FOR CLARITY

CATHODE ISOLATOR – "VAPORIZER" MAY BE REMOTELY LOCATED.

Figure 7-23. Boeing 100 cm ion thruster single cathode design concept [12].



● THRUST:	9.81 N
● WEIGHT:	40.0 kg
● SPECIFIC IMPULSE:	10000 s
● POWER:	587 kW
● EFFICIENCY:	0.82
● PROPELLANT:	ARGON
● VOLTAGE:	300 V
● CURRENT:	1958 A
● FLOWRATE:	10^{-4} kg/s
● MAGNET:	40 000 A–TURNS
● COST:	TBD (2000–20 000)

Figure 7-24. Boeing 10 cm MPD thruster design concept [12].

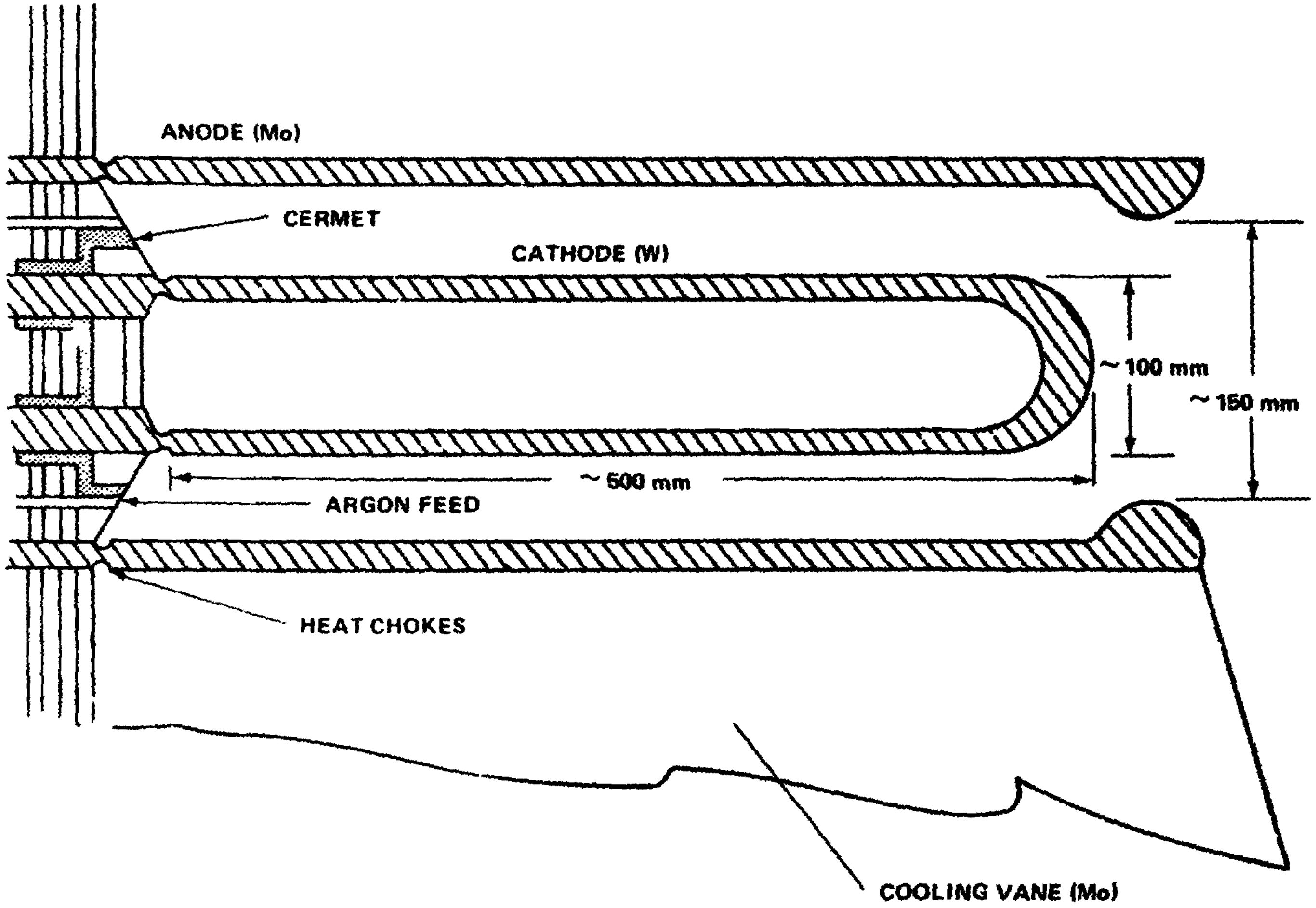


Figure 7-25. JPL MPD thruster design concept [13].

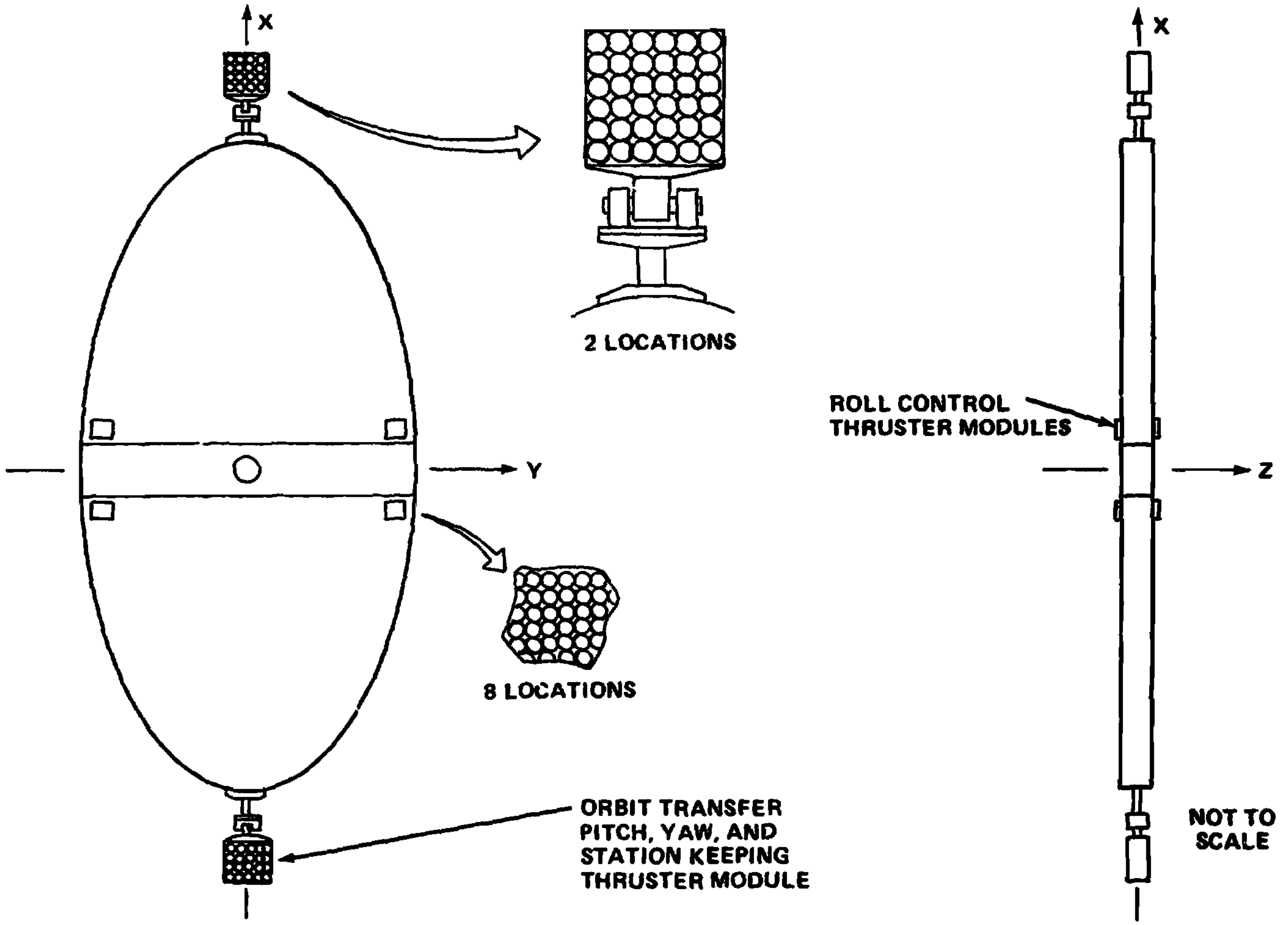


Figure 7-26. Electric thruster module locations.

excessive. Therefore, mounting of the thrusters on the back of the antenna (assuming the antenna is located in the center of the SPS) is being considered for use in conjunction with the end mounted modules. An undesirable plume impingement problem results, however, as the antenna rotates during thrusting. Another propulsion system mounting concept involves locating most of the thrusters on the back of the antenna and using smaller thruster modules around the periphery of the SPS for attitude control. Still another concept involves mounting the thrusters on each curved side of the SPS. This latter mounting concept is advantageous in that the thrust is uniformly distributed across the surface of the SPS and bending is minimized. However, this mounting concept incurs steering losses and requires undesirable on-off operation of the thrusters. The thrusters on one side operate for approximately one-half orbit and are turned off; then, the thrusters on the other side are turned on for the other one-half orbit. This process would continue until the SPS attains geosynchronous orbit.

In summary, an O_2/H_2 chemical propulsion system is proposed to maneuver the SPS from its low assembly orbit to a continuous sunlit orbit for electric thruster operation. Chemical propulsion must also provide attitude control during occultation. Based on cost, an MPD electric propulsion system is proposed to perform the SPS LEO to GEO transfer maneuver. To provide tangential thrust during transfer and to provide attitude control using transfer thrusters, end mounted SPS thruster modules, mounted on gimbals, appear necessary.

7.1.6 ATTITUDE CONTROL SUBSYSTEM

7.1.6.1 GROUND RULES/GUIDELINES

The following ground rules and guidelines are applicable to the attitude control subsystem:

- Accuracies — Attitude Control and Pointing Control
  - Main Body: ±1° in three axes
  - Microwave Antenna Pointing

    Coarse: ±1 arc min in two axes; less than 8° about line of sight

    Fine: ±3 arc s in two axes with respect to the ground rectenna center

- **An antenna gimbal system will provide coarse antenna pointing.**

- **Fine pointing and stability of the microwave beam in two axes will be provided by a phase control system.**

- **Reaction control system thrusters will provide the actuation forces for attitude control and station keeping.**

- **A pilot beam (or other means) from the ground rectenna will provide the information necessary to compute the microwave beam pointing errors.**

7.1.6.2 ATTITUDE CONTROL COORDINATES

Figure 7-27 defines the main body and microwave antenna coordinates that are to be used in the pointing and attitude control definitions and discussions. In the reference position the antenna coordinates X_A, Y_A, and Z_A are in alignment with the main body coordinates X_B, Y_B, and Z_B, where:

1. The Z_B axis is normal to the solar array and aligned to the Sun.

2. The X_B minimum inertia axis varies seasonally between X-POP and X-PEP.

3. The Y_B axis in the plane of the solar arrays forms a right hand coordinate frame.

7.1.6.3 OPERATIONAL ATTITUDES

The reference operational attitude (Sun oriented at all times) has the body axis Z_B aligned with the solar vector. At the solstices (Fig. 7-28), the X_B axis is perpendicular to the ecliptic plane, and at the equinoxes, X_B is perpendicular to the orbit plane. Between these major seasons, the attitude varies to meet the Sun pointing conditions and to maintain structural clearance for the rotating microwave antenna.

The antenna, in tracking the ground rectenna, rotates 360° per day about the X_B axis. In addition, at the solstices the antenna must tilt and roll ±23.5° each orbit. A constant tilt angle per orbit is required during the equinoxes. This tilt angle at equinox is a function of the geographical location of the rectenna. Between the major seasons, the total tilt angle varies around the equinox tilt angle.

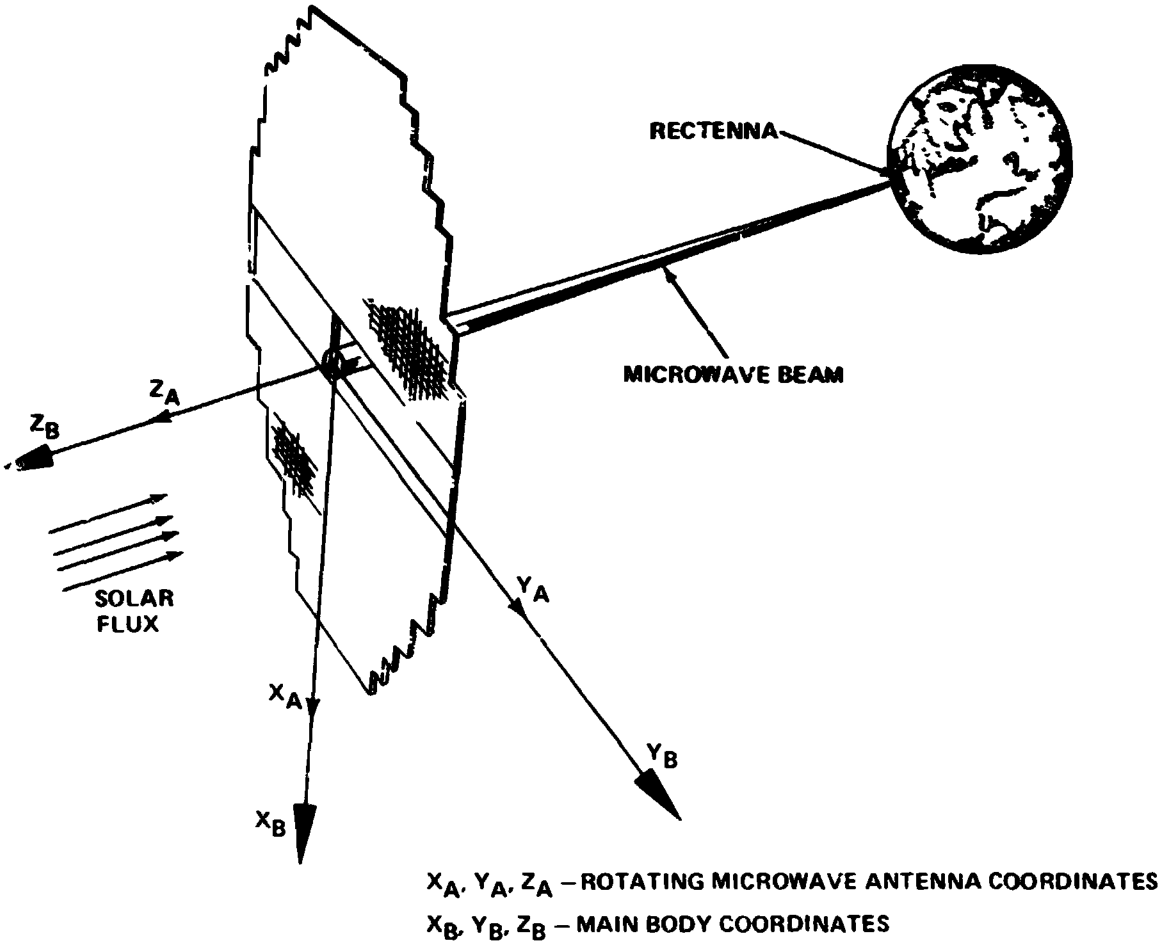
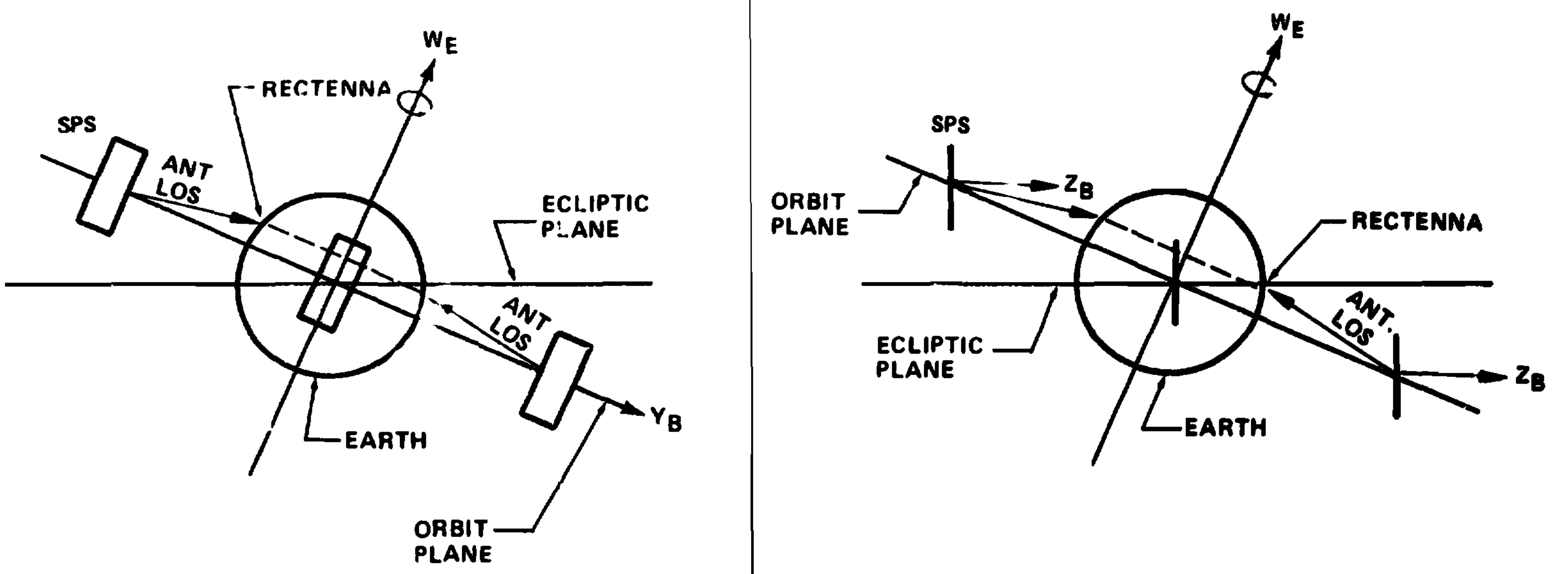


Figure 7-27. Solar photovoltaic SPS attitude control coordinates.

The ±23.5° roll of the SPS main body Z_P axis about the sunline that occurs yearly provides structural clearance for the microwave antenna as it passes through the X_B-Y_B plane during its 360° per day rotation.

An alternate operational attitude maintains the X_B axis perpendicular to the orbit plane, the Y_B axis in the orbit plane, and the Z_B axis in a sunward direction but varying through a ±23.5° yearly misalignment with the sunline. This eliminates the large per orbit antenna tilt and roll angle variations and should improve the overall pointing accuracy and simplify the antenna pointing system design. However, an increase in solar panel area is required to allow for the deviations from the sunline.



EQUINOX

(a)

SOLSTICE

(b)

Figure 7-28. Photovoltaic concept operation attitudes.

7.1.6.4 DISTURBANCE TORQUES

The external torques on the SPS, listed in the approximate order of importance, result from the following sources:

- Gravity gradient
- Antenna rotary joint friction
- Solar pressure
- Microwave antenna recoil pressure
- Aerodynamic
- Magnetic.

The gravity gradient torques are the predominant disturbances. Reaction control thrusters are used to counteract these torques. Figure 7-29 provides a plot of the peak gravity gradient torques about each of the three axes of an SPS configuration oriented in the reference attitude. The X-axis torques vary cyclically with a period of 12 h and have the peak values shown. The large Y- and Z-axis torques are secular in nature and are a result of the ±23.5° offset from the orbital plane required seasonally to maintain the Z-body axis aligned with the solar vector.

The alternate X-POP orientation greatly reduces the magnitude of the Y- and Z-torques and consequently the reaction control system propellant consumption. In this alternate orientation the magnitudes of the gravity gradient torques about Y and Z are essentially functions of the attitude control system deadband which has been assumed to be 1°.

The magnitudes of the gravity gradient torques are also direct functions of the inertia differences of the three axes. From a fuel consumption standpoint, it has been found generally desirable to minimize the inertia difference $(I_Y - I_Z)$ to reduce the cyclical torques about the X-axis.

7.1.6.5 PROPELLANT CONSUMPTION

Reaction control system (RCS) propellant consumption estimates for both station keeping in geosynchronous orbit and attitude control for gravity gradient torques of a Z-solar oriented SPS have been calculated (Table 7-8).

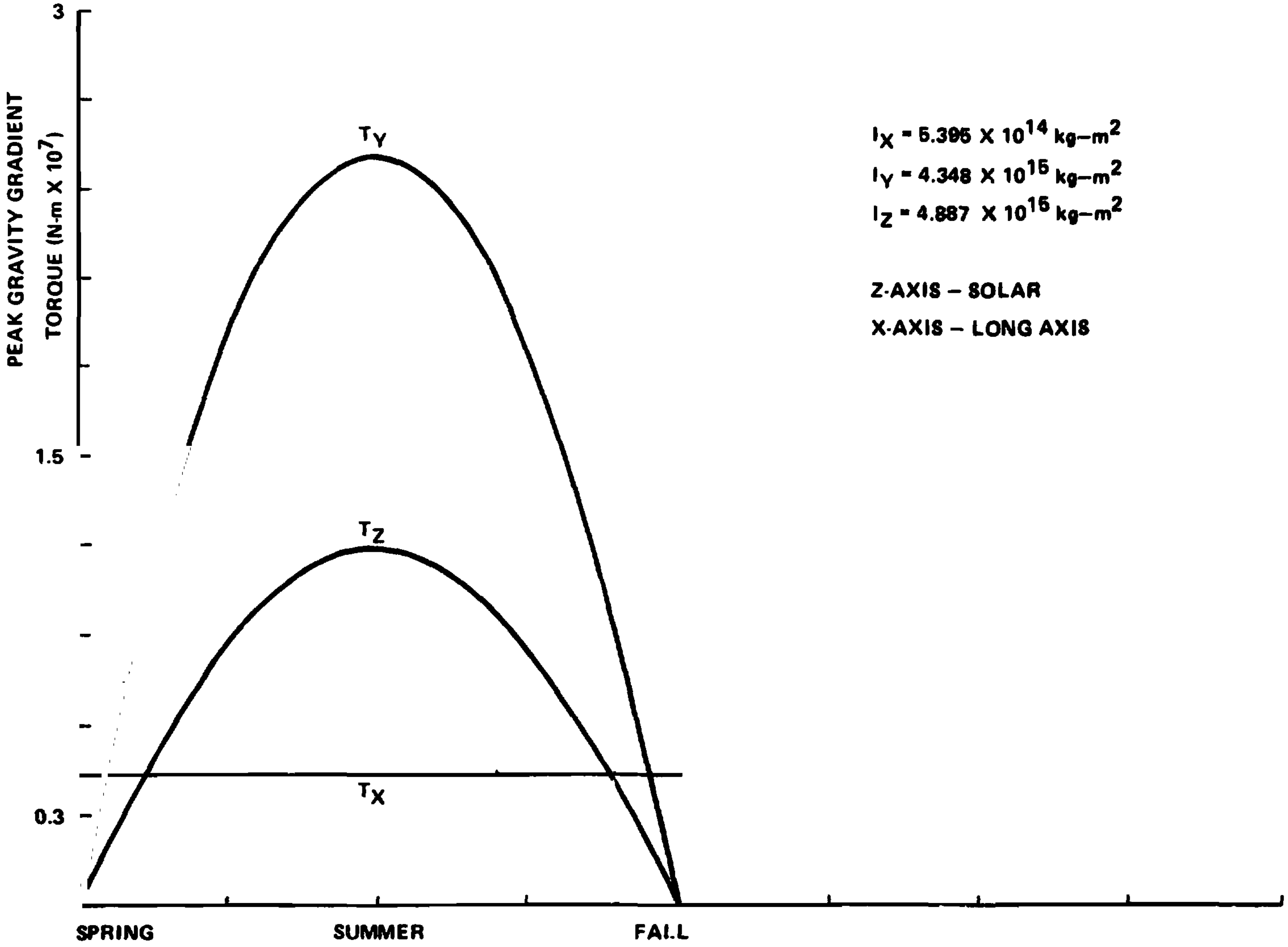


Figure 7-29. Seasonal peak gravity gradient torques.

TABLE 7-8. PHOTOVOLTAIC SOLAR POWER SATELLITE FUEL CONSUMPTIONS (Z-AXIS SOLAR ORIENTATION)

● Station Keeping		
● Solar Pressure		
— Complete Compensation		261 743
— Adjust Orbit Period Only	60 200	
● Lunar/Solar Inclination Changes	29 440	29 440
● Earth Ellipticity Drifts	4 000	4 000
Total	93 640 kg/year	295 183 kg/year
● Attitude Control Gravity Gradients		
● X-Axis Torques	219 276	219 276
● Y-Axis Torques	425 308	425 308
● Z-Axis Torques	203 138	203 138
Total	847 722 kg/year	847 722 kg/year
● Totals	941 362 kg/year	1 142 905 kg/year

Note: Specific Impulse = 714 Ns kg^{-1}.

The two total estimates differ in the concept for station keeping in the presence of solar pressure. The greater value results from complete compensation of the solar pressure by the RCS. The smaller value was obtained by assuming that the cyclical orbit eccentricity with a period of approximately 1 year was not corrected, but the more critical orbit period was corrected.

The alternate orientation of X-POP requires approximately 274 000 kg/year of propellant to compensate for gravity gradient torques.

A systems tradeoff analysis was made using the systems design analysis model for two X-POP and one Z-solar orientation concepts. In one X-POP case the array area was increased approximately 4 percent to compensate for solar angle-of-incidence losses to give an average output of 10 GW. The other X-POP case was for a minimum output of 10 GW in which the array area was increased approximately 8 percent. Output for the Z-solar case was a constant 10 GW since the array was maintained perpendicular to the solar vector. The analysis was based on the large trough concentrators with an array of elliptical planform and 2 to 5 GW antennas. For each case a concentrator angle and an array planform length/width ratio were chosen which gave the minimum total program cost. The optimum concentrator angle for the X-POP cases was larger than for the Z-solar case due to the lower average solar incidence. Also the optimum length/width ratio for X-POP was larger. As shown in Figure 7-30 the dry weight is less for the Z-solar concept. However, the propellant requirements are substantially greater resulting in a larger total 30 year mass (Table 7-9). Initial cost for Z-solar is less than X-POP primarily due to the smaller array area. However, the total cost over the 30 year lifetime accounting for the time value of money is slightly less for the X-POP case with an average output of 10 GW. The significance of these alternate orientation fuel mass trades has a great impact on the antenna pointing control system complexity as shown in Figure 7-30.

TABLE 7-9. PHOTOVOLTAIC SPS ORIENTATION COMPARISON

Orientation	Vertical Microwave Beam Motion	Antenna Rotation About X	Roll About Microwave Line of Sight	SPS Roll About Sun Required	Number of Antenna Gimbals
Z-Solar	47°/Orbit	360°/day	±23.5°	Yes	3
X-POP	0°/Orbit	360°/day	[illegible]	No	2

7.1.6.6 THRUSTER MODULE

As mentioned in subsection 7.1.5.3, the SPS will require a propulsion system to satisfy control and station keeping requirements while in LEO and GEO. Attitude control will also be required during the transfer from LEO to GEO. While in LEO, attitude control and station keeping of the SPS will be necessary during and after assembly, and the demands on a propulsion system to satisfy these requirements are expected to be considerable. Atmospheric drag makeup is expected to be the single greatest contributor to this problem. The problem of determining the requirements and sizing a propulsion system for controlling and station keeping the SPS in the LEO is complicated and is still under investigation. Therefore, a discussion primarily of the propulsion requirements for controlling and station keeping the SPS in GEO is contained here.

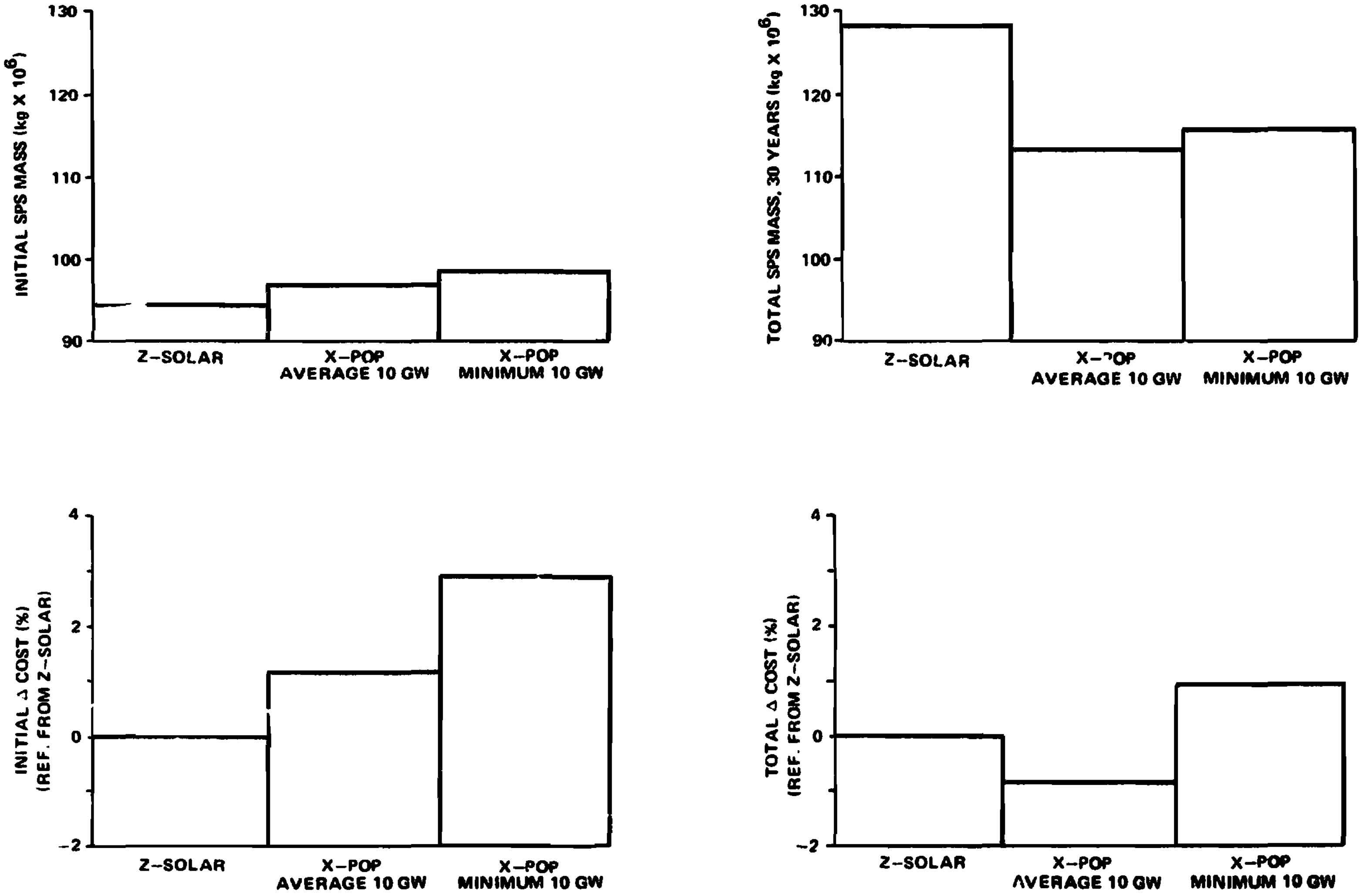


Figure 7-30. SPS study orientation comparison.

A worst case impulse requirement for controlling and station keeping a 1.16×10^8 kg photovoltaic SPS in GEO is calculated to be 8.0×10^9 kg-s per year. Thus, the yearly propellant mass required will be large. To minimize this amount of propellant, and subsequently the cost of yearly resupply, electric propulsion appears necessary for controlling and station keeping the SPS. Assuming that SPS end mounted thruster modules are baselined for the LEO to GEO transfer, these modules can also be used to provide pitch and yaw control as well as station keeping. In both cases, the number of thrusters necessary is much smaller than the number required for transfer. The remaining thrusters are considered to be spares. SPS roll control, however, will have to be provided by smaller separate modules located on the Y-axis of the spacecraft. A graphic representation of the location of the thruster modules on the SPS was shown previously in Figure 7-26. Thrusters required to produce torques about the X-axis are at four locations on each side of the SPS, while the proper thrust direction for producing torques about the Y- and Z-axes results from rotating the end mounted thruster modules.

Using each of the three electric thruster concepts discussed in subsection 7.1.5.3, a propulsion system was sized to satisfy the requirements for controlling and station keeping the SPS. The results of this analysis are summarized in Table 7-10. With gravity gradient torques being the design driver, the total number of thrusters required to counteract the disturbance torques about the appropriate axis was determined. It is realized that thruster contingency will be required for the modules used to provide roll control. The propulsion system mass is based on the specific mass of each propulsion system type, i.e., 3.66 kg/kW for the Boeing ion, 1.9 kg/kW for the Boeing MPD, and 1.35 kg/kW for the JPL MPD. The propulsion system mass consists of the sum total of the masses of such items as power conditioning units, control systems, cabling, vaporizers, actuators, isolators, thrusters, neutralizers, support structures, etc. Masses attributed to such items as propellant, propellant tank structure, propellant expulsion, valves, plumbing associated with tankage, residuals, reserves, etc., are not included. The propellant mass is based on a control and station keeping total impulse requirement of 8.0×10^9 kg-s per year (worst case). In this case, not only is the orbit period of the SPS continuously adjusted because of the influence of solar pressure, but the SPS is maintained over one point on the Earth at all times. The propellant tank mass was calculated assuming a tank mass fraction of 0.93 and that the required propellant was contained in a single tank. Propellant contingency was not taken into consideration. Items such as propellant expulsion devices, plumbing, insulation, valves, etc., are included in the propellant tank mass.

TABLE 7-10. ELECTRIC THRUSTER CONSIDERATIONS FOR PHOTOVOLTAIC SPS ATTITUDE CONTROL AND STATION KEEPING

Item	Boeing Ion	Boeing MPD[a]	JPL MPD
Number of Thrusters Required to Counteract Disturbance Torques About Appropriate Axis[b]	(X) 416 (Y) 608 (Z) 290	(X) 160 (Y) 234 (Z) 112	(X) 8 (Y) 8 (Z) 4
Propulsion System Mass (kg), Propellant and Tankage Excluded	1 914 078	564 342	202 500
Propellant Mass (kg)[c], Worst Case per Year	444 464	800 034	3 136 157
Tank Mass (kg), Including Plumbing, Insulation, etc., (Mass Fraction = 0.93)	33 455	60 218	236 055
Total Propulsion System Mass for Control and Station Keeping (kg)	2 391 997	1 424 594	3 574 712
Total Power Required per Axis (kW)	(X) 82 784 (Y) 241 984 (Z) 115 420	(X) 46 960 (Y) 137 358 (Z) 65 744	(X) 30 000 (Y) 60 000 (Z) 30 000
Approximate Minimum Thruster Planform Dimensions (Area), m × m (m^2)	(X) 13 × 13 (169) (Y) 28 × 28 (784) (Z) 20 × 20 (400)	(X) 3 × 3 (9) (Y) 6 × 6 (36) (Z) 4 × 4 (16)	(X) -- (Y) -- (Z) --

a. Preliminary baseline selection.

b. Thrusters required to produce torques about the X axis are fixed at four locations on each side of the SPS while the proper thrust direction for producing torques about the Y and Z axes result from rotating the two end mounted thruster modules.

c. Based on a control and station keeping total impulse requirement of 8.0×10^9 kg-s per year.

A comparison of the three total propulsion system masses indicates that the use of the Boeing MPD thruster results in the least total propulsion system mass required for attitude control and station keeping. Lower mass usually means lower cost; however, the yearly propellant requirement of 800 000 kg for the Boeing MPD system in comparison to 444 464 kg for the Boeing ion system over a period of 30 years could be significant and should be given further consideration.

The total amount of power needed to operate the required number of thrusters for each axis was calculated. For the roll control thruster modules, it is assumed that, while thrusting in one direction, the power to the thrusters in the opposite direction is off. However, operating electric thrusters in an on-off manner is questionable (because of lifetime considerations); therefore, power to the proper elements of each thruster may be necessary to keep the operating temperature constant. The propellant flow could, perhaps, be cut off to the thrusters that are not needed at a particular time. Hence, the power required for the roll control axis could be double that shown in Table 7-10. Since the end mounted thruster modules are required to provide control about two axes as well as maintain station keeping, power to the required thrusters in these modules is also never expected to be turned off.

Assuming the thrusters can be mounted on the SPS in a square pattern, the approximate minimum thruster planform dimensions and area were calculated for each electric propulsion concept.

7.1.7 MAINTENANCE AND REPAIR

7.1.7.1 COLLISION DAMAGE

Requirements for maintenance will arise from two primary sources: collision damage and component failure. Satellite population provides the basis for estimating expected collision damage. The increase in satellite population projected for the SPS era (Fig. 7-31) demands close scrutiny to assure that the large areas of SPS at LEO and GEO are safe from potential collision. For the large SPS spacecraft (approximately 150 × 10^6 m^2) and large satellite population, it is expected that during construction in LEO 10 collisions/month will occur, which will decrease to several per year in GEO. The spike of 20 collisions/month at 1000 km should not significantly increase the total number of hits, since the SPS will quickly pass through this region en route to GEO (Fig. 7-32).

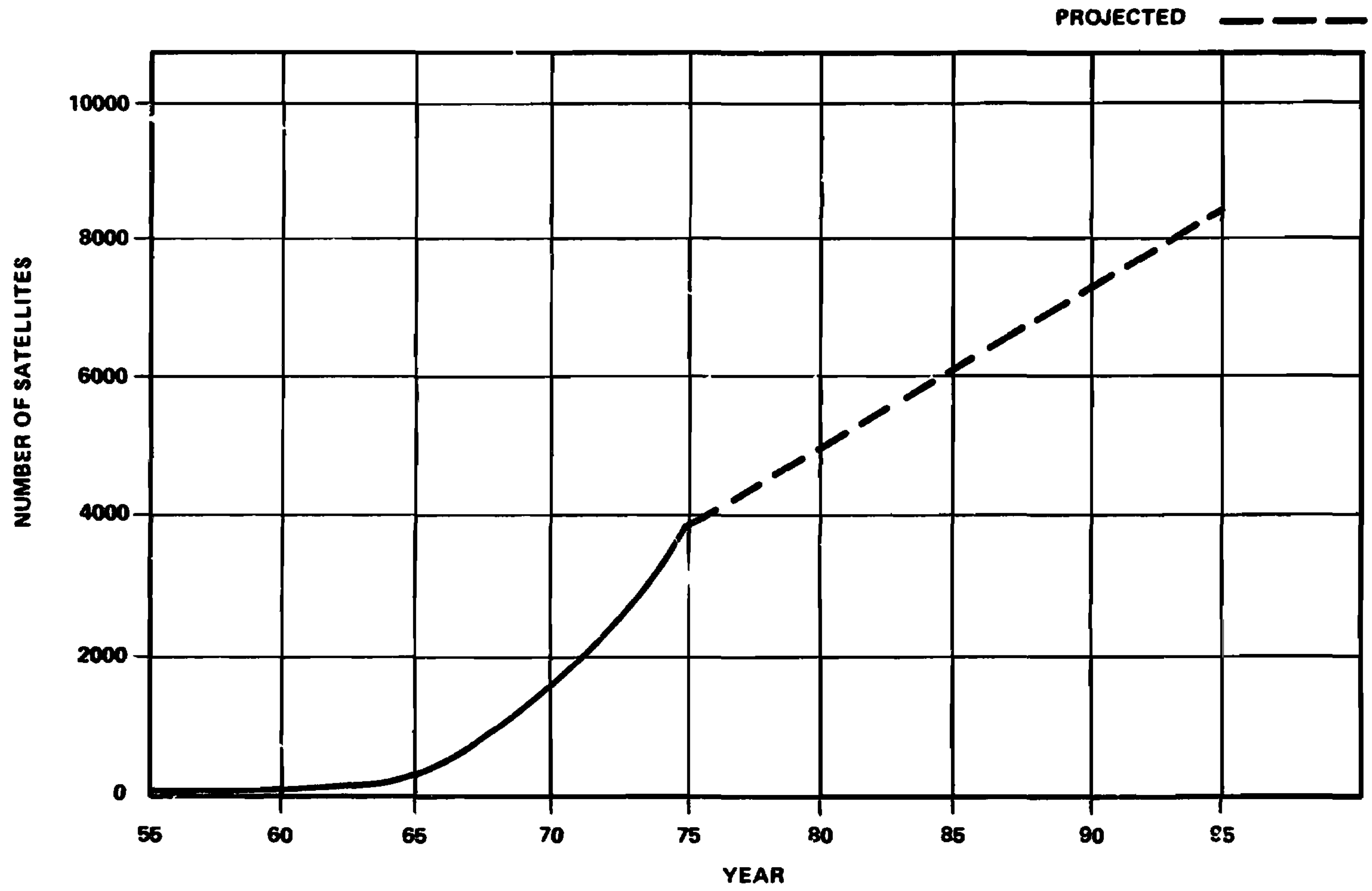


Figure 7-31. Population growth of Earth satellites.

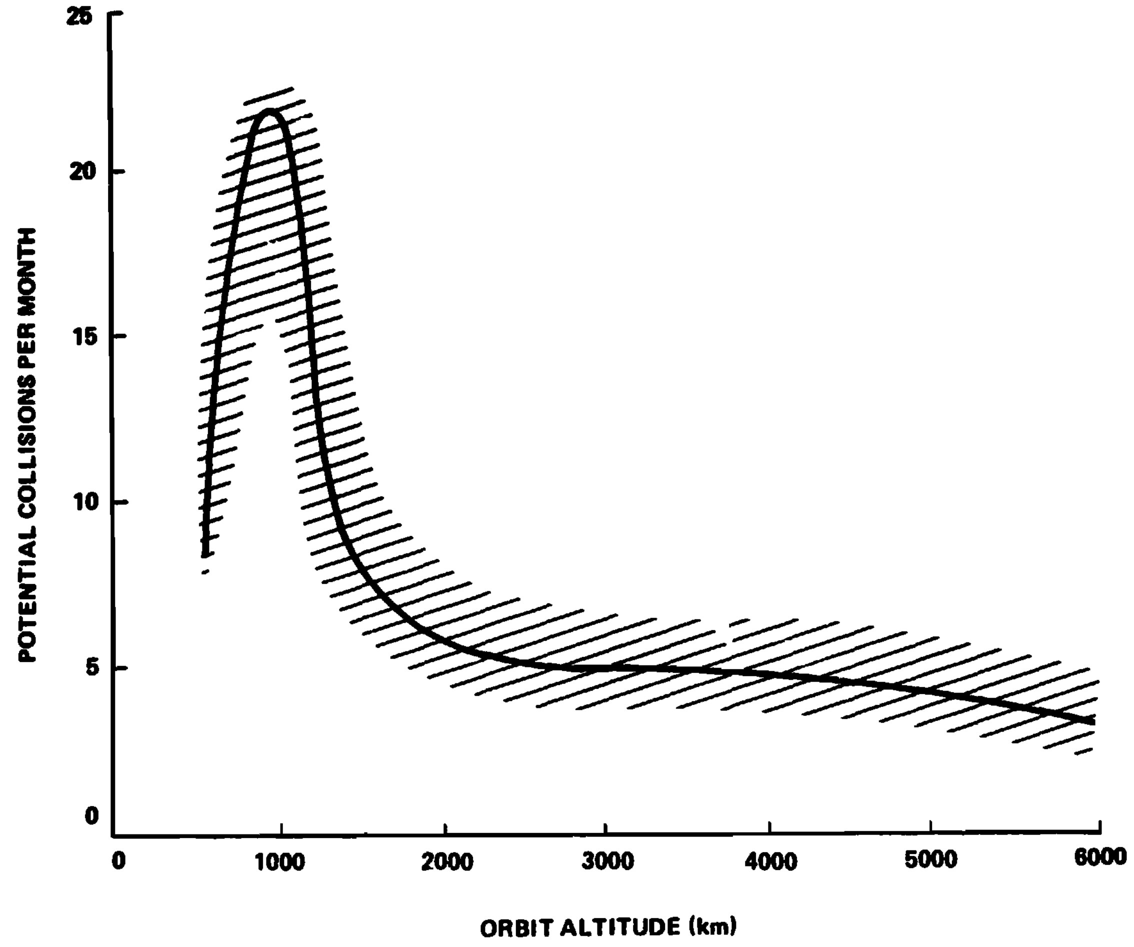


Figure 7-32. Potential collision frequency.

Severity of collision damage will depend upon what part of the spacecraft is impacted. The relative susceptibility of various subsystems to collision damage is a function of exposed planform area (Table 7-11). Note that any one collision encounter may impact more than one subsystem. Thus, the total number of impacts exceeds the total SPS collision encounters. Table 7-11 indicates that the reflectors, solar cell blanklets, and array structure are the elements most likely to sustain collision damage. These elements, however, are the most easily repaired and incur the least probable consequences of a collision. The space station and other relatively small elements would probably expect only two or three LEO impacts and only one in GEO, during the 30 year program.

TABLE 7-11. DAMAGE RATES FROM COLLISIONS

Unit	Area	LEO (500 km)	GEO
Total SPS	155×10^6	120.0	4.00
Blankets	75×10^6	58.0	1.94
Reflectors	150×10^6	116.1	3.88
Array Structure	31×10^6	24.0	0.80
Dielectric	10×10^6	8.0	0.27
Antenna	0.8×10^6	0.62	0.021
Power Mast	1×10^6	0.774	0.026
Propulsion System	3×10^6	2.40	0.08
Rotary Joint	0.08×10^6	0.062	0.002
Space Station	0.08×10^6	0.062	0.002

7.1.7.2 COMPONENT FAILURES

The number of components involved in an operating SPS will demand sophisticated maintenance philosophies even if the components are designed for exceedingly long life. For analysis of maintenance, logistics, etc., some failure rates of components must be postulated, and these are given in Table 7-12. With a 30 year operational life, amplitrons present the greatest problem. Present similar devices have a life of slightly over 1000 h or one-eighth year. The 2.5 million units of an SPS will have an expected failure rate of approximately 10 per hour. Other antenna and propulsion system elements have a lesser, but still quite significant, failure rate. If the assumed reliabilities cannot be achieved, maintenance requirements of the SPS will be more demanding.

Automatic repair facilities are assumed. One concept for antenna maintenance would consist of an automated truck that would continuously drive over the back side of the antenna exchanging entire subarray wafers having failed components with spares. When a full load of failed subarrays has been gathered, they would be delivered to an automatic repair station, exchanged for repaired subarrays, and the route continued. The automatic repair stations would disassemble amplitrons, rebore, exchange conduction bands, recoat electrodes, etc., retaining 80 percent of the original component.

Repair of the automatic repair facilities themselves would be handled by the crew if the failure rates of the 20 expected stations can be kept to approximately 20 failures per week. As shown in Table 7-13, the total crew per shift for operations and maintenance is approximately four.

TABLE 7-12. SPS MAINTENANCE

Failure Rates		**Failures**
Amplitrons	30 year Life, Approximately 4 Failures per Million Hours (FPMH) @ 2.5×10^6 Units	10 per hour
Other Antenna Elements	30 year Life, Approximately 4 FPMH @ 0.5×10^6 Elements	2 per hour
Propulsion System	25 000 hours, Approximately 4 FPMH @ 10^4 Elements	0.4 per hour
Solar Arrays	Graceful degradation over 30 years	
Rotary Joint Sections	Considering motors, brushes, electrons, etc.	Approximately 1 per month
Communication and Data Equipment	30 year Life, Approximately 4 FPMH @ 10^4 Elements	0.4 per hour
Local Transportation Equipment (LTE)	1000 hour Mean Time Between Failure (MTBF) @ 20 Units	Approximately 4 per week
Implications		
Automatic repair facilities will be required.		
Repair Facilities	200 hour MTFB @ 20 Stations	Approximately 20 per week

TABLE 7-13. SPS CREW FOR GEO OPERATIONS AND MAINTENANCE

Activities	Man-Hours/Year	
Systems Monitor and Traffic Control	8760	
Repair Station Maintenance	2 × 8760	
Other Maintenance		
Solar Arrays	1764	0.875 men/shift/year
Mast	77	
Rotary Joint	930	
Communications	624	
Propulsion	1280	
Antennas	1225	
Space Station	416	
LTE's	832	
Consumables Resupply	520	
	33 948	4 men/shift/year
Total for Three Shifts: 12 men		

7.1.8 REQUIRED TECHNOLOGY ADVANCEMENTS

The technology advancements required to build an SPS solar array as described herein are indicated by the discussions in subsections 7.1.1 and 7.1.2 and will be summarized here.

Solar cell performance characteristics will need improvement to reach the assumed range of 17 to 19 percent. This improvement must be accompanied by a reduction in mass and radiation sensitivity (particulate radiation), by a reduced power degradation over the operational life, and probably increased cell sizes, although this study has used present cell sizes as the baseline.

The support and exploitation of silicon-alternate technologies may produce more cost effective solutions to the photovoltaic conversion problem. Several candidates should receive further study and support. Development of silicon-alternate technologies should not, however, ignore the availability of raw materials. Silicon, for example, is approximately 45 000 times more abundant than gallium, and the energy economics of resource isolation must not be ignored (see subsection 9.1.3). A further study of resource and energy economics should be conducted at an early stage to assure that the selected photovoltaic system can supply enough energy to pay for its extraction from the raw material. Some simplified calculations indicate the revenue will offset the monetary investment in a silicon photovoltaic SPS after a few months of operation, but similar work for other systems remains to be done and will require access to information that has not yet been made available because of proprietary and other interests. While some studies indicate great potential savings in the energy used to refine and process silicon, these savings have not been demonstrated.

Many advanced cell concepts require special processing steps such as selective etching to texture silicon surfaces, fine geometry control for vertical multijunction cells, and complex contact metallurgy. These aspects of cell manufacture must all be examined to identify and exploit advantages in performance.

Solar cells have been used with thick, heavy glass or silica covers for protection from radiation and other damage. The SPS baseline has assumed that a very thin, light plastic can be used for this function in the GEO environment. This assumption has little evidence to support it, and much effort will be needed to verify and test light plastic covers. There is evidence that some of the silicon alternates will not require such protection; if this is proven, such characteristics may override the desirable attributes of the silicon.

Similar requirements hold for the flexible substrate, but the problem is simplified by the fact that the substrate need not be transparent. Structural and life properties of the substrate need investigation. Test and engineering data for detailed design of the supporting structures must be derived and verified.

Finally, the environmental aspects of the solar array technology selection will need study. These include material toxicity, handling, waste products, and impacts of manufacturing large quantities of solar cells and related components.

7.2 THERMAL CONVERSION SYSTEM

The SPS solar thermal concept configuration has a smaller area and a higher efficiency than the photovoltaic concept. Of the possible thermal conversion systems available, the cascaded thermionic-Brayton system is the most efficient and requires the least concentrator/radiator area and, hence, the smallest satellite planform area.

7.2.1 REQUIREMENTS AND ANALYSIS

The power requirements for the microwave antenna of the solar thermal concept are the same as for the photovoltaic concept. However, because of lower power distribution losses, the bus power required at the output of the thermal conversion equipment is less than that required at the output of the solar arrays. This was not considered in this study, however, and the bus power required of the thermal conversion system was set at 17.6 GW$_e$ for a ground power output of 10 GW$_e$. This margin, which could be up to 10 percent, can be considered as a contingency to compensate for increased pointing requirements of the reflectors or the total satellite (perpendicular to sunline instead of perpendicular to orbit plane) in the solar thermal concept.

One approach to the study of the solar thermal concept has resulted in four extremely large power modules to produce the 17.6 GW$_e$ required bus power. Component sizes are attendantly very large. Liquid metal radiators are required that must be assembled and welded on-site. It is expected that the on-orbit assembly and maintenance equipment could be smaller and have a larger utilization factor if a large number of smaller power modules were used. A small power module will likely be required for the early phases of the SPS development program, and the impact of stopping the development cycle at a small size was assessed. Advantages of a small module include:

1. Absorber/receiver/conversion system (excluding radiator) can be ground assembled and remain a high density launch package.

2. Surface and pointing accuracy of reflector surface (due to the shorter focal length) are reduced.

3. Redundancy is provided (i.e., several modules may fail before a significant SPS power loss is experienced).

4. One turbomachine per absorber/receiver allows a power module to be shut down for maintenance or repair without affecting the operation of other turbomachines.

5. Primary radiator fluid distribution line and manifold requirements are reduced.

6. Integration into a shallow, planar, modular structure is allowed.

7. Smaller more manageable components are used.

8. "Boot-strap" startup is allowed with small start up energy storage requirements.

The disadvantages of a small module include:

1. Weight-to-power ratios are higher.

2. There is a lower Brayton cycle conversion efficiency for a given ΔT.

3. The small module is incompatible with the two-axis orientable "flat-facet" concentrator approach.

After the baseline was initially established with 544 turbomachine/absorber modules, an analysis was conducted to study the effect of variations in the number of modules for a 10 GW SPS. There are certain advantages and disadvantages for a small number of modules as shown in the previous list. All of the factors listed were not quantified, but data were generated or obtained for most of the primary items. Factors that were considered include turbomachinery weight and outage time as a function of turbomachine generating capacity. Historical data indicate that the time to repair or perform maintenance for large turbomachines is greater than that of smaller units. Absorber weight was analyzed based on the unit size. Since the radiator requires the same area to dissipate a given amount of heat for small or large radiators, the effect of radiator weight was neglected. Total SPS weight is shown in Figure 7-33 as a function of the number of turbomachines for variations in the number of turbomachines per absorber. The discrete points are indicated by points on the curves for more than one turbomachine per absorber. For example, if there are 40 turbomachines per absorber, the total number of turbomachines will vary in increments of 40 as 40, 80, 120, etc. A preliminary assessment indicates that

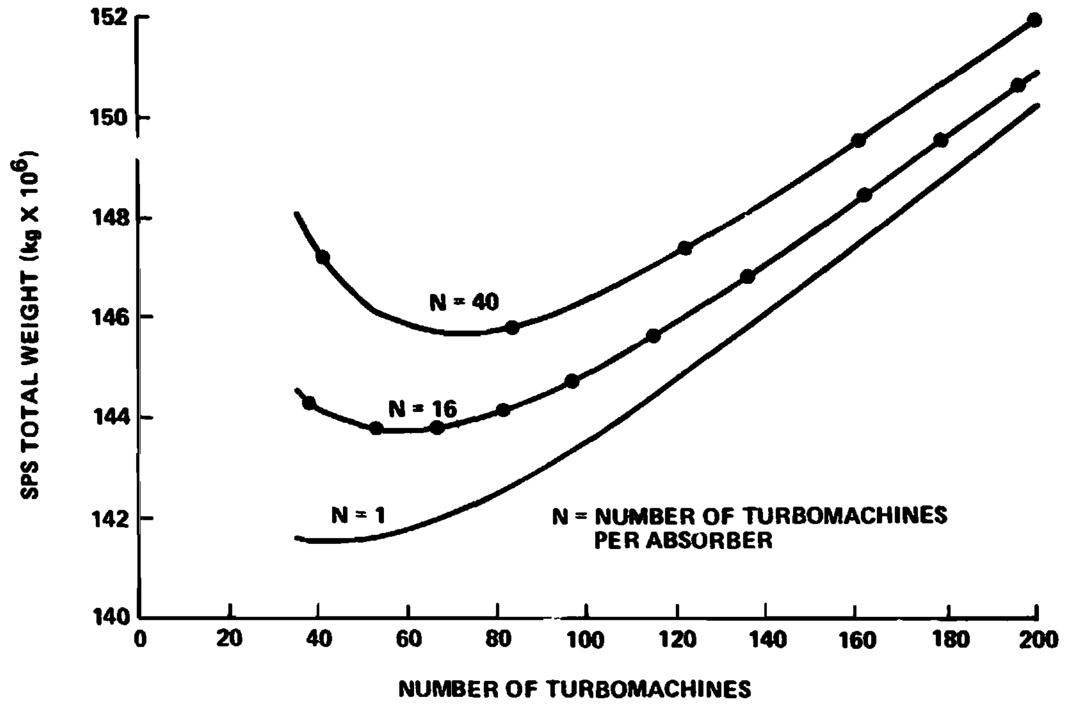


Figure 7-33. Optimum number of turbomachines per absorber.

40 to 50 turbomachines would be more optimum than the current 544 units. Also, there are indications that it may be somewhat advantageous to have a small number of turbomachines per absorber. This secondary conclusion could change as the pointing requirements for the concentrators are further studied. This analysis shows that the total weight of the thermionic-Brayton SPS could be reduced by approximately 20 percent primarily because of reductions in the turbomachinery weight.

The thermionic-Brayton cascaded system has merit only if the thermionic diodes can operate at temperatures well above the maximum possible Brayton turbine inlet temperatures. For 1985 technology, a Brayton turbine inlet temperature limit of 1026°C was projected. The performance efficiency of the thermionic diode projected for the same time period was 17 percent at an emitter temperature limit of 1826°C and a collector temperature of 1026°C. The Brayton cycle conversion efficiency is approximately twice that of the thermionics for approximately the same ΔT. If higher Brayton turbine inlet temperatures could be allowed (up to 1426°C), the complexity of adding the thermionics in a cascaded system could not be justified on a performance basis. However, with the projected 1985 technology and the proposed hybrid thermionic/ Brayton configuration, an effective overall cycle efficiency of 45.5 percent can be achieved for the hybrid system, compared to 36.7 percent for a Brayton cycle alone. This results in approximately 20 percent less thermal output required from the concentrator/absorber and 30 percent less primary radiator area required for the hybrid system in comparison to a Brayton only system.

7.2.2 HYBRID THERMIONIC-BRAYTON SYSTEM DESCRIPTION

Some recent studies [7] have based the concentrator design on the use of several thousand orientable "flat facets" that reflect the Sun's image to a common point, a classical "black body" cavity. The cavity is shared by several Brayton engines. Smaller modules, double curved "spherical facets," a light pipe selective surface absorber, and a single thermionic-Brayton cascaded conversion system coupled to each absorber were examined in this study. The unique characteristics of this concept will become apparent in the following description.

Figure 5-4 depicts the overall thermal SPS concept. The structure is a hexagonal honeycomb composed of struts and cables with the absorber/receiver located on one face of the hexagon and the reflector facets mounted across the opposite face. Each of the 37 required facets is assumed to have a ring around the perimeter that supports a thin shell reflector. Two facet types may be used, a pure reflector or a combination reflector/radiator facet. While a pure reflector facet surface may be a highly polished metal or metallized surface with low emissivity, the reflector/radiator facet must have a specular reflecting, low absorptivity, high emissivity surface, such as rear metallized teflon or acrylic film, or a metal surface with a silicon monoxide overcoat. One approach to this construction of the reflector facet uses the ECHO II technique of a laminate of polyester film and aluminum foil. In forming, the aluminum is stretched, and thereby work hardens, and results in a rigid thin shell. The baselined radiator is a conventional tube-fin radiator. The definition of a lightweight radiator with a liquid manifold about the perimeter and heat pipe distribution throughout a thin shell reflector/radiator was deemed beyond the scope of this study. The 37 facets have a curvature and are oriented to have overlapping images at the absorber/receiver location. Each facet has a concentration ratio of approximately 56, and the combination of 37 results in a concentration ratio or more than 2000.

A block diagram and energy balance for the system is shown in Figure 7-34. Characteristics are given for one of the required 544 modules. The dc-ac rotary converter (not shown) is a necessary part of the thermionic diode power circuit before the energy goes onto the satellite power bus. The reflector efficiency is assumed to be 90 percent with silver or an enhanced

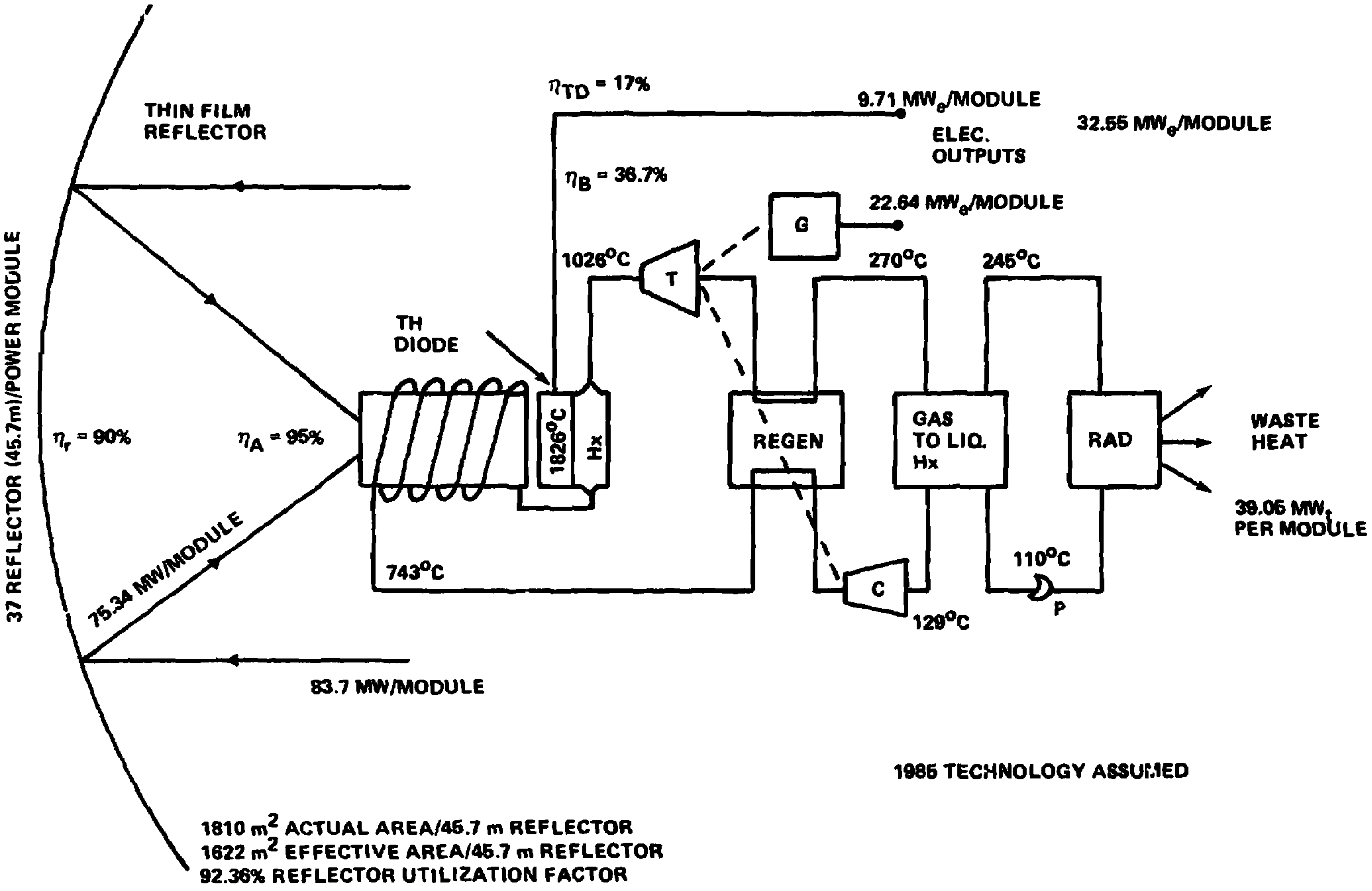


Figure 7-34. Thermionic-Brayton solar thermal concept with light pipe selective surface absorber.

reflectivity coating. The light pipe selective surface absorber has 95 percent efficiency. The light pipe section has an inner surface, such as silvered quartz or polished metal with a silicon monoxide coating, that has low solar absorptivity, high specular reflectance, and high emissivity. High absorptivity and low emissivity characteristics are desirable for the base region.

It is unknown whether such properties can be attained at thermionic temperatures with tungsten or other pure metals, or whether such performance can be attained with optical filters. If the emissivity of the base region increased from the 0.2 assumed, the energy passing through the thermionic diodes would be reduced but the losses to the outside of the absorber/receiver would not be increased significantly. The thermionic diodes receive all the energy except the 5 percent cavity loss, the 20 percent that is absorbed by the light pipe from the visible spectrum on its way toward the base region, and that which is absorbed by the light pipe from the infrared spectrum on its way from the base region toward the mouth of the cavity. The Brayton cycle receives all the thermal energy except the 5 percent cavity loss and that which is converted to electricity in the thermionic diodes. Conduction losses through the sidewall insulation have been assumed to be negligible. Table 7-14 is · summary of the characteristics of this system.

7.2.3 THERMAL RADIATOR SUBSYSTEM

Radiator location is a primary consideration since the radiators are large and comprise a high percentage of the total mass. Considering the options available, three locations were studied, each of which had specific advantages. These options were assessed to determine if any one had an overriding advantage. The locations assessed are indicated in Figure 7-35.

Options 1 and 2 offer the smallest integration problem since the feed lines are in the vicinity of the absorber and turbomachinery. Option 2 seems to have an advantage over option 1 because part of the inner truss structure could be common with the radiator support. However, the radiator sizing is made difficult by option 1 because the view factor degrades as the required depth is increased. Because of the view angle from the reflector, the available radiator width using option 2 is much smaller. Using option 1 a 360 m width is available, while option 2 provides, at maximum, 210 m; thus, the radiator height has to be greater in option 2 than in option 1. As a result, the feasibility of option 2 is questionable.

TABLE 7-14. SUMMARY CHARACTERISTICS OF SOLAR THERMAL CONCEPT WITH SPHERICAL REFLECTOR FACETS AND THERMIONIC-BRAYTON CONVERSION SYSTEM

- Spherical Surface, Hexagonal Shaped Reflector Facet
  - 45.7 m across flats, 1810 m^2
  - Intercepted sunlight with 92.36 percent reflector utilization factor, 2.26 MW_t at 1353 W/m^2
  - With reflector efficiency η_r = 90 percent and absorber efficiency η_A = 95 percent an average facet produces 1.93 MW_t
  - Thermionic conversion efficiency is 17 percent
  - Brayton conversion efficiency is 36.7 percent

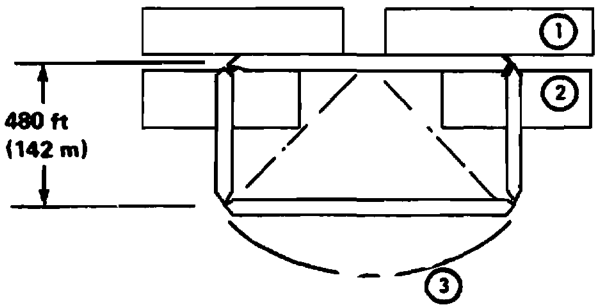
    $\Big\}$ Effective total efficiency is 45.54 percent
  - Thermal input available to thermionics is estimated at 1.93 MW_t - 20 percent (1.93) = 1.544 MW_t
  - Thermal input available to Brayton is estimated at 20 percent (1.93 MW_t) + 1.544 MW_t - 17 percent (1.544) = 1.6675 MW_t
  - Electrical power output of thermionics is 17 percent (1.544 MW_t) = 262.48 kW_e/facet
  - Electrical power output of Brayton is 36.7 percent (1.6675 MW_t) = $\underline{611.97}$ kW_e/facet
  - Total electrical power/facet is .... 874.45 kW_e
- For 17.6 GW_e at Generator Terminals, 20 127 Facets are Required; with 37 Facets/Module, 544 Modules are Required

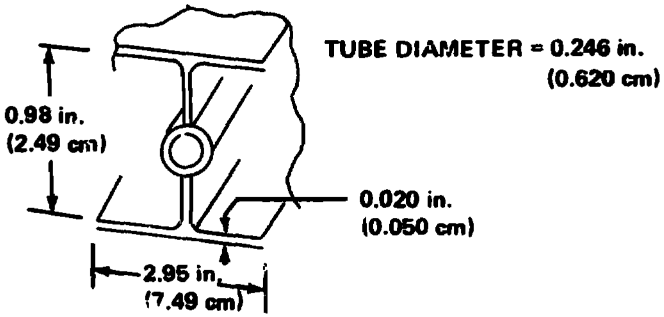
TABLE 7-14. (Concluded)

- Conversion Cycle Heat Rejection Required is 1.93 MW_t - 0.87445 MW_e = 1.0555 MW_t + Solar Input to Radiator
    - Assuming $\epsilon = 0.75$, $\alpha = 0.10$ for Sun side and $\epsilon = 0.90$ for space side of radiators; 177°C effective radiator temperature. (The required radiator area is 950 m^2 or 478 m^2 of a two sided radiator.)
    - For 37 facet module, 10 facets must be a combination reflector/radiator
- One Thermionic Module/Brayton Engine on Each Module
    - Each engine/generator output is 22.64 MW_e high voltage ac
    - Each module thermionic array output is 9.7 MW_e low voltage dc

CONFIGURATION		VIEW FACTOR	SURFACE EMISSIVITY	REQUIRED AREA PER MOD, ft^2 X 10^3 (m^2 X 10^3)	RAD. MASS[6] PER MOD, lb X 10^3 (kg X 10^3)	NAK MASS[2] IN TRANS. LINE, lb X 10^3 (kg X 10^3)	NAK RAD.[4] MASS, lb X 10^3 (kg X 10^3)	RAD. SYS. MASS, lb X 10^3 (kg X 10^3)	FLOW[5] POWER, %
EXTERNAL ①		0.5	0.88	475 (44)	220 (100)	– (3)	17 (7)	237 (110)	– (3)
INTERNAL ②		0.45	0.88	525 (53)	242 (110)	– (3)	18 (8)	260 (120)	– (3)
③ INTE-GRATOR	SUN SIDE	0.90	0.78[1]	289 (27)	133 (60)	176 (80) 12 in. (30.48 cm) LINE 24 in. (60.96 cm) LINE 314 (142)	10 (5)	319 (140) 457 (210)	0.8 0.2
	ANTI-SUN SIDE	1.0	0.88						

1. α = 0.11
2. 1920 ft (584 m) LINE
3. NOT DETERMINED
4. 0.035 lb/ft^2 (0.17 kg/m^2)
5. RADIATOR FLOW POWER NOT INCLUDED
6. 0.4625 lb/ft^2 (2.25 kg/m^2)





BOEING'S OPTIMUM RADIATOR PANEL WITH TUBE ARMOR

Figure 7-35. SPS radiator location trade, tube-fin construction.

Option 3 offers potential advantages since the radiator can be an integral part of the facet structure. However, there are two major problems. The first is the NAK mass between the radiator and the turbomachine. The NAK alone can weigh more than 136 000 kg, depending on the allowed flow losses. This mass is greater than the radiator mass for option 1. Secondly, the facet surface is not continuous, having NAK line connections between the facets. This study has not revealed any mass advantage for option 3.

Based on the study to date, option 1 is the best choice. The radiator will consist of two panels that are 180 m long and approximately 60 m high. A summary of the mass assessment for the three options is presented in Figure 7-35.

These conclusions are based on an inlet temperature of 245°C, an outlet temperature of 110°C, and tube-fin type construction. The SPS concept configuration does not complement the radiator size, since the radiators have to be structured relatively close to the main structure. With the exception of option 3, the view factor can never be greater than 0.5. Nevertheless, minimum combined radiator/support structure mass is represented by a combination of options 1 and 2.

The radiator size is very sensitive to its inlet and outlet temperatures. Some consideration has to be given to the effect of turbomachinery efficiency on radiator temperature. The sensitivity of radiator size to the inlet temperature is illustrated in Figure 7-36. If the inlet temperature is too low, the entire system suffers a severe mass penalty. Since total SPS cost is most sensitive to mass, tradeoffs are necessary between turbomachinery performance and total system mass. For this study a design point, as indicated, was selected. Optimization of the radiator mass and area on a cost basis has not been completed.

7.2.4 POWER DISTRIBUTION AND CONTROL SUBSYSTEM

Based on a preliminary analysis, an operating voltage of 20 kV dc was selected for the thermionic-Brayton system. Low voltage dc power from the thermionic diodes is converted to 20 kV by a motor-generator set or an equivalent static converter. The output of the thermal engine is converted to 20 kV dc with rotating generators. Thermionic-Brayton modules producing 32.25 MW are connected through circuit breakers in groups of 40 to 16 main power lines. Thin wall aluminum tubes are used for all power conductors.

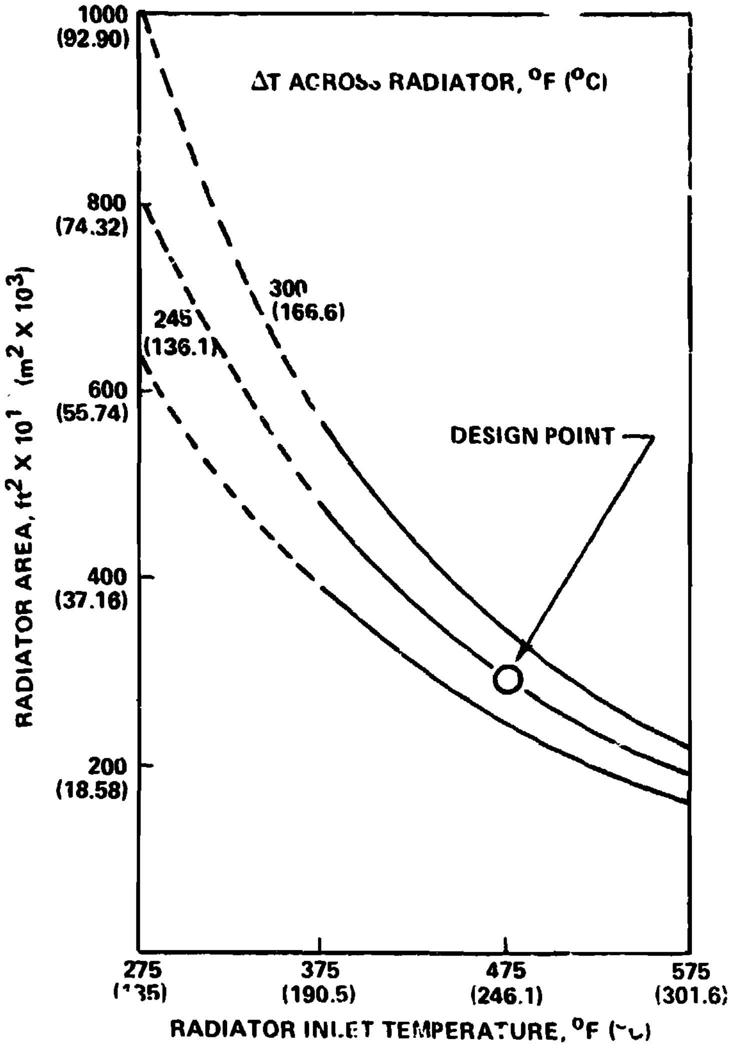


Figure 7-36. Radiator sizing sensitivity for option 1 location.

Generator mass required for 14 GW output from the thermal engine is a major concern. Even with a specific mass of 0.453 kg/kW, the mass would be 6.3 × 10^6 kg. However, this specific mass appears to be an achievable goal for the SPS. The projection is based on Atomic Energy Commission studies [17] for liquid metal cooled machines, on work underway by Wright-Patterson Air Force Base on machines for laser and radar applications, and on large commercial generators. Specific weights below 0.9 kg/kV A were defined through point designs in an Atomic Energy Commission funded study, while Wright-Patterson had demonstrated short term operation of 0.045 to 0.113 kg/kV A machines [16]. Commercial water-cooled generators are available with specific weights of 0.335 kg/kV A at 1500 MW ratings.

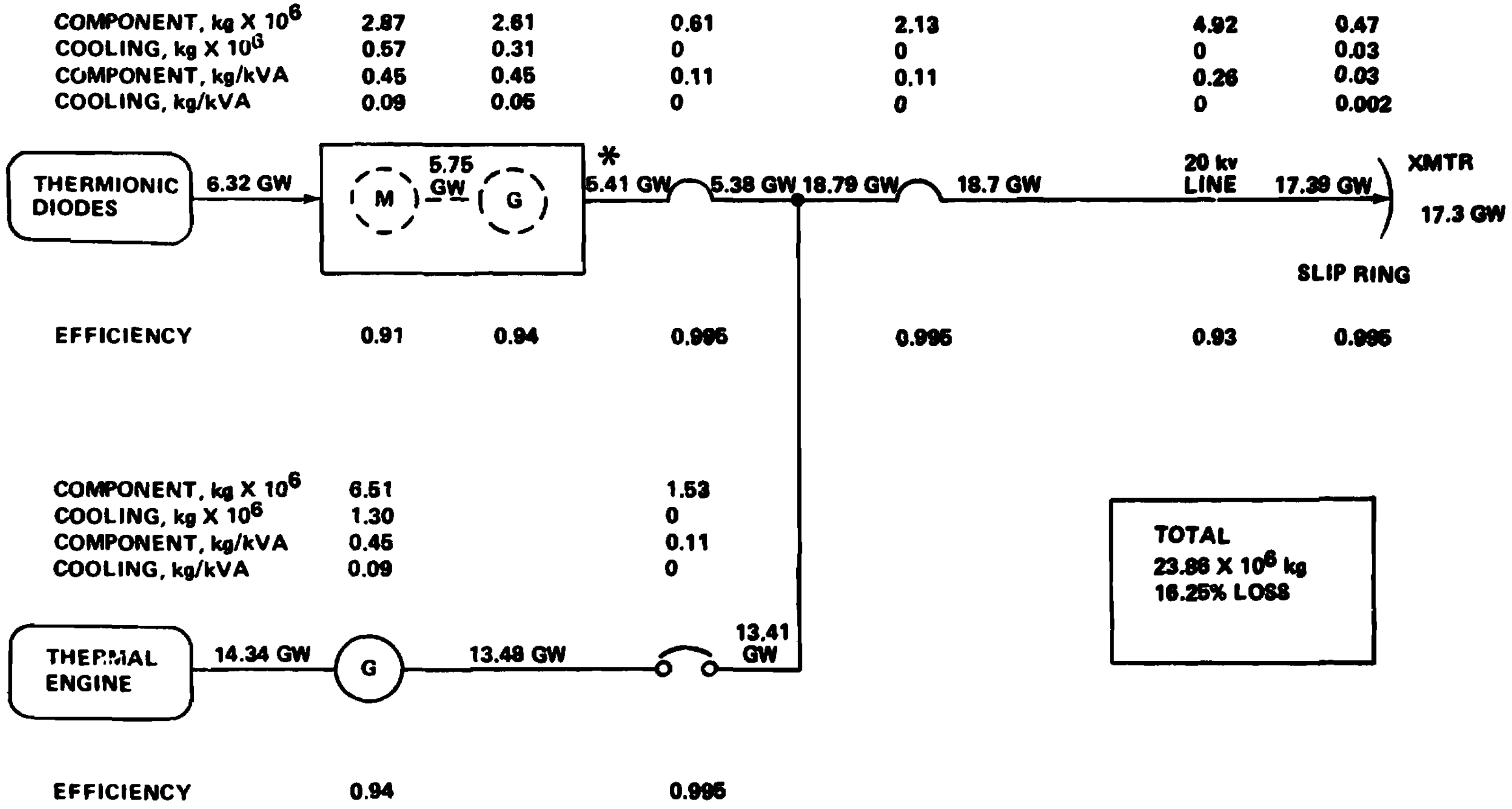
The efficiency chain for the system postulated for the SPS is shown in Figure 7-37. Estimates of component masses are included in the figure. In comparison to the solar array distribution system, power loss was approximately the same but system mass was doubled for the thermionic-Brayton system.

A 400 kV, 400 Hz transmission system was studied for comparison. Step-up transformers were added at the power sources to increase the voltage level to 400 kV for transmission. The slip-ring assembly used for the baseline dc system was replaced by a rotary transformer/rectifier circuit to provide 20 kV dc for the transmitter tubes. Power line mass and loss were reduced considerably from the 20 kV dc system, but the advantage was lost through added requirements for transformers and rectifiers. As a result, the system power loss for the two systems was equal but the mass of the 400 kV ac system exceeded the mass of the 20 kV dc system by 9 × 10^6 kg.

7.2.5 MAINTENANCE AND REPAIR

Collision damage and component failure will be the sources of maintenance requirements for the thermal conversion system just as it is for the photovoltaic system. Since the total area is approximately one-fourth that of the photovoltaic system, the collision rates should be proportionally smaller. The consequences of impacts on the concentrators are more significant, however, in view of the maze of plumbing, valves, radiators, etc. Fluid loss could become significant even if individual reserves are provided for each concentrator cell.

Plumbing component failures will probably be more serious than amplitron failures. Automatic installation of plumbing spares will be required to maintain operation. Automatic plumbing maintenance will demand innovations and a designed maintainability philosophy. Automatic maintenance of the antenna systems is as described in subsection [illegible] 7.2.



* ROTATING MACHINERY OR STATIC POWER CONVERTERS

Figure 7-37. Power distribution efficiency chain, thermionic-Brayton.

7.2.6 FLIGHT MECHANICS

7.2.6.1 STATION KEEPING

Station keeping requirements for the SPS thermal conversion concept will basically be the same as those for the photovoltaic concept, described in subsection 7.1.5.1. Both concepts will be subject to the same type of disturbances, although the magnitude of the aerodynamic and solar pressure perturbations will be different because of different areas and masses. A separate analysis has not been conducted for the thermal conversion concept.

7.2.7 ATTITUDE CONTROL SUBSYSTEM

7.2.7.1 GROUND RULES/GUIDELINES

For the thermal conversion system, the ground rules and guidelines are identical to those given in subsection 7.1.6.1. Here, of course, the Z_B body axis in the reference attitude is normal to the plane of the concentrator array.

7.2.7.2 ATTITUDE CONTROL COORDINATES

The attitude control coordinates are identical to those given in subsection 7.1.6.2, with the plane of the solar array replaced by the plane of the concentrator array.

7.2.7.3 OPERATIONAL ATTITUDES

The reference operational attitude discussion presented in subsection 7.1.6.3 for the photovoltaic conversion system also applies to the thermal conversion system. However, the alternate operational attitude with X_B perpendicular to the orbit plane is not acceptable with fixed facets in the solar concentrators, and the Z_B axis of the thermal conversion system must be maintained in alignment with the solar vector to an accuracy of ±1°.

7.2.7.4 DISTURBANCE TORQUES

The gravity gradient torques are the predominant external disturbances. Peak values of the gravity gradient torques are:

$$T_X = 9.76 \times 10^5 \text{ N-m}$$

$$T_Y = 3.35 \times 10^6 \text{ N-m}$$

$$T_Z = 1.44 \times 10^6 \text{ N-m} \quad .$$

The thermal conversion system contains a large number of rotating machines. If all the machines are oriented to have the individual momentum vectors additively aligned, a total momentum of approximately 6.14×10^8 N-m-s is present on the SPS. A brief study was made of the effect of this state on an uncontrolled SPS in the presence of gravity gradient torques. The three cases studied were for zero momentum, nominal momentum, and a momentum 1000 times the nominal. The results indicate that the SPS performed identically with the zero and nominal momentum values. Therefore, no special machinery orientation is required to provide cancellation of the individual momentum vectors. The large momentum case provided a significant increase in the stability of two axes because of gyroscopic stiffness. Also, the large momentum case indicated the need for a control law that conforms to gyroscopic torque relations. Figure 7-38 provides plots of the angular rotations versus time for each of the cases studied.

7.2.7.5 PROPELLANT CONSUMPTION

For thrusters having a force of 6.85 N each and a specific impulse of 714 Ns kg^{-1}, the propellant required to counteract gravity gradient torques is 3.35×10^5 kg per year.

7.2.8 REQUIRED TECHNOLOGY ADVANCEMENTS

Technology advancements are required on each major component of the solar thermal SPS.

7.2.8.1 CONCENTRATOR

The last solar concentrator technology work was done in the 1960's for the Sunflower program [20] and the solar Brayton (LeRC) work. Concentrators were made of foam-backed, aluminized, polyester film "Japanese Fan" with petals of ALZAK aluminum and rigid glass substrate mirrors. Sizes up to approximately 9 m in diameter were constructed and tested. None of these approached the lightweight requirements of SPS. Some work was done by Goodyear in the same time frame on lightweight inflatable antennas approximately 9 m in diameter. Ground tests, but no flight tests, were performed. The closest

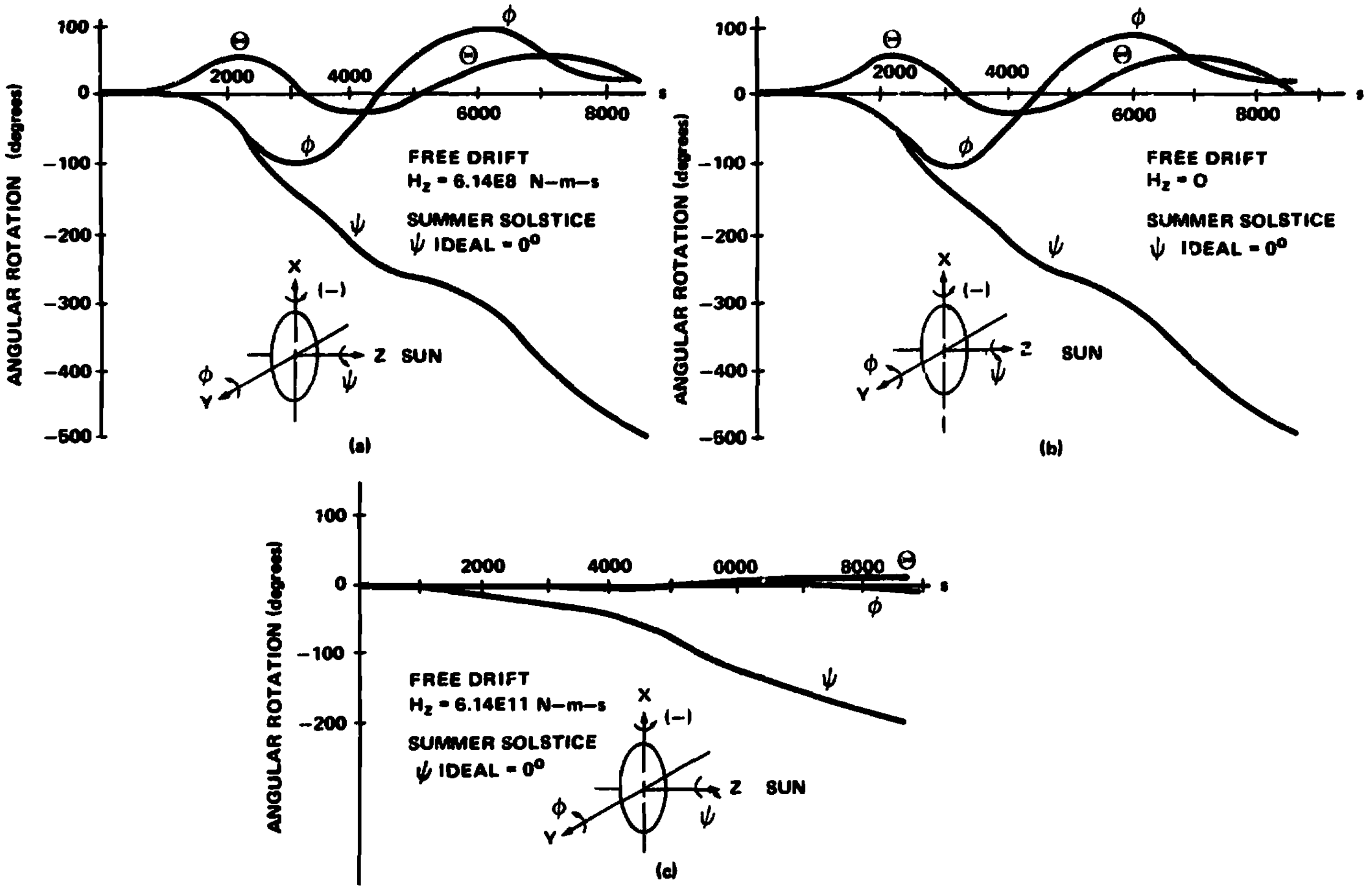


Figure 7-38. Solar concentrator SPS rotating machinery momentum effects on uncontrolled attitude.

configuration to a large area, lightweight, reflecting structure having a weight-to-area ratio comparable to the SPS that has been flown was the ECHO II 41 m diameter, gas pressure inflated, rigidized sphere.

It has been suggested that a better low curvature reflector, as required for SPS, would be one made up of a composite, laid up on a mold on the ground, and impregnated with an ultraviolet curing epoxy. The structure would be gas pressure deployed and would be allowed to cure in the ultraviolet space environment.

One proposal is to create the required concentration by overlapping the images from thousands of flat facets. Construction of the individual facets can be simpler than construction of the double curved reflectors usually used. A large number of facets are required for a high concentration ratio. Large module sizes are desirable to reduce the total number of facets required to collect a given quantity of energy. This concept does not scale readily to small power module sizes.

7.2.8.2 ABSORBER

The absorber required for the solar thermal version of SPS has not been developed beyond small scale (small aperture) test modules. A solar absorbing cavity was tested as part of a solar Brayton program in the 1960's [20]. This program did not progress beyond the component test phase. Present efforts to develop large solar absorbing cavities are concentrated on ground solar power applications. The operating temperatures of the ground systems will be lower than those required for SPS, but the analytical tools developed will be applicable to the space system analysis. Small laboratory models of light pipes have been built and used to provide uniform illumination in laboratory testing [21-23]. However, a light pipe, with the selective surface properties required for SPS, has never been constructed.

7.2.8.3 THERMIONIC-BRAYTON CONVERSION SYSTEM

The conversion system equipment technology required for this system is similar to that required in other space and ground power systems. Consequently, a reasonably consistent technology development program already exists for the basic components. Thermionic diodes for use with a reactor source and Brayton equipment for use with isotope or reactor sources have been under development for several decades. Extensions from this base required for SPS encompass increasing the operating temperature and efficiency of the Brayton equipment and increasing the efficiency of the thermionic doides. The diodes are restricted from using tungsten for the emitters because of a limitation of available tungsten, and molybdenum is being substituted for SPS use.

7.3 MICROWAVE POWER SYSTEM

For all SPS concepts under consideration, the microwave system is a common element. Current SPS system concepts require microwave antennas to be capable of transmitting several gigawatts of power at 2.45 GHz. The microwave system is currently focused on a 5 GW system.

7.3.1 REQUIREMENTS AND ANALYSIS

The microwave power system consists of those elements that are necessary to receive power from some source, transfer it across a rotary gimbal system, convert it to RF, form a nearly coherent beam, and transmit it from synchronous orbit to Earth with high accuracy. The current baseline microwave system consists of two major subdivisions, each contributing to the overall efficiency of the system. The major components are: (1) the 1 km transmit array with its power distribution system, dc-RF conversion devices, phase control subsystems, and radiating waveguide; and (2) the 8.5 × 11 km rectenna for receiving the RF energy, efficiently converting it to dc, and recombining in an appropriate manner for practical power distribution systems. The manner in which the efficiencies of individual devices and systems are weighted in producing an end-to-end dc-to-dc efficiency of 58 percent is indicated in Table 7-15. These efficiencies ultimately determine the performance of the system.

TABLE 7-15. MICROWAVE POWER SYSTEM EFFICIENCIES

	Efficiency (%)
Antenna Power Distribution	99
DC-RF Conversion	82
Phase Control	97
Waveguides	97.5
Propagation	98
Total	75
(DC to DC System Efficiency - 58%)	

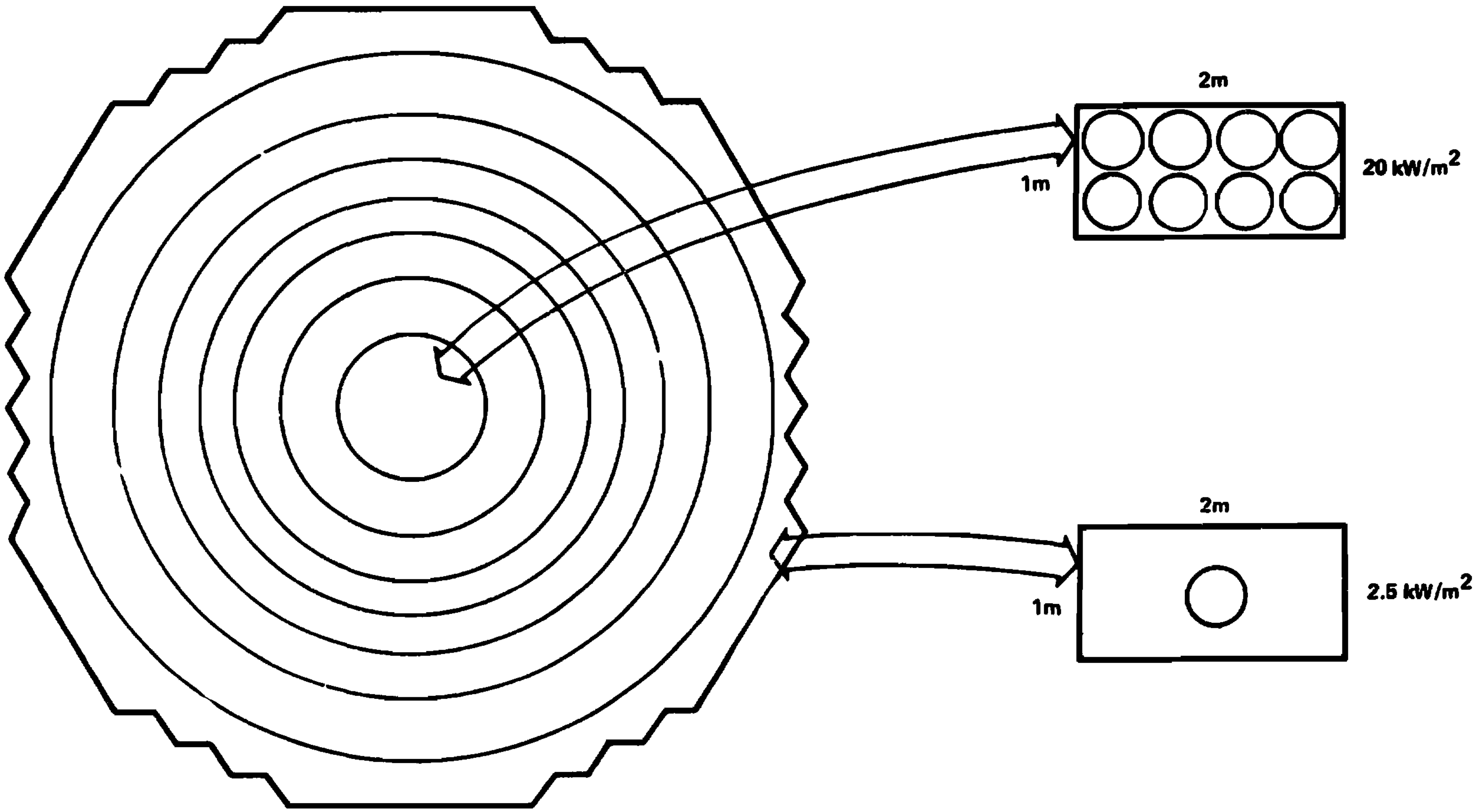


Figure 7-39. 9 dB, 1 km transmit antenna.

Studies have shown that a tapered gaussian power distribution of transmitted energy is most useful and efficient. A value of 9 dB has been selected in this case as the near optimum power taper level. Because of the physical limitations of discrete components and in order to standardize on some fundamental building blocks in constructing the transmit array, this 9 dB power taper is accomplished in eight steps from the center to the edge of the antenna. This approximates an ideal gaussian distribution (Fig. 7-39). The array is built up of 1652 rectangular subarrays which in turn are constructed from the 1 × 2 m basic waveguide dc-RF converter assembly unit. Each subarray contains 230 of these basic elements.

Another important item in the antenna system is the structure. It relates directly to the phase control of the microwave beam and to the pointing accuracy of the overall system. A very rigid structure will be required to support the 1 arc min pointing requirement and the thermal distortion limits of the subarrays. The structure requirements are such that performance should be given priority over cost.

The receiving portion of the microwave system is referred to as a rectenna because of the use of many small dipole elements. Each dipole element receives RF power and, through a highly efficient arrangement of special diodes, filters, and distributing bus system, converts the power to useful dc energy with high efficiency. The baseline system characteristics are contained in Table 7-16. A further analysis of each element of the microwave power system is included in the following discussion.

7.3.2 TRANSMITTING ANTENNA STRUCTURE

The selected structural configuration for the microwave transmitting antenna is built from a single truss column structural element which, in turn, is assembled with other elements to form equilateral pyramids. These then are joined with other like pyramids to form a TRI/HEX structural module. The assembly sequence is portrayed in Figure 7-40, which also includes assembled waveguide/amplitron subarrays. Each of the waveguide/amplitron subarrays are mounted on the main structure by adjustable (possibly screw-jack type) mounts that permit in-plane alignment of all subarrays. Structural load determination, element sizing, and material selection are still in progress with aluminum and graphite epoxy the two most promising materials.

The desired characteristics and requirements of the structural system are:

TABLE 7-16. CHARACTERISTICS OF 9 dB MICROWAVE SYSTEM

Antenna Transmit Aperture	1 km diameter (0.785 km^2)
Subarray Size	20 × 23 m
Basic Element Size	1 × 2 m
Number of Subarrays	1652
DC Input Power to Antenna	8.5 GW
Power Tube	Amplitron
Array Aperture Illuminations	Eight step, truncated gaussian amplitude distribution with a 9 dB edge tape
Phase Control	Adaptive and Command system
Mechanical Pointing	±1 arc min
Electronic Beam Steering	±3 arc s
Rectenna Dimensions at 35° Latitude	8.5 × 11 km
Peak Power Density	23 mW/cm^2
Density at Edge of Rectenna	2 mW/cm^2
Beam Interception	90 percent
Power Output of Rectenna	5 GW
Total Microwave System Efficiency	58 percent
Total Antenna Weight	7.24 × 10^6 kg

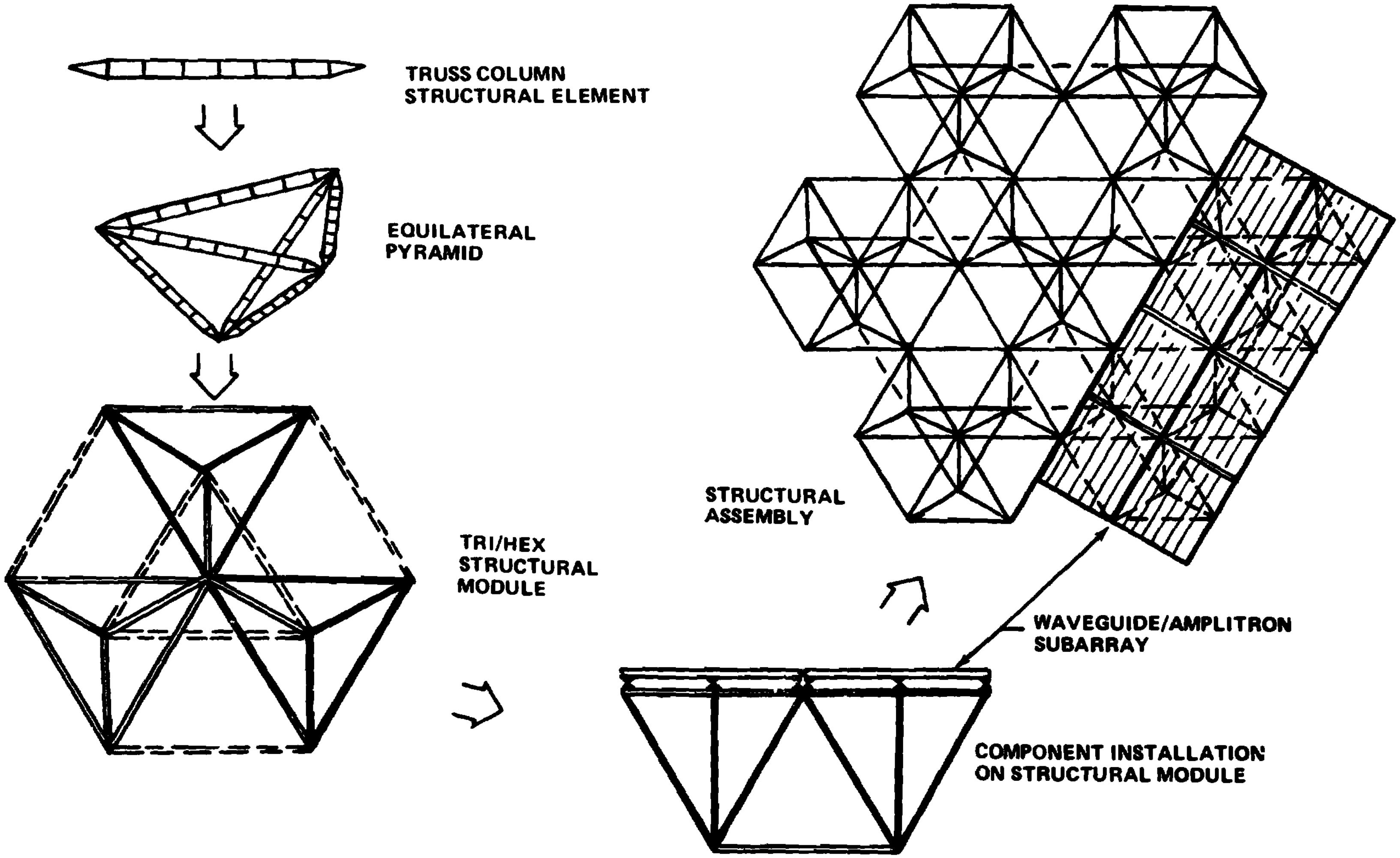


Figure 7-40. Structural module and assembly buildup from common structural element.

- **High Structural Stiffness**
  - **— Beam Pointing Accuracy of 1 min of arc**
  - **— Slew Rates**
  - **— High Mass Area Density of Waveguide/Amplitron**
- **Low Thermal Distortion/Thermal Stress**
  - **— Beam Pointing Accuracy**
  - **— Relatively Severe Thermal Environment**
    - **— Moderately High Structural Temperatures**
    - **— Thermal Gradients Across Structure**
- **Load Paths for Waveguide/Amplitron Subarray Panel Support and Contour Control**
- **Zero Joint Stiction**
  - **— Linear Response to Changing Structural Load Conditions**
- **Structural Dampening**
  - **— Rapid Recovery from Dynamic Disturbances**

Near zero joint stiction can be approximated by excluding bearing-type joints in the structure design. Improved methods to increase structural dampening are design goals, but inherent material dampening may prove to be satisfactory when weighed against other possible alternatives.

7.3.3 ANTENNA THERMAL CONTROL

7.3.3.1 STRUCTURE THERMAL ENVIRONMENT

Figure 7-41 shows the configuration and arrangement on the antenna structure of the amplitron with its passive fin radiators. Since the waveguides are insulated from the amplitron radiators by reflective radiation shields, the bulk of the amplitron waste heat is directed toward the antenna primary structure. Distribution of radiated microwave energy, and corresponding waste heat, is tapered according to a gaussian function from the center to the edge of the antenna.

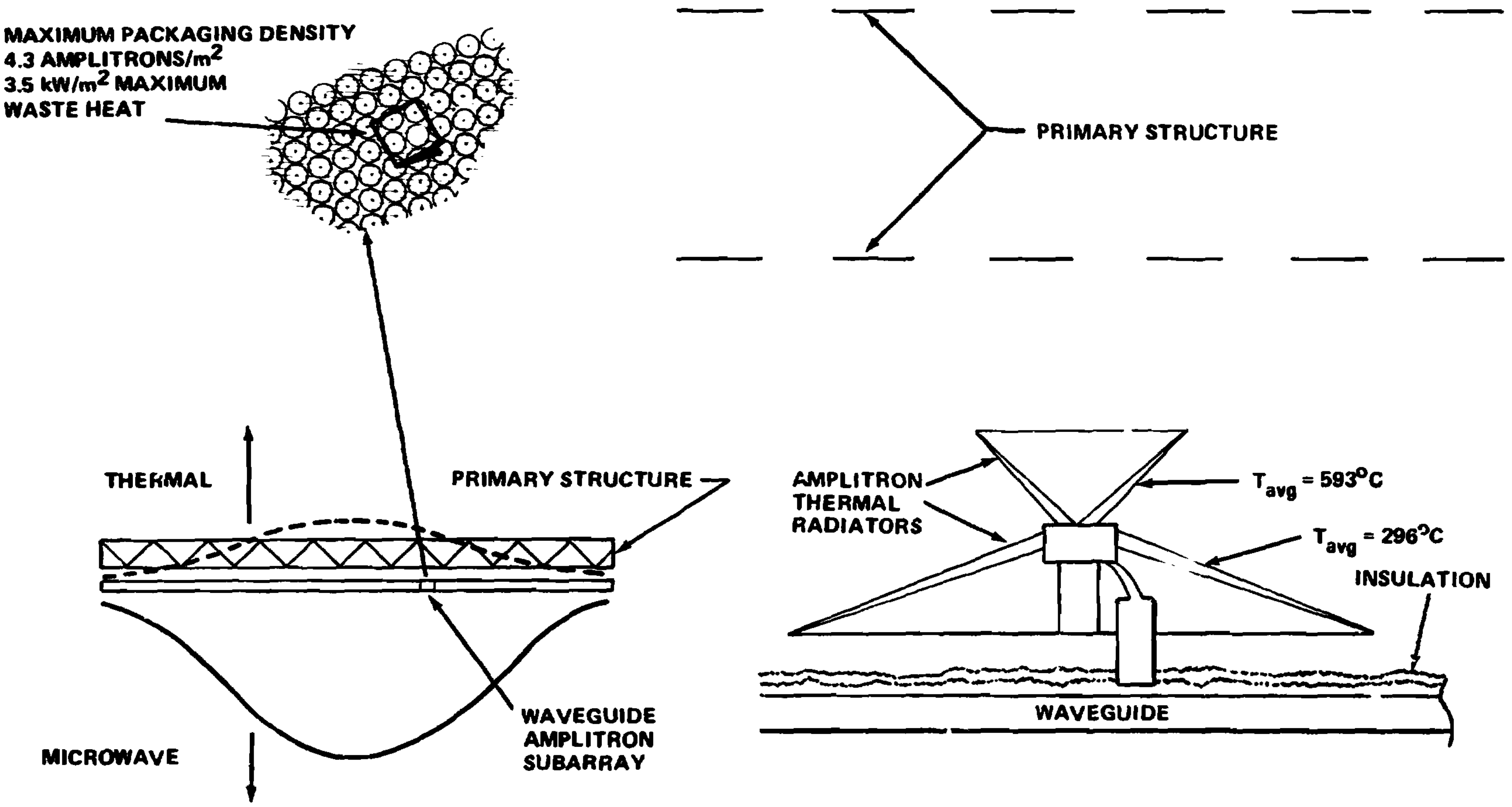


Figure 7-41. Structural system thermal environment.

The maximum waste heat flux from the amplitron subarrays is established by the maximum operating temperature of the amplitron and the size of the fin radiator required to dissipate the waste heat. For an amplitron rated at 5 kW of radiated microwave energy and 85 percent efficiency, 0.882 kW of waste heat must be dissipated, which requires a 48 cm diameter fin radiator. At minimum spacing between radiators, the maximum package density of the amplitrons is 4.[illegible] per m^2. Thermal analysis studies indicate that 87.5 percent of the waste heat is directed toward the primary structure. An assumed additional 200 W of amplifier waste heat produces a total waste heat flux of 3.5 kW/m^2. This value determines the maximum anticipated structural element temperatures.

7.3.3.2 STRUCTURAL TEMPERATURES

Structural temperatures of the microwave antenna are shown as a function of radiated waste heat for flat structural elements both parallel and perpendicular to the antenna plane. For parallel structural elements, one side absorbs heat from the antenna while the other side radiates heat to space, so that its temperature can be lowered by lowering the emissivity value of the absorbing side. For a perpendicular member, however, the sides of the elements are both absorbing and radiating, so that its temperature is independent of emissivity and is given by the curve $\epsilon = 1$ (Fig. 7-42). Although several factors not considered here can increase structural temperatures (solar heating, structural shapes curved for efficiency, and degraded emissivity values), maximum structural temperatures are not expected to preclude the use of aluminum or graphite epoxy at the maximum waste heat flux of 3.5 kW/m^2.

7.3.3.3 STRUCTURAL TEMPERATURE GRADIENTS

The in-plane structural temperature gradients from the center to the edge of the antenna are extreme and can be associated directly with the waste heat profile. Transverse structural temperature gradients are minimal over much of the antenna. Average flux to a point on the structure does not change very much with distance, as can be seen by the view angle depicted in Figure 7-43. Near the edge of the antenna, transverse gradients become more pronounced because of the increased view of space with increased distance from the antenna.

7.3.3.4 STRUCTURAL COOL-DOWN RATE IN THE EARTH'S SHADOW

The cool-down rate for a 0.05 cm aluminum plate parallel to and near the center of the antenna is approximately 9°C min (Fig. 7-44). The combined structural/thermal design impact to the antenna will require much more detailed study.

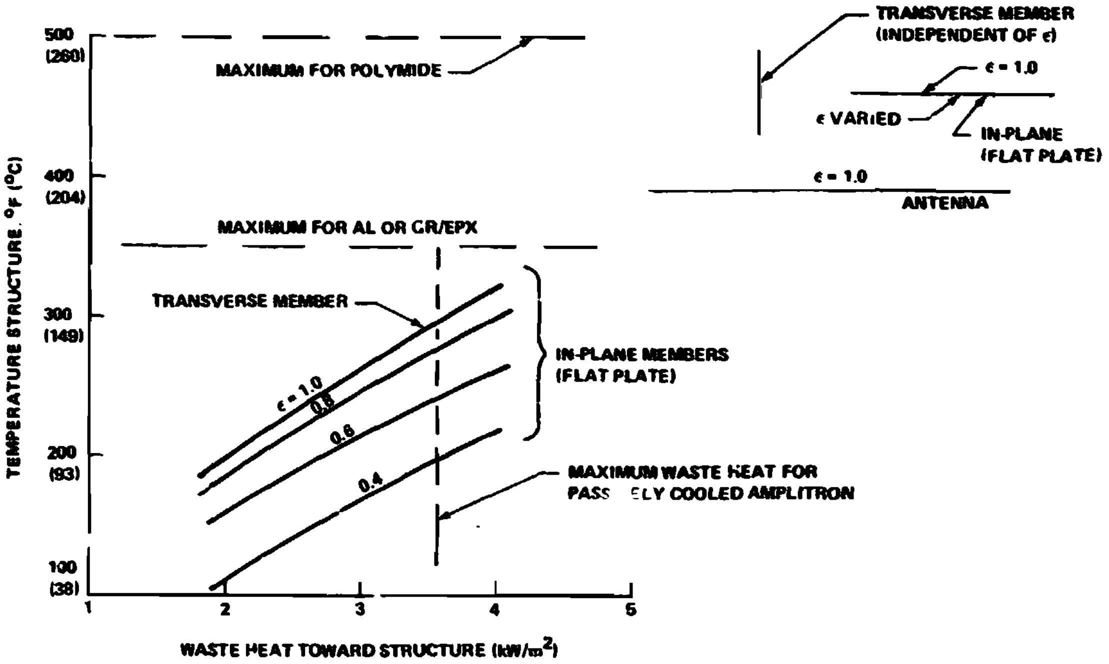


Figure 7-42. Structural temperature versus waste heat.

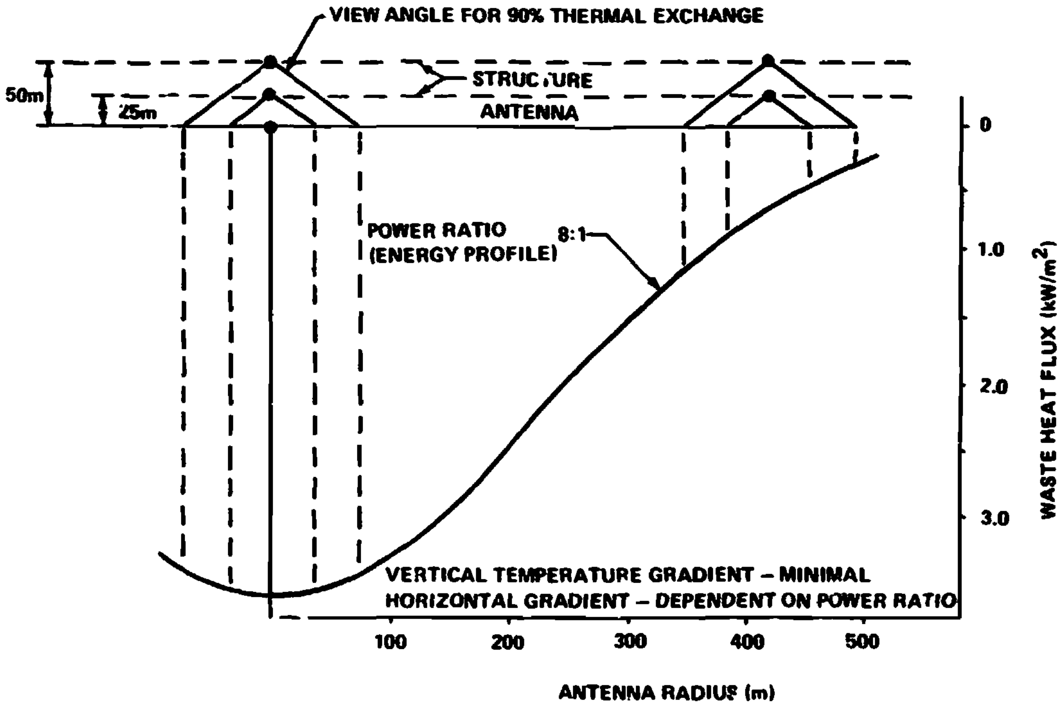


Figure 7-43. Waste heat energy profile.

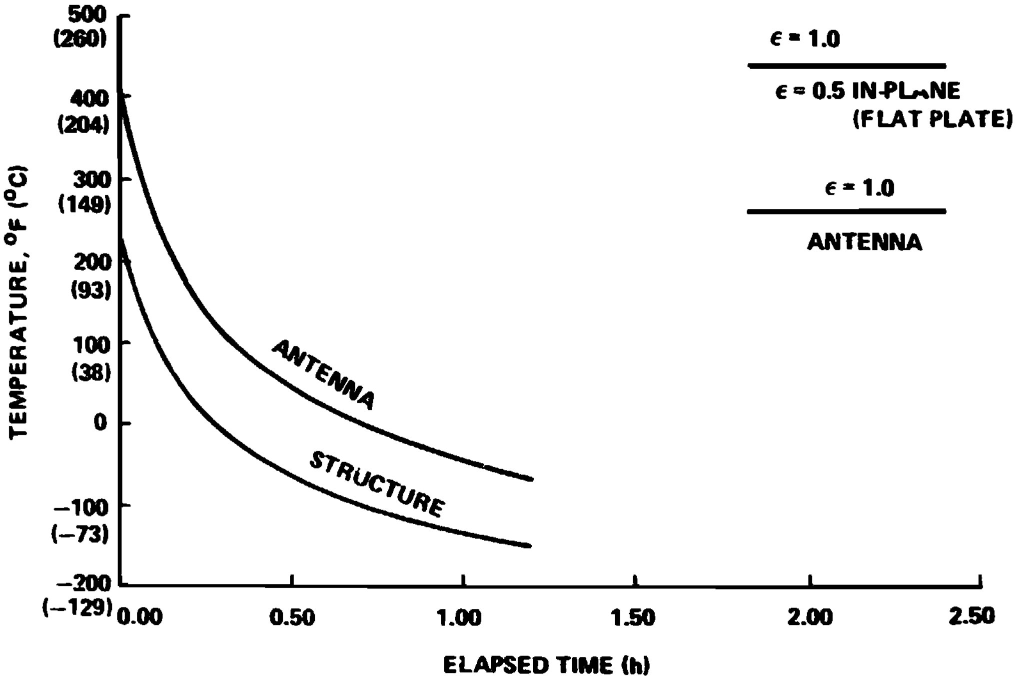


Figure 7-44. Cool-down rate in Earth's shadow.

7.3.4 MICROWAVE GENERATION SUBSYSTEM

7.3.4.1 INTRODUCTION

The amplitron or crossed-field amplifier was selected as the candidate RF generator for this study. It is to be emphasized that a final selection cannot be made at this time because of necessary technological advancements in all RF generators under consideration. There are no known RF generators available today that have the required characteristics to make the SPS economically or technically feasible; however, the required advancements are within the forecast technology with little risk or need for technological breakthroughs.

Primary factors to be considered in the selection of the generators are reliability, efficiency, lifetime, size, mass, and radio frequency characteristics. The selection was narrowed to vacuum tube devices, since solid state devices, at present, produce lower power and have low efficiency compared to vacuum devices. The most promising vacuum devices are the klystron and the amplitron, which have the best potential of being developed to the extent necessary for the SPS.

7.3.4.2 THE KLYSTRON

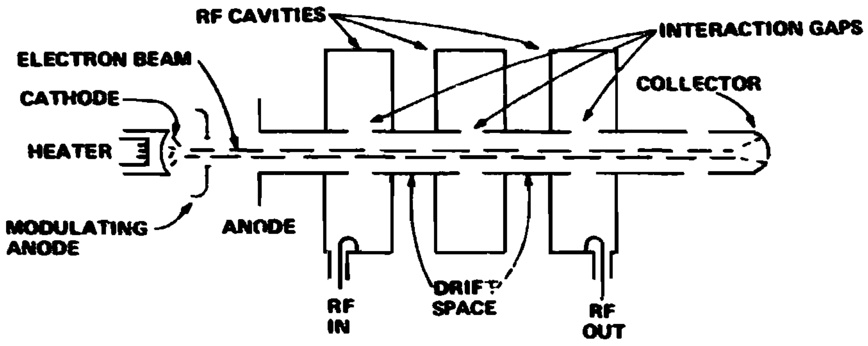
A diagrammatic representation of the klystron amplifier is shown in Figure 7-45(a). An electron beam is emitted and formed by a heated cathode and electron gun. The electron beam is then accelerated by a positive potential on the RF cavities. The beam is also velocity modulated by the RF input. The resultant action between the velocity modulation and the tube geometry causes the electrons to "bunch" or become density modulated. Power is then extracted from the beam by conventional means such as employing a loop or iris coupling. The electrons, after power extraction, are collected by the collector anode. The primary advantage of the klystron over the amplitron or other tubes in producing high power lies in its geometry. The beam formation, RF interaction, and beam collection are separate and independent regions; thus, each region may be designed to perform its own function independent of the others. Since the collector electrode is the primary heat dissipator, its shape and size can be designed for the power under consideration without regard to the other functional regions of the tube. A 50 kW tube has been proposed as an RF generator for the SPS. The gain of the klystron is dependent on the number of cavities and, therefore, its length. Gains in excess of 80 dB are possible.

7.3.4.3 THE AMPLITRON

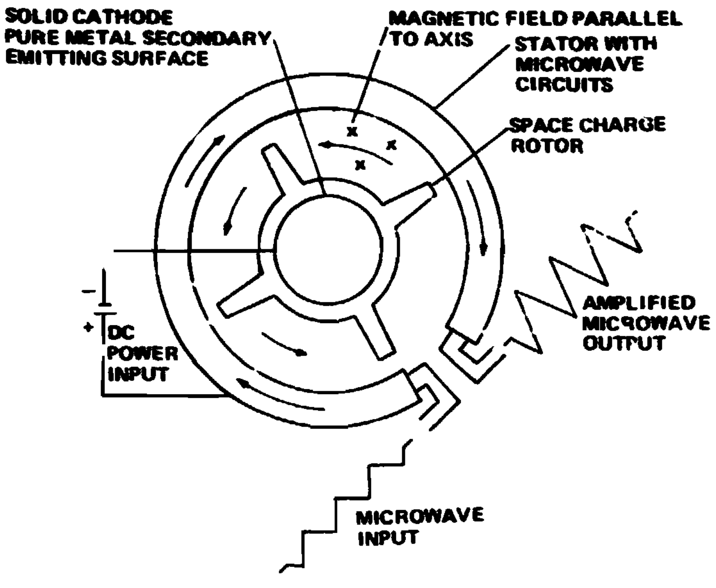
The amplitron is a crossed-field device in which the electron beam is perpendicular to the magnetic and electric fields. The basic structure of the amplitron is shown in Figure 7-45(b). The electrons are emitted by a cold cathode upon RF signal injection. Amplification is similar to that of the traveling wave tube and is a result of the interaction between a traveling electromagnetic wave and a rotating space-charge wave. The space-charge waves are formed by the interaction between the electron beam and the crossed electric and magnetic fields. The amplitron is a relatively low gain device with typical gains of 7 to 10 dB. The low gain is not a handicap, since the RF input power is not lost but appears as a part of the output. Also, the amplitron acts as a low insertion transmission line when not energized. These characteristics are employed by cascading the amplitrons in the transmitting array. Should a tube fail, the RF power will feed through the nonoperating tube without a serious effect on the SPS.

7.3.4.4 AMPLITRON/KLYSTRON COMPARISON

The amplitron was selected for study purposes primarily because of its greater efficiency and potentially greater lifetime. Since both tubes will be operating as dissipation-limited devices, the useful output power is proportional



a. Diagrammatic representation of the principal parts of a three-cavity klystron.



b. Principle of operation of the amplitron; rotating spokes of space charge induce currents into the microwave circuit and provide efficient amplification of the microwave input signal.

Figure 7-45. Klystron and amplitron vacuum devices.

to $\eta/1-\eta$, where η is the device efficiency. The amplitron is predicted to be approximately 4 percent more efficient than the klystron. When comparing an 81 percent efficient klystron versus an 85 percent efficient amplitron, the above proportional factor causes the klystron to have approximately 25 percent more power to radiate or reject than the amplitron, assuming that all factors are equal except the 4 percent disparity in efficiency. Figure 7-46(a) is a plot of the proportional factor $\eta/1-\eta$ for efficiencies ranging from 80 to 90 percent. As can be seen, a device which is 85 percent efficient can produce an output power 5.67 times its dissipation, whereas a device which is 81 percent efficient can only produce 4.26 times its dissipation. Figure 7-46(b) is a plot of the klystron output relative to an 85 percent efficient amplitron versus the klystron differential efficiency. In reality the difference would be smaller, since the klystron's geometry allows active cooling and its magnetics may run approximately 50°C hotter than the amplitron, which makes its heat radiator more efficient. A klystron system would generate more heat for a given power output and place more demand on the heat rejection subsystem.

The amplitron has a potentially long life, since a pure metal secondary emitting cathode will be used for which there is no known life limitation. In addition, since the cathode is not heated, the overall construction of the system is simplified. The klystron, on the other hand, has a heated cathode and an active (heat pipe) cooling subsystem, which add complications and limited life components. The klystron amplifier has better radio frequency interference (RFI) characteristics than the amplitron. RFI characteristics are important since interference may result to communication channels and other related users of the radio frequency spectrum because of the very high power levels of the SPS microwave beam and its spurious noise and harmonics. The klystron is a linear device and has spurious noise of 90 dB relative to the fundamental, whereas the amplitron is a saturated amplifier with spurious noise of 40 dB relative to the fundamental, making the klystron more suitable at this time relative to RFI.

7.3.5 SUBARRAY SUBSYSTEM

The 1 km transmitting antenna consists of 1652 20 × 23 m subarrays (Fig. 7-47), each individually adjustable by mechanical screw-jacks. The subarray consists of 230 basic elements (1 × 2 m). These elements are the lowest replaceable units (LRU) in the transmit array. The LRU was sized based on two primary considerations. First, the 2 m^2 area allows one to eight amplitrons to be mounted on any LRU, which provides an eight-step power taper. Second, the LRU should be small enough to be handled by a man inside a repair module. The LRU is small enough to be removed or replaced from the rear of the

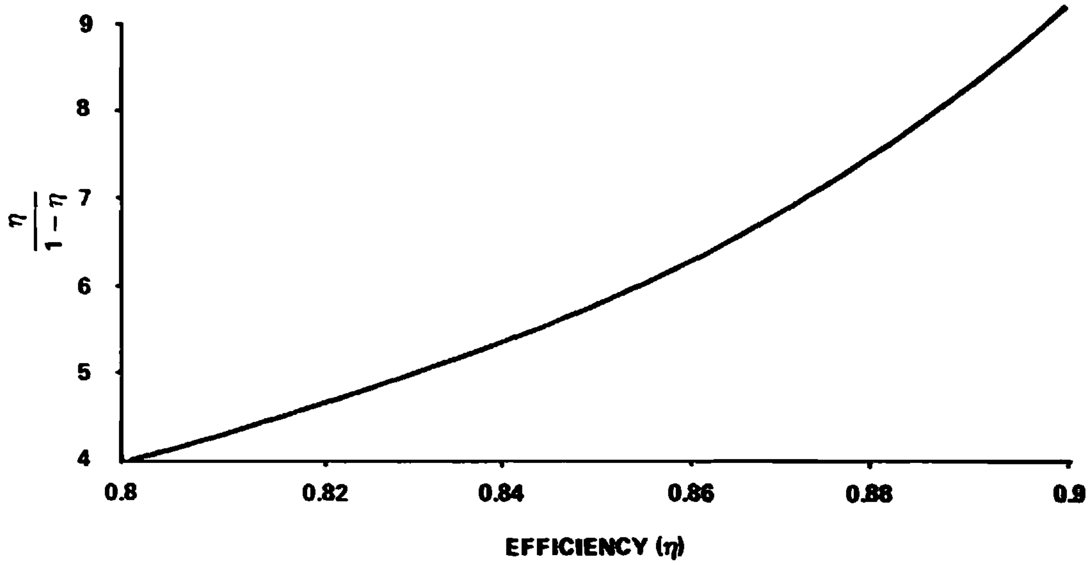


Figure 7-46(a). Plot of $\eta/1-\eta$ for efficiencies ranging from 80 to 90 percent.

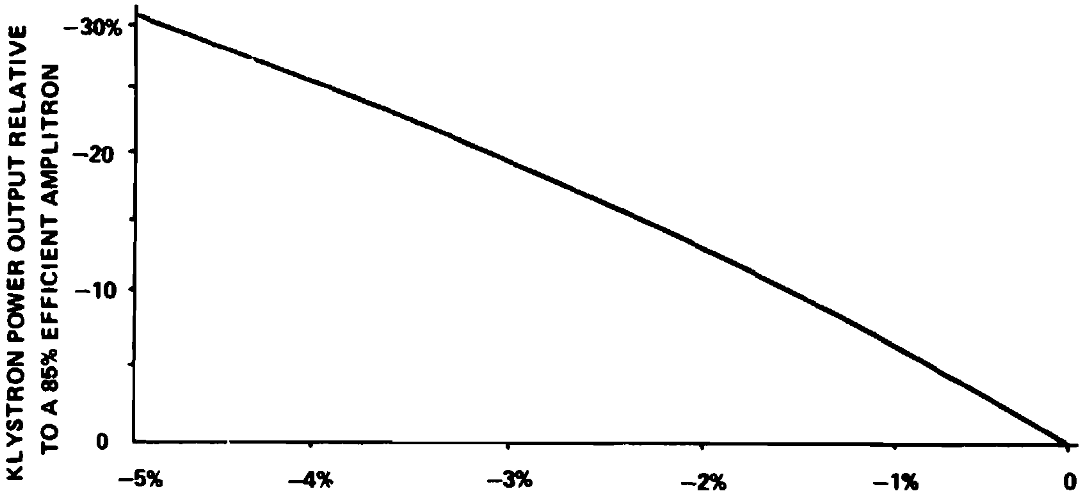


Figure 7-46(b). Klystron differential efficiency relative to an 85 percent efficient amplitron.

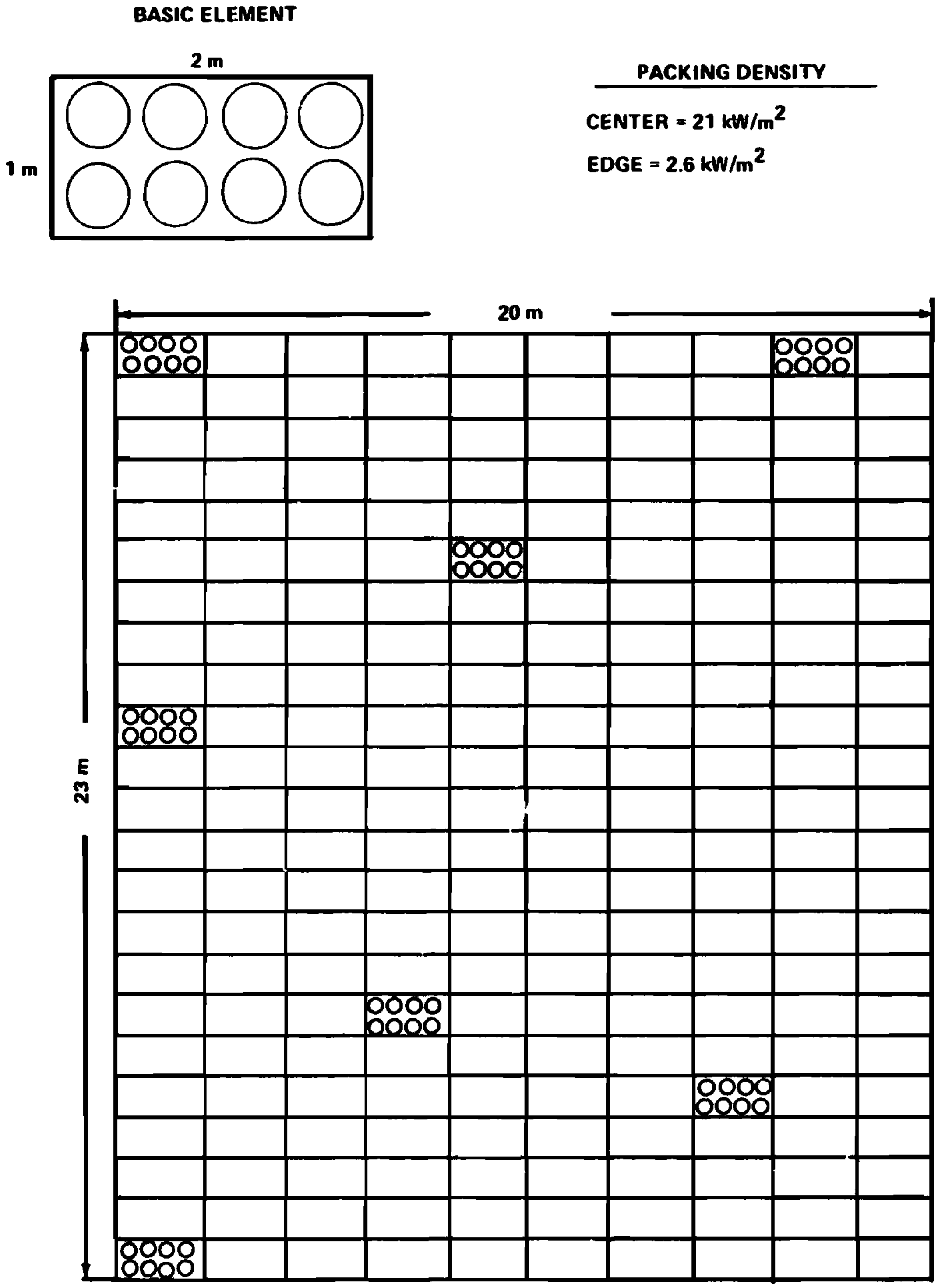


Figure 7-47. Microwave antenna subarray.

antenna, so the machine doing the removal does not have to cross the face of the antenna while it is radiating. This design does not preclude removal of an entire subarray, but this would have to be accomplished on the front or radiating portion of the antenna with the disconnection occurring from the rear.

The size of the subarray resulted from three primary considerations. First, the subarray should be large enough to minimize the loss resulting from gaps between the subarrays. Second, a dimension was chosen that was workable with the LRU. Third, since it is necessary to maintain the flatness of the subarray, a dimension was chosen that coincides with the 60° structure load path and matches the hard points of the structure. Figure 7-47 is the basic subarray design with an example of an eight-amplitron LRU. This subarray would actually be completely covered with 1840 amplitrons. The basic subarray size and LRU size would be identical throughout the 1 km antenna. Since there are eight steps or eight power quantizations from the center of the antenna to the edge, there are eight different feed systems for the LRU's. The subarrays have slotted waveguide radiating apertures with dimensions corresponding to the transmission frequency of 2.45 GHz. The current design uses amplitrons in a cascade arrangement, where one amplitron provides the drive for the next amplitron in the chain. A maximum of eight and a minimum of one are cascaded. Each LRU has a driver amplifier and a phase controller that allows phase control at the LRU level. Also, each amplitron has an adjustment for phase control. The waveguide metal thickness was chosen as 0.5 mm to minimize the weight of the antenna. Manufacturing tolerances for the slot dimensions will be on the order of 0.025 to 0.050 mm.

Although the subarray has been sized at 20 × 23 m, this design is one of many that can satisfy the requirements of the microwave system.

7.3.6 PHASE CONTROL

One of the most critical technology items necessary for the microwave power system is the phase control subsystem. Although mechanical means of pointing the 1 km planar phased array antenna may be achievable to within approximately +1 arc min, overall efficiency and safety for an acceptable system will demand greater beam pointing accuracy. Beam pointing and focusing accuracies in the single digit arc second range require electronic means of phase control. Phase relationships between elements of the planar array determine the transmitted beam pattern, directivity, and degree of side lobe suppression. These factors have a significant effect on overall system efficiency. Since there are many variable parameters that potentially change the phase relations and, thus, the transmitted beam pattern, a positive adaptive form of phase control is needed to compensate for ionospheric and atmospheric changes, thermal deformation of structures, and phase variation in transmission lines and electronic system components. Figures 7-48 and 7-49 illustrate the two basic methods for array phase control. One or both of these methods, with refinements, will accomplish the very critical control of beam pointing and focus.

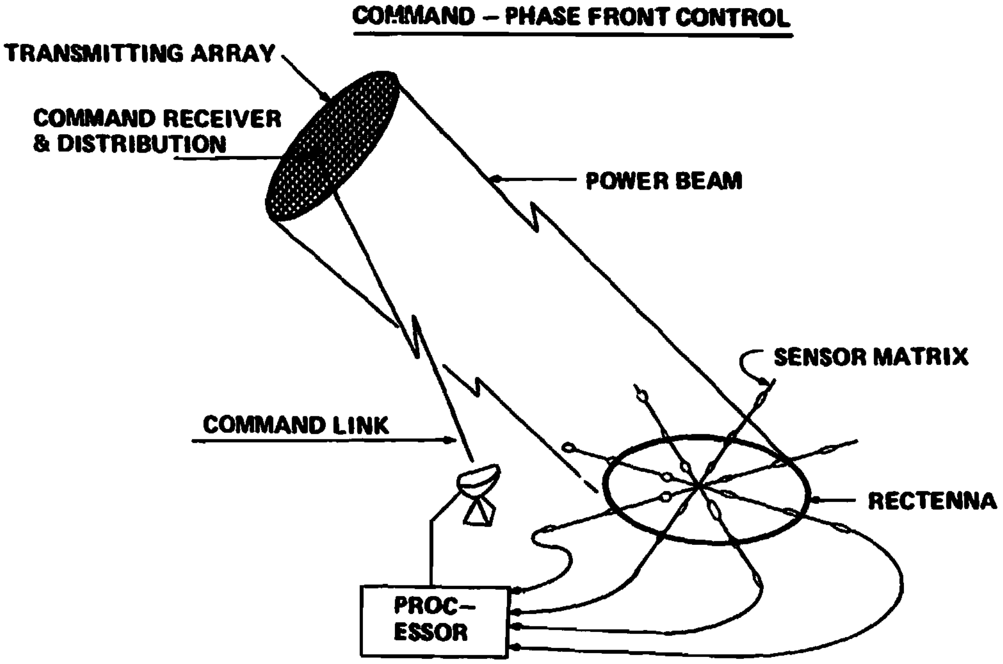


Figure 7-48. Microwave power transmission system phase front control concepts.

The simplest system of the two, the Command system (Fig. 7-48), utilizes a network of sensors on the ground receiving antenna to compare the received pattern to a computerized model. Changes in received pattern are interpreted by the ground computer as representing errors in the transmitted phase and are relayed via telemetric links to the solar power control station as a remedial command. This system obviously requires a sophisticated interpretive program relating ground antenna variations to the nature and location of changes required on the transmit antenna. The exact number and type of sensors required for such a system remain undefined, but they will probably be

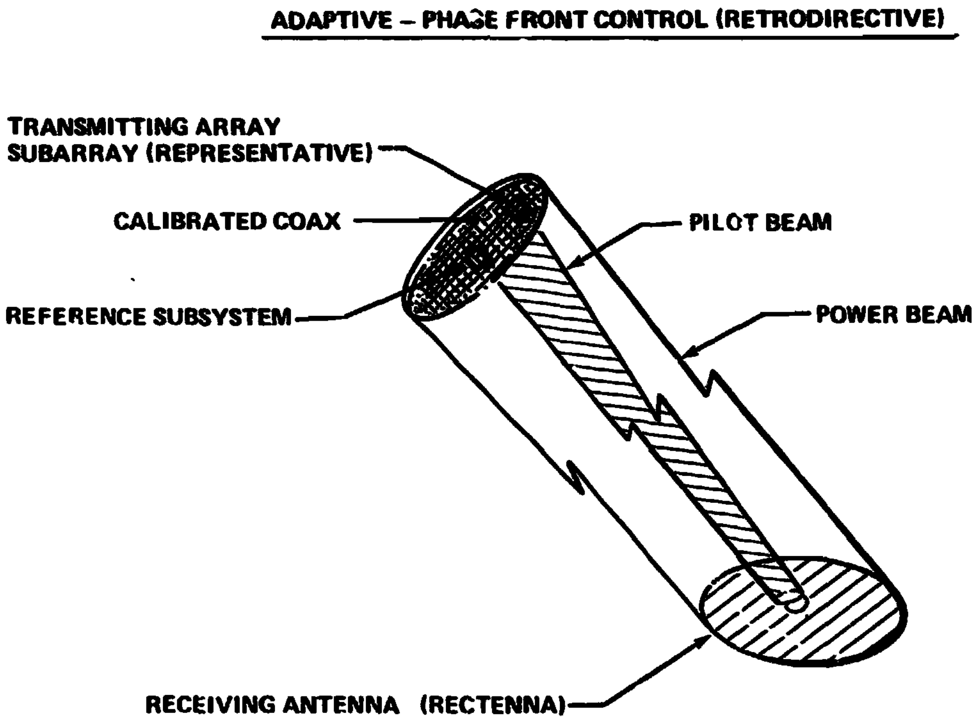


Figure 7-49. Microwave power transmission system phase front control concepts.

sensitive power density indicators with rapid response characteristics. If one such sensor were required for each 100 m^2 of rectenna area, the total number required for the proposed system would approach one million.

The Command system offers a valid system for safety monitoring and control, but does not appear to possess a capability of resolution to the degree necessary for high order phase corrections.

The Adaptive phase front control system (Fig. 7-49) requires a calibrated phase reference distribution system at the transmitter array, linking subarray elements to the central reference signal. It also requires at least 1652 receiver units (one for each subarray) to accept the ground developed pilot beam, compare it to the reference signal, and make appropriate phase adjustments. The heart of this system is the reference signal and its distribution, and this is the phase that requires the most development. A number of ways to carry out phase reference distribution have been discussed including calibrated coaxial line, space-fed laser techniques, and combinations of these. It appears, currently, that some form of calibrated hard-line transfer will be required in conjunction with a form of the subarray-to-subarray transfer scheme proposed by JPL. Space-fed and other concepts require continuing study and analysis to determine the best design candidate.

Both Adaptive and Command systems require complex computerized systems for control algorithms. Control capability will improve as historical data are compiled. The Command system appears to be the most direct system for safety control of the microwave beam, because it directly senses microwave intensity and intensity changes on the rectenna itself. Multiple sensor networks on both the transmitter array and rectenna will monitor distributed thermal and strain characteristics. These sensor outputs will also be fed into the computerized phase control subsystem.

7.3.7 COMMUNICATIONS, CONTROL, AND DATA MANAGEMENT

Although detailed design data are not available on this system, it is meaningful to discuss some observations that have evolved. Based on the current photovoltaic baseline, there will probably be at least three separate data management systems, one for each solar wing and one for the antenna. These three systems will be interfaced to a control complex that will be located in the manned module. The design of each of these systems will be affected by the same requirements affecting the SPS, i.e., thermal environment (especially on the antenna), long data transmission distances across the SPS, space charging, and high current voltage considerations. However, with the possible exception of the unique effects of space charging, present technology should be adequate for all foreseeable needs. Another important design criterion will be the reduction of the potentially very large data rates (up to 150 megabits). Considering the total number of active components that will contribute information to the data management system, the importance of schemes to reduce this rate becomes evident. Maximum use of microprocessors at the data source and a large number of data distributors will be required in addition to the three separate data centers and the central control system. Figure 7-50 represents a preliminary block diagram of the microwave power transmission system. Only one computer processor unit is shown, although there would probably be several linked to a master system. Each of the boxes shown is considered to contribute significantly to the data system. In comparison to any known data management system to date, this system would be rather costly, complex, and massive. However, when compared to the overall SPS, the communications and data management subsystem will probably represent approximately 1 percent of the total cost and less than 1 percent of the total weight.

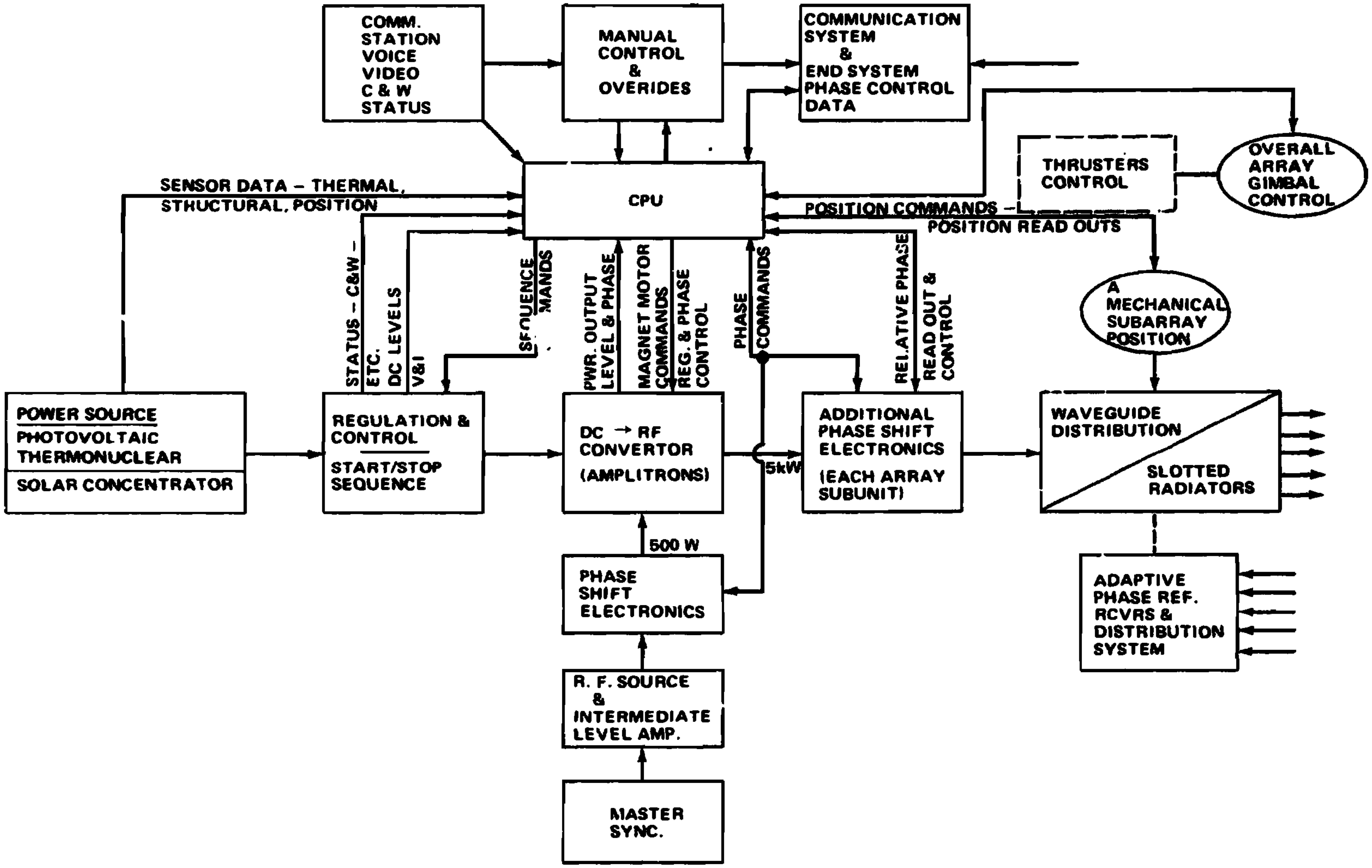


Figure 7-50. SPS microwave power transmission system.

7.3.8 MICROWAVE POINTING AND CONTROL

7.3.8.1 GROUND RULES/GUIDELINES

The ground rules and guidelines given for the photovoltaic conversion system apply to the microwave power system pointing and control subsystem. One possible exception could be the use of control moment gyros (CMG) as stabilizing and control actuators for the antenna structure. For this exception to be feasible, however, an improvement in the present momentum to weight ratio CMG technology must be realized.

7.3.8.2 POINTING AND CONTROL CONCEPT

Pointing the transmitting antenna array to the ground rectenna with the desired accuracy requires the following approach:

- Orientation of the SPS main body to the reference attitude will be to an accuracy of ±1° in three axes.

- Acquisition of the ground rectenna by the microwave beam, and maintenance of this pointing direction, to an accuracy of ±1 arc min in two axes with an error of less than 8° about the line of sight will be accomplished by a gimbaled antenna pointing control system.

- Fine pointing of the microwave beam to an accuracy of ±3 arc s will be accomplished by the antenna phase control subsystem.

Initial acquisition of the ground rectenna will be accomplished by processing orbital parameters, the known attitude of the SPS with respect to Sun and Earth coordinates, and the rectenna location with respect to the SPS. After coarse pointing of the antenna, a scan sequence may be required for final acquisition. If a scan sequence is required, attitude errors will be obtained by processing a portion of the pointing information obtained by the phase control subsystem.

For a given orbit and SPS attitude, the predominant antenna motions required to maintain pointing are well defined. For example, a rotation of 360° per day about the X_B axis is required as the Earth and SPS rotate synchronously and the Z_B axis is oriented in a sunwardly direction. The antenna tilt motion and roll about the line of sight are also well defined once the SPS reference attitude has been acquired. Therefore, these predominant motions can be preprogrammed to take place in a well defined time sequence.

The effects of external and internal disturbances will be sensed by the phase control subsystem. These sensed pointing errors will then be used to modify the preprogrammed antenna motions to maintain pointing acquisition to the desired accuracy.

7.3.8.3 OPERATIONAL ATTITUDE IMPACTS

The two operational attitude alternates were discussed in subsection 7.1.6.3 for the photovoltaic conversion system. Table 7-10 summarizes the predominant antenna motions that result from each of the two attitudes. The basic 360° per day rotation is required for either attitude. However, the Z-solar orientation requires large angular vertical and roll motion. Since the large motions for the Z-solar attitude must be controlled to the specified pointing accuracy requirements, the pointing task for this attitude is critical. The X-POP orientation greatly simplifies the pointing task, since the requirements are for a 360° per day rotation with the antenna tilted at the fixed angle required to point to the ground rectenna. Therefore, even though additional solar panel mass is required for the X-POP mode, the simplified antenna motions appear to make the X-POP mode the desirable attitude orientation.

7.3.8.4 GIMBAL CONFIGURATIONS

The antenna gimbal system (rotary joint) must provide the following functions:

- Rotate the antenna 360° per day with respect to body coordinates.
- Tilt the antenna as a function of the geographical location of the ground rectenna.
- Maintain the desired orientation of the antenna with respect to the rectenna about the microwave beam line of sight.
- Correct for any perturbations of the microwave beam pointing direction.
- Minimum loss power transfer.

The mechanical assembly requires the equivalent of two or three gimbals depending on the geometry of the mounting structure and the SPS attitude orientation. If the active area of the solar conversion system is maintained normal to the solar vector, structural offsets of 23.5° can be supplied for the antenna mount to permit antenna rotation of 360° per day about a plane perpendicular to the orbit. These structural offsets eliminate the need for large tilt and roll motions of the antenna.

There are many options possible for the gimbal configuration. An extensive trade study and design effort is required to optimize the system from the standpoints of transportation, assembly, complexity, and pointing performance.

7.3.9 POWER DISTRIBUTION AND CONTROL SUBSYSTEM

Electrical power is supplied by 24 main buses. Circuit breakers are located at the power sources for protection of the power subsystems, and additional circuit breakers are located on the antenna section to provide protection and bus control for the amplitrons. Power is distributed on the antenna system by thin-walled aluminum conductors. Power conditioners are located at the power source, and load voltage is sensed remotely.

7.3.10 ROTARY JOINT SUBSYSTEMS

Present concepts for SPS require the power transmitting antenna to rotate about one or more axes with respect to the main structure. Rotation is necessary to allow the solar collectors to be Sun oriented while the transmitting antenna remains pointed to a fixed ground rectenna. (See subsection 7.1.6.3 for a discussion of the required antenna operational attitudes.) Rotary joints must provide structural stability and mechanisms to provide the proper antenna orientation as well as a means to transfer power.

Preliminary analysis has indicated that the use of slip rings and brushes for transferring power across a 360° continuously rotating joint will be more efficient and have less mass than other concepts. Flexible cables are prime candidates for use across joints that have limited freedom.

The highest rate of rotation necessary is 360° per day. With a design lifetime of 30 years, brush wear should not be a problem if slip ring diameters are held in the order of tens of meters. Several existing combinations of brush and slip ring materials have shown low enough wear rates to be considered for this application. Unless active cooling is incorporated, indications are that the brush areas that limit brush current densities to reasonable levels will not be the determining factor in sizing the rotary joint, rather structural and thermal considerations will determine the joint dimensions.

The number of individual power circuits required across the rotating joint also influences the design. For simplicity of design, it would be beneficial to minimize the number of circuits across the joint; however, this complicates the power distribution network and switch gear necessary to handle extremely large blocks of power.

7.3.11 REQUIRED TECHNOLOGY ADVANCEMENTS

The SPS could be produced with today's technology but would be grossly inefficient and prohibitively expensive. Many advancements in technology are required to produce an SPS with the projected efficiencies and acceptable economics. The following paragraphs will summarize the more important advancements needed on the microwave power system.

The transmitting antenna structure will require advancement in space manufacturing of large beams, thermal coatings that maintain their stability over 30 years, assembly and service techniques for the overall structure, and improved ground analytical techniques and methods for simulating static and dynamic properties of large structures in space.

Although the actual power tube has not been selected, both prime candidates require some technology advancements. The amplitron projected in this study and in the Raytheon report does not exist; therefore, the development of a low noise and highly efficient tube is a necessity. The open cathode construction and high power will require safeguards against arcing. Materials will have to be developed that withstand the heat generated and to maintain the retentivity of the magnet. The klystron, although more widely used, would require technology advancement to operate at the projected level of efficiency. Advancements are required in active cooling, safeguarding against arcing because of open cathodes and high current, and cathode material for hot cathodes that will last 30 years.

The subarray subsystem will require technology advancement for space manufacturing of waveguides that meet the tolerances and low loss requirements of the system. New materials will have to be developed that are insensitive to thermal distortion and yet easily conduct microwaves.

Current phase shifting devices operate in discrete steps of several degrees. New devices will be needed that are continuously variable with resolutions in the order of fractions of degrees. Completely new methods of phase control need to be developed.

Current technology is adequate for the pointing control subsystem. However, the large size of the systems required will necessitate new assembly techniques. If control moment gyros are necessary, they would require technology advancement to increase the momentum-to-mass ratio.

The rotary joint subsystem requires new assembly techniques and development of materials that degrade very little over 30 years.

8.0 GROUND RECEIVING AND DISTRIBUTION SITE

8.1 REQUIREMENTS AND ANALYSES

Only preliminary analyses have been made to define the requirements and concepts for an SPS ground system required to interface the SPS with the existing utility systems. Subsequent integrated system studies that include the involvement of the utility industry are planned to establish specific realistic design and operational criteria.

The primary functional requirements of an SPS ground system are summarized as follows:

- Receive microwave energy from the SPS and convert it to electrical energy.
- Condition power with the proper characteristics.
- Measure and control the power routing.
- Deliver electrical power to existing utility system(s).
- Provide a reliable means of monitoring and controlling the microwave beam and SPS subsystems, as well as ground subsystems.
- Provide the necessary protection and fault isolation capabilities for the system.
- Provide communications and data management capabilities to enable monitoring, assisting, or directing operational activities and to ensure crew safety.

The major objectives of the SPS ground system design will be:

- To provide low maintenance subsystems and equipment capable of handling up to 10 GW of power.
- To assure that the overall SPS system will be safe and provide dependable service for at least 30 years.
- To minimize the size of operational crews and costs.
- To economically optimize system performance.

A concept of an SPS ground system that accommodates the foregoing requirements is shown in Figure 8-1. Requirements for energy storage have not been confirmed (dashed lines) and were not included in the SPS studies. The SPS concepts studied were based on an output power requirement between 5 and 10 GW from the rectenna/power grid and the efficiency chain given in Figure 7-1. As indicated by the last dashed block of the chain, an additional loss of 8 percent would be incurred if subsequent ground systems were included.

Conclusions can be summarized as follows:

1. A separate, essentially self-sufficient SPS ground system will be required for each SPS deployed.

2. A major part of this system will be centralized at the SPS receiving site.

The conclusions were based on the observations that:

- Studies indicate a separate rectenna will be required for each SPS microwave/antenna system.

- Most of the monitoring, control, and operational requirements are inherently related to a given SPS and its corresponding rectenna and ground subsystems. These must be closely coordinated and dedicated to the given system.

- The location, size, and capacity of the SPS receiving site will be primarily determined by microwave/rectenna requirements. Thus, the control center and other ground subsystems should be located at the SPS receiving site to enhance simplicity, performance, and operational convenience of the system and reliability of the controls. The system, however, must provide and be responsive to external interfaces as depicted in Figure 8-1.

Considering the rectenna described later and other requirements, it appears that a receiving site would occupy an area of approximately 120 km^2.

8.2 RECEIVING ANTENNA SYSTEM

8.2.1 SUPPORT STRUCTURE

No explicit analysis has been made for the rectenna supporting structure; however, some preliminary observations are apparent as follows:

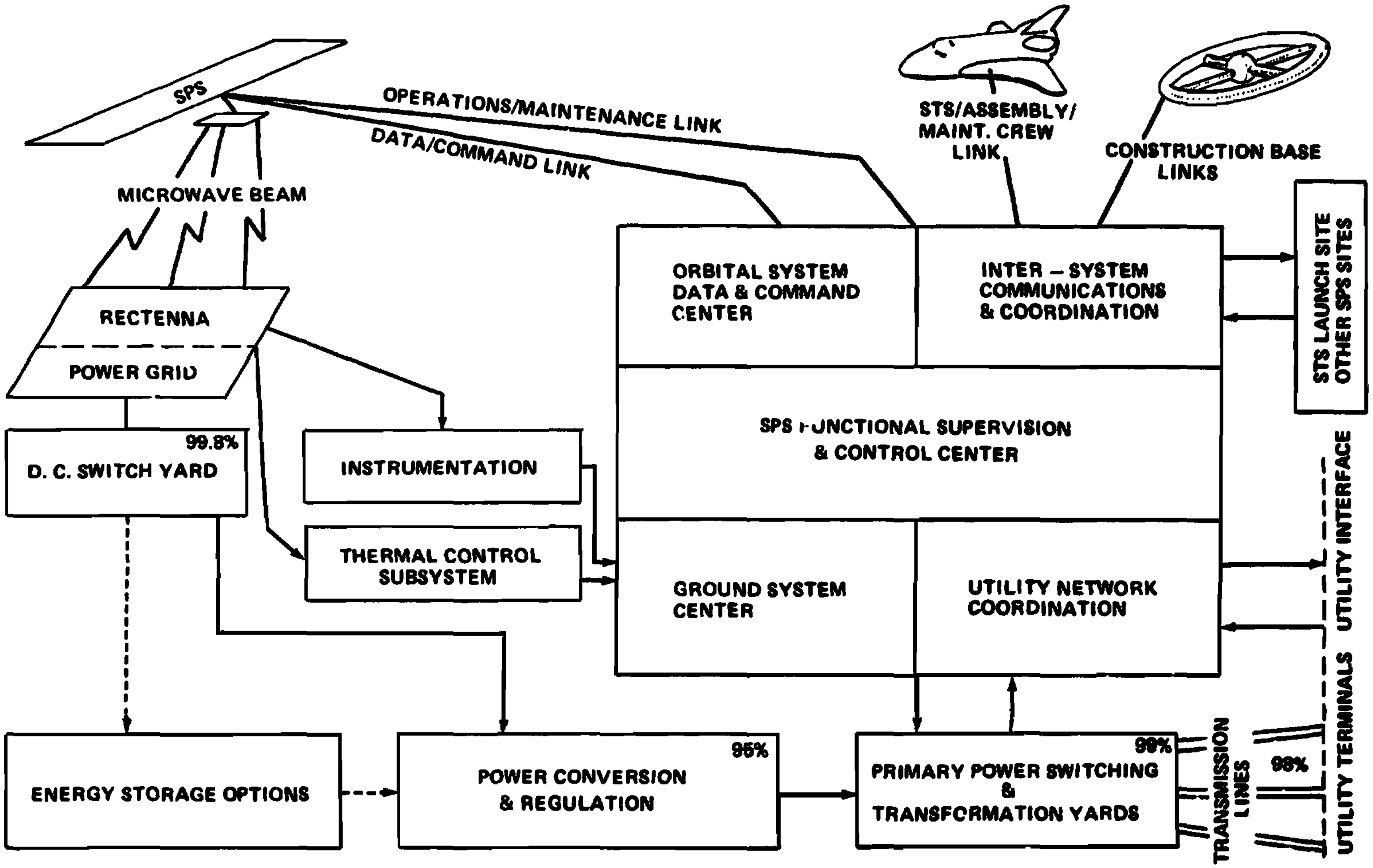


Figure 8-1. SPS ground system concept.

1. The large size of the rectenna, 8.5 × 11 km, will require a significant quantity of supporting structures to develop appropriate look angles and shapes.

2. Alignment and accuracy requirements are not expected to be critical.

3. Conventional construction materials are expected to be adequate.

4. Construction allowing accessibility for conventional long term maintenance and repair will be required.

5. No unusual thermal environments are expected to exist in and around the antenna.

8.2.2 RF-DC CONVERSION

This subsystem, commonly called the rectenna, collects the RF energy and converts it to dc. The power grid accumulates the power at a relatively low dc voltage and distributes it to the dc switch yard which centralizes and controls the flow of power to other subsystems, as explained in subsection 8.3. This discussion will be limited to the collection and conversion of the RF energy and the characteristics of the safety zone. There are numerous variables that determine the overall size of the rectenna including RF frequency, size of the transmit array, power taper on the transmit array, geographic location, fraction of the total beam interception, pointing accuracy of the antenna, and random phase error. These variables can be analyzed and evaluated for a given SPS design. Figure 8-2 represents the current status of these variables. The elliptical rectenna is 8.5 × 11 km and consists of approximately 13.6 billion conversion elements. A single element consists of a half-wave dipole, an integral low pass filter, a diode rectifier, and an RF bypass capacitor. The dipoles are dc insulated from the ground plane and appear as RF absorbers in parallel to the incoming RF wave. Their dc outputs are in a parallel and series combination to result in the desired output voltage and current levels.

The RF ground distribution pattern (Fig. 8-3) produced by the current baseline antenna system allows determination of the size of the ground receiving site once the acceptable power density level is established. A lack of standards prevents the establishment of exact parameters on minimum RF density; however, the effect can be analyzed by arbitrarily selecting various minimum RF densities. For an RF density of 0.01 mW/cm^2, the safety zone

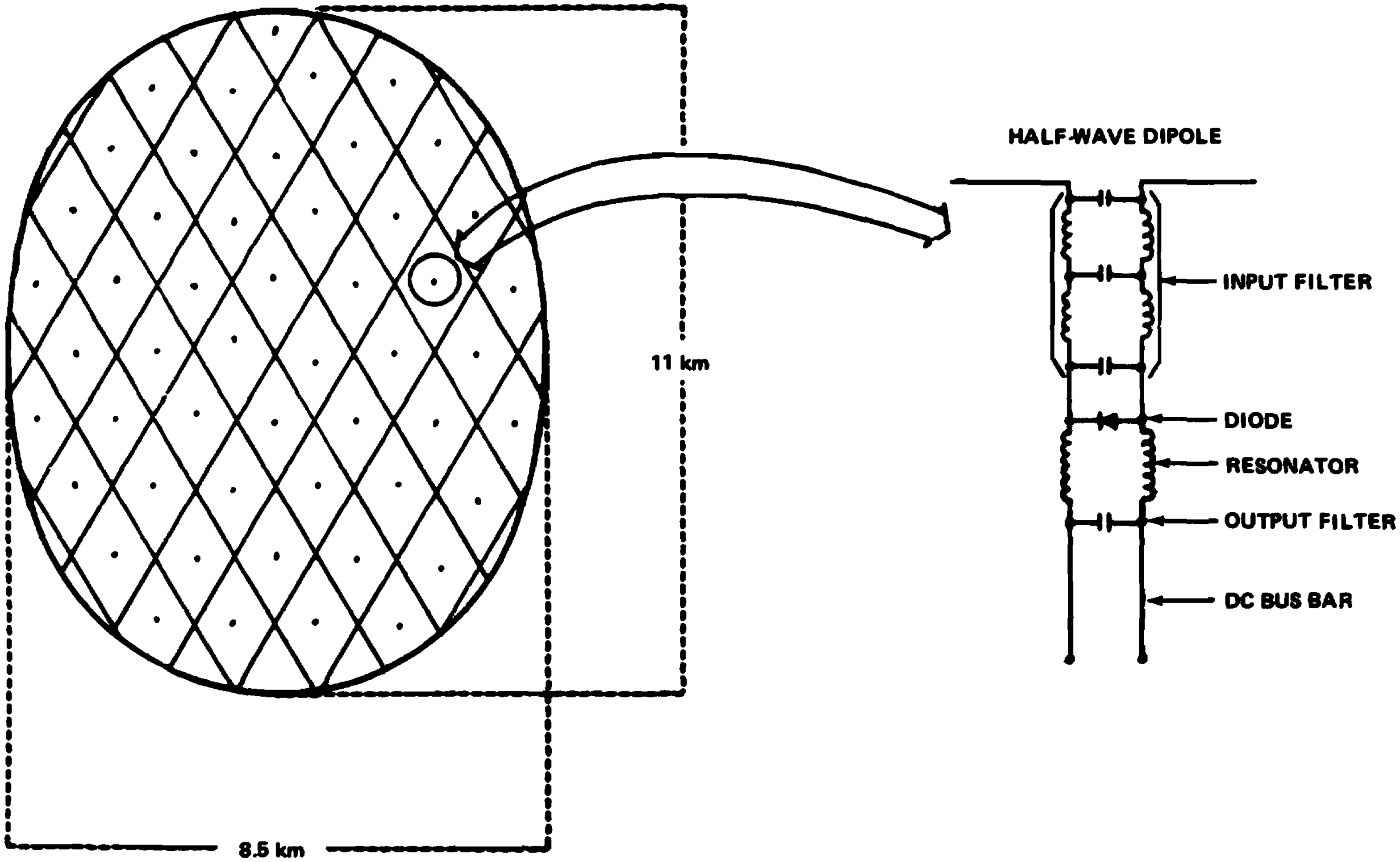


Figure 8-2. 9 dB system rectenna.

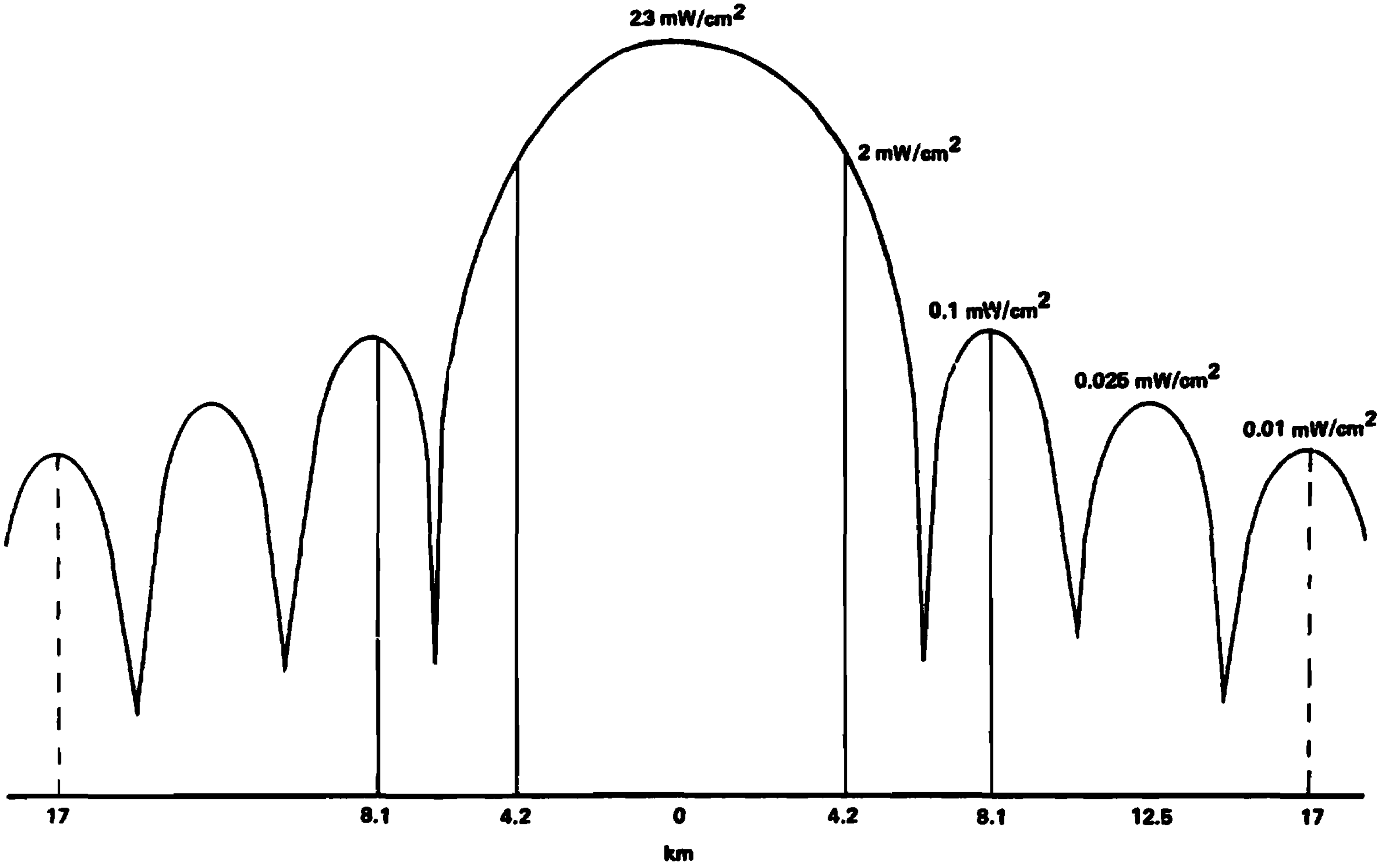


Figure 8-3. Solar power systems, 9 dB RF ground distribution pattern (1 km transmit antenna).

would be an ellipse with a minor axis diameter of 34 km, and for 0.1 mW/cm^2, it would be 16.2 km. The difference in the area for the two examples is approximately 900 km^2. Another important aspect of the rectenna is the conversion efficiency of the dipole element, which tends to fall off rapidly at low power densities. The baseline was constructed to keep the density at the edge of the rectenna at 2 mW/cm^2.

A design feature that has concerned designers of the rectenna is the waste heat of approximately 585 MW generated by a 90 percent efficiency of conversion. While this is a large amount of waste heat, it does not represent a problem to the rectenna. The 585 MW constitute an average of 40 mW per diode for each of the 13.6 billion diodes in the rectenna. This level of heat is well within the design limits of the diode.

Future analyses will attempt to derive techniques for terrestial application of waste heat in the proximity of the rectenna.

8.2.3 COMMUNICATIONS, CONTROL, AND DATA MANAGEMENT SUBSYSTEMS

The emphasis in this study has not been on the communications, control, and data management subsystems; therefore, detailed design requirements are not available. However, considering the functional requirements of subsection 8.1 and the central role of the SPS receiving site in the supervision of the system and operational activities, it becomes apparent that several communications and data management systems will be required. Typical communication links associated with space and ground interfaces have been indicated in Figure 8-1.

The greatest most complex requirements will be those for monitoring and control of the microwave beam and the numerous elements of the SPS subsystems. These must be closely coordinated and functionally integrated with the control and data management subsystems servicing the rectenna and other subsystems of the receiving site.

The command and adaptive phase control systems will be part of the central system with many sensors located in the safety zone. The size of the receiving site may dictate many intermediate data collection points, with RF relay to the central control system. If this is true, RF interference could become a problem. However, present technology should be adequate to solve such problems associated with the data management subsystem.

Communications and control interfaces are required to coordinate operations and to be responsive to utility interface demands. Considering the magnitude of the system and component numbers, it is anticipated that a highly automated data management subsystem will be needed for the utility interfaces. Similar requirements may be expected between SPS receiving sites when the system is expanded. Typically, such communications are via dedicated land lines; however, in later cases RF links may be necessary.

8.3 POWER CONDITIONING, DISTRIBUTION, AND UTILITY INTERFACES

The capacity of an SPS receiving site, rated for 5 to 10 GW, would be approximately two to three times that of the larger generating plants currently being constructed. This should be an appropriate size for the era beyond 1990. In some regions a site may be able to supply more than one utility system, while in areas with large load concentration, several sites that are linked together by a large transmission network may be needed. Such links would have requirements similar to the utility interfaces and would not affect the concepts previously described (Fig. 8-1).

Practically all the power transmitted and distributed in the United States today is 60 Hz ac. Although transmission standards may change, it is not likely that distribution systems will be changed significantly before the SPS era, because of standards established with consumers and equipment suppliers. Thus, it may be expected that the bulk of SPS power will ultimately be converted and distributed to consumers as 60 Hz, ac, three phase/one phase power at 13.2 kV.

Therefore, it was assumed in this study that 60 Hz, ac, three phase power would be transmitted from the SPS to the utility interfaces. However, the option to transmit high voltage (approximately 1 MV) dc power is retained, and technical performance/economic trade studies of ac versus dc transmission are to be performed. Typically, three to five separate transmission lines operating between 300 and 400 kV would be used for a 10 GW plant. Assuming that the utility interfaces will be within 150 mi of the receiving site, a transmission efficiency of 98 percent was projected.

The projected efficiencies of transmission and of the major power handling subsystems at the receiving site are shown in Figure 8-1. Since ac transmission was selected, the power conversion and regulation subsystem would consist of multiple, static, semiconductor dc-to-ac inverters that convert the primary power to 60 Hz, ac, three phase power and regulate the ac voltage.

Inversion efficiency was optimistically projected as 95 percent. The inverter outputs would then be transformed to the transmission voltage and routed to the transmission lines within the primary transformation/switch yard.

It is noted that, if high voltage dc transmission is selected, dc/dc conversion at the SPS site and dc/ac inversion within the utility system would be required. The two conversion steps will make it difficult for the dc approach to surpass the efficiency of the ac system.

The power conversion subsystem receives relatively low voltage dc power from the dc switch yard of Figure 8-1, which centralizes and controls the dc power received from the rectenna/power grid. The output voltage may be in the order of several hundred volts; therefore, large buses and switch gear will be required. This yard also provides essential backup protection for the rectenna and fault isolation/power rerouting capabilities for the receiving site and ground system. The efficiency of 99.8 percent indicated for the dc switch yard also includes the power required to energize the receiving site and to support the control centers.

8.4 SYSTEM SAFETY

8.4.1 SAFETY ZONE CONTROL SUBSYSTEM

An important point concerning microwave power transmission is the effect that microwave exposure may have on man and other life forms in the vicinity of the rectenna. Diverse opinions exist as to what, if any, biologically adverse effects are prevalent as a result of high power density microwave transmission. In the United States the current microwave exposure standard for man is 10 mW/cm^2. In other countries, (for example, the U.S.S.R.) the exposure level is duration dependent and is much lower at 1 mW/cm^2 for 6 s. Over an 8 h exposure period, the standard drops to 0.01 mW/cm^2. Based on the present United States standard, no safety zone around the rectenna would be required.

If the United States standard was lowered to 0.1 mW/cm^2, a safety zone representing an ellipse with a minor axis of 16.2 km and a major axis of 21.1 km would be required around the rectenna.

Since microwave exposure standards are somewhat loose in the United States, further studies should be made and more precise standards adopted with these results factored into the overall design of the SPS.

8.4.2 ON-ORBIT MAINTENANCE SAFETY

A preliminary safety analysis has been performed for several identifiable mission and task phases associated with constructing and servicing the SPS. The phases cover fabrication, assembly, preoperational checkout, and maintenance. Potentially hazardous situations were identified in virtually all phases. In view of the limited actual experience data base available in the assembly and maintenance of large spaceborne structures under zero gravity conditions, a large portion of the analysis relies on engineering judgment; further analyses are required to identify additional potentially unsafe conditions that may exist. The following assumptions are made on astronaut participation in the previously mentioned mission phases:

1. Astronaut skills are at a monitor and maintenance level.

2. Equipment used to fabricate structural members and the antenna is automated.

3. The SPS is assembled essentially by sophisticated assembly jigs supported by a few free-flying teleoperators.

4. Pre-operational checkouts are automated, and malfunctions are corrected by free-flying teleoperators.

5. Maintenance is performed by free-flying teleoperators and robot trucks.

This analysis was performed assuming that assembly operations are based in an orbiting construction facility and that maintenance operations are based in a space station attached to the SPS.

The investigative effort consisted of determining the degree of crew involvement in the mission phases in order to gain an overview of astronaut participation and man-machine-structure relationship. Then, gross potential hazards were identified. This information is listed in Table 8-1. Where astronaut participation is listed as a monitoring function, he will take the required corrective action in case of equipment malfunction. This may involve extravehicular activity or the reprogramming or manual control of a teleoperator. As hazards become better defined, special safety procedures and equipment will be evolved to protect the crew.

TABLE 8-1. OVERVIEW OF MANNED PARTICIPATION

Phase	Function	Crew Involvement	Potential Safety Hazards
Orbital Fabrication	Stock Load	Monitor	Packaging Failure Causes Equipment Damage
	Fabricate	Monitor	Material Breakage
	Remove Assembly	Operate	Jammed Manipulator
	Stock Unload	Monitor	None
Assembly Storage	Deploy and Remove Remote Manipulator System and/ or Free-Flying Teleoperators	Monitor	Collision/ High Damage Potential Jet(s) Fail to Cut Off
Storage Retrieval	Deploy/ Free/ Remove/ Stabilize Remote Manipulator System and/ or Free-Flying Teleoperator	Monitor	Failed Jet(s), Fuel Loss, and System Failure Sunlight/ Darkness Errors
Transport Segments	Stabilize Thrust	Monitor	Failed Jet(s), Fuel Loss Thrust Early/ Late Contact Other Vehicle

TABLE 8-1. (Continued)

Phase	Function	Crew Involvement	Potential Safety Hazards
Rotary Joint Assembly	Orient Secure	Operate/ Extravehicular Activity	Extravehicular Activity/ Structure Collision, High Damage Potential
	Operate	Initiate	Extravehicular Activity Irradiation
			Extravehicular Activity Tether Breaks
Antenna Segment to Segment Assembly	Install	Monitor	ailed Jet(s), Fuel Loss Premature Latch and Need for Redock Sunlight/ Darkness Extremes
	Rigging	Monitor	Cable Overload
Activate Assemblies Individually	Checkout Initiate	Monitor	
Preoperation	Final Alignment Clear Equipment	Monitor and Verify	

TABLE 8-1. (Concluded)

Phase	Function	Crew Involvement	Potential Safety Hazards
Activate Antenna	Checkout	Monitor	Electrical Shorts Microwave Leakage
Scheduled Maintenance Cycle	Refurbish/ Repair Array Components	Monitor	Electrical Shorts
	Orbital Decay Correction		Temperature, Sunlight/ Darkness Extremes
Unscheduled Maintenance	Refurbish / Repair Array Components	Monitor	Structural Failure, Microwave Leakage, Electrical Shorts
	Refurbish/ Repair Damaged Girder(s)	Maintain	Electrical Shorts, Sunlight/ Darkness Extremes

The estimated rem dosage for astronauts on board the space station or construction base is listed in Table 8-2. This is predicted for a 435 km orbit, and it is expected to be approximately the same for the geosynchronous orbit. The suggested exposure limit is given in Table 8-3. The alignment process of the SPS may involve the use of laser/reflector devices, turnbuckles, screw jacks, cables, or other tensioning apparatus to attain and maintain structural alignment. Extravehicular activity would possibly be required to finalize the alignment activities, or at least provide redundancy or a double-check. Safety measures should be taken to avoid astronaut pressure suit contact with the laser beam during alignment, the possibility of a tensioning cable breaking and striking an astronaut, and tearing of the pressure suit on sharp edges of the structure. Alignment of the antenna during power transmission is possibly the most serious safety problem of all to the astronauts. Some of the potential safety problems could be minimized by providing "protective areas" or special protective containers for the astronaut to work in or move into during the more hazardous phases of the alignment.

TABLE 8-2. ESTIMATED DOSE ON BOARD SPACE STATION (rem)

Constraint	Bone Marrow ($20\ g/cm^2$)	Skin ($10\ g/cm^2$)	Ocular Lens ($10\ g/cm^2$)	Testes ($15\ g/cm^2$)
Average Daily Dose	0.3	0.6	0.6	0.4
30 Days	9.1	17	17	12
Yearly	111	207	207	146

TABLE 8-3. SUGGESTED EXPOSURE LIMITS (rem)

Constraint	Bone Marrow	Skin	Ocular Lens	Testes
Average Daily Rate	0.2	0.6	0.3	0.1
30 Days	25	74	37	13
Yearly	75	225	112	38
Career	400	1200	600	200

The SPS will have large scale sensor networks which will be monitored by a computer. The computer will be programmed to give warnings and take precautionary action in case of hazardous malfunctions.

8.4.3 GROUND MAINTENANCE SAFETY

The rectenna field and surrounding area will be instrumented to measure the radiation levels, and a ground warning system will indicate when these rates exceed hazardous limits. These data will also be relayed to a central computer complex that has control over the beam. In case the beam exceeds the safe intensity limits within the rectenna field or outlying areas, a message will be relayed to a command point to defocus the beam. The rectenna field will be maintained by vehicles with remote manipulator systems to remove and replace failed components. The high voltage areas will be governed by procedures that conform to the national electric code.

8.4.4 COMMUNICATIONS SUBSYSTEM SAFETY APPLICATIONS

The communications link between the SPS and the ground receiving and distribution site will be a vital part of [illegible] overall system safety plan. A conventional two-way link will relay routine voice and data communications between the SPS and ground computers as they interpret the multiple large scale sensor networks on the transmitter array, the rectenna, and its associated safety buffer zone. It will not be necessary for the ground computer complex to have direct access to all SPS sensor network outputs nor for the SPS computers to access all ground sensor data. Each system will operate independently utilizing the developed or processed information from the other for control, status, and safety outputs. For example, one of the safety outputs would be in the form of a command to the SPS phase control system to defocus the microwave beam when intensity safety limits are exceeded. The "command phase control" system (subsection 7.3.6) forms a set of inputs that could result in issuance of a "defocus" command. Critical commands of this nature rely on multiple sensors or systems that are cross-referenced and linked with time delays, where appropriate. Beam defocus will be the primary means of top level safety and could be accomplished by direct command as described or initiated by an extended interruption of monitoring capability, such as communication link failure.

The large size of both SPS and the ground receiving systems will necessitate extensive monitoring, caution, and warning networks with audible and visual warnings for out of tolerance RF levels or impending shutdown conditions. Although highly automated, both systems would also call for central monitoring stations with continuous surveillance crews.

8.5 REQUIRED TECHNOLOGY ADVANCEMENTS

Technology advancements are needed in the following areas:

1. Understanding the effects of microwave radiation on the human body

2. Understanding the effects of radiation in outer space on the human body

3. Understanding safety hazards to the astronaut crew during maintenance and construction of the SPS

4. Development of procedures, codes, and equipment to protect the crew during on-orbit construction and maintenance of the SPS

5. Development of procedures, codes, and equipment to protect ground personnel during the operation of the SPS

6. Development of an improved pressure suit for astronaut use during extravehicular activity

7. Development of service vehicles for on-orbit and ground maintenance.

9.0 RESOURCE ANALYSIS AND FUEL CONSUMPTION

9.1 GROUND OPERATIONS

9.1.1 INDUSTRIAL CAPACITIES

An analysis was performed for a program of thirty 10 GW SPS's to be built over a 30 year period. A photovoltaic silicon cell concept with a mass of 116 × 10^6 kg was chosen for the industrial capacities study. The study also included the transportation and propellant requirements for both the HLLV and LEO to GEO transfer. To obtain an idea of the potential impact on, and commitment for, material resources that would be needed for a program of this magnitude, the average annual requirement was compared with current United States production in Table 9-1.

TABLE 9-1. MATERIAL REQUIREMENT AND PRODUCTION

Item	Requirement (kg × 10^6)	Current U. S. Production (kg × 10^6)	Predicted Production in 2000 (kg × 10^6)
Oxygen	3521	15 500	58 500
Hydrogen	703	155	—
Argon	44	22	—
Silicon	33	500	900
Aluminum	167	5 000	24 000
Graphite	11	—	86
Kapton	9	—	—
Copper	5	2 930	4 900

The conclusions of this analysis are as follows:

1. The requirements for oxygen, hydrogen, and argon indicate a need for expanded processing capabilities.

2. The requirements for aluminum and copper indicate little impact on the industry.

3. Silicon is the second most abundant element on Earth but is never found in the free state. The problem of manufacturing low cost solar cells is recognized as a major development effort that is not associated with material availability.

4. The production of graphite and Kapton was not assessed for this analysis.

5. If galium arsenide is chosen as an alternate to silicon for solar cells, the availability of both galium and arsenic must be considered.

9.1.2 ENERGY BALANCE ANALYSIS

An analysis was conducted to assess the energy required to produce a power satellite and the time required for an SPS to generate an equivalent amount of energy. A concentrator concept with a mass of 185×10^6 kg was used in this study since it is heavier and, hence, more energy intensive. This analysis included all energy necessary for processing of SPS materials and replacement parts and for ground transportation including both rail and truck transportation. Orbital transportation was included for both the ground to LEO and LEO to GEO phases. Rectenna and microwave transmission corridor (approximately 100 km^2) was assumed to be unavailable for agricultural use, specifically corn production.

The production of one SPS requires 4.5×10^{11} kWh of thermal energy as summarized in Table 9-2. If instead of producing an SPS this thermal energy was converted directly to electrical energy with a conversion efficiency of 30 percent, the electrical energy produced would be 1.35×10^{11} kWh. The SPS produces 8.0×10^{10} kWh electrical energy per year, consequently the SPS would return the energy in 1.7 years. This compares favorably with estimates for conventional ground systems.

9.1.3 MATERIAL RESOURCES

In general, none of the required resources are expected to be in short supply. The expected United States consumption of some materials until the year 2000 is shown in Figure 9-1 compared to United States resources. The term "Reserves" applies to resources that are known to exist and that can be mined with current technology and economics. "Identified Marginal Resources" apply to those that are known to exist and could conceivably be produced economically in the future at higher prices or with proper advances in technology. "Hypothetical Resources" attempt to quantify deposits that have so far eluded discovery in known areas of favorable ground. "Speculative Reserves" refer to new regions that have potential for new discoveries.

TABLE 9-2. ENERGY CONSUMPTION FOR PRODUCTION OF SPS

Item	Total Energy (kWh)
Satellite Mass	7.6×10^9
Ground Transportation (Satellite Materials)	4.6×10^7
Replacement Parts	8.0×10^8
Low Orbit Transport System	4.3×10^{11}
Orbit Transfer of SPS and Parts	4.1×10^8
Assembly Station (Prorated)	4.0×10^6
Rectenna and Transmission Corridor	1.5×10^{10}
Total	4.5×10^{11}

At present, the United States imports 90 percent of its aluminum as alumina and bauxite. The remainder is produced from domestic bauxite of which 70 million tons are known to exist. However, aluminum is the third most abundant element in the Earth's crust (8 percent), being exceeded only by silicon (28 percent) and oxygen (47 percent). The enormous identified resources are contained in a variety of other aluminous materials that pose technological problems. The most promising of these are high alumina clays [illegible]sonite in the rich oil shale deposits of Colorado.

United States reserves of tungsten are much less than the projected demand through the year 2000. Hence, the use of tungsten in the solar thermal and nuclear concepts was eliminated. Approximately half of the known world reserves would have been required for this application (most of which are in China). Molybdenum which is far more abundant was chosen as an alternate.

For the photovoltaic baseline concept, the solar cells were assumed to be silicon. If galium arsenide is chosen as an alternate, the availability of both galium and arsenic must be considered.

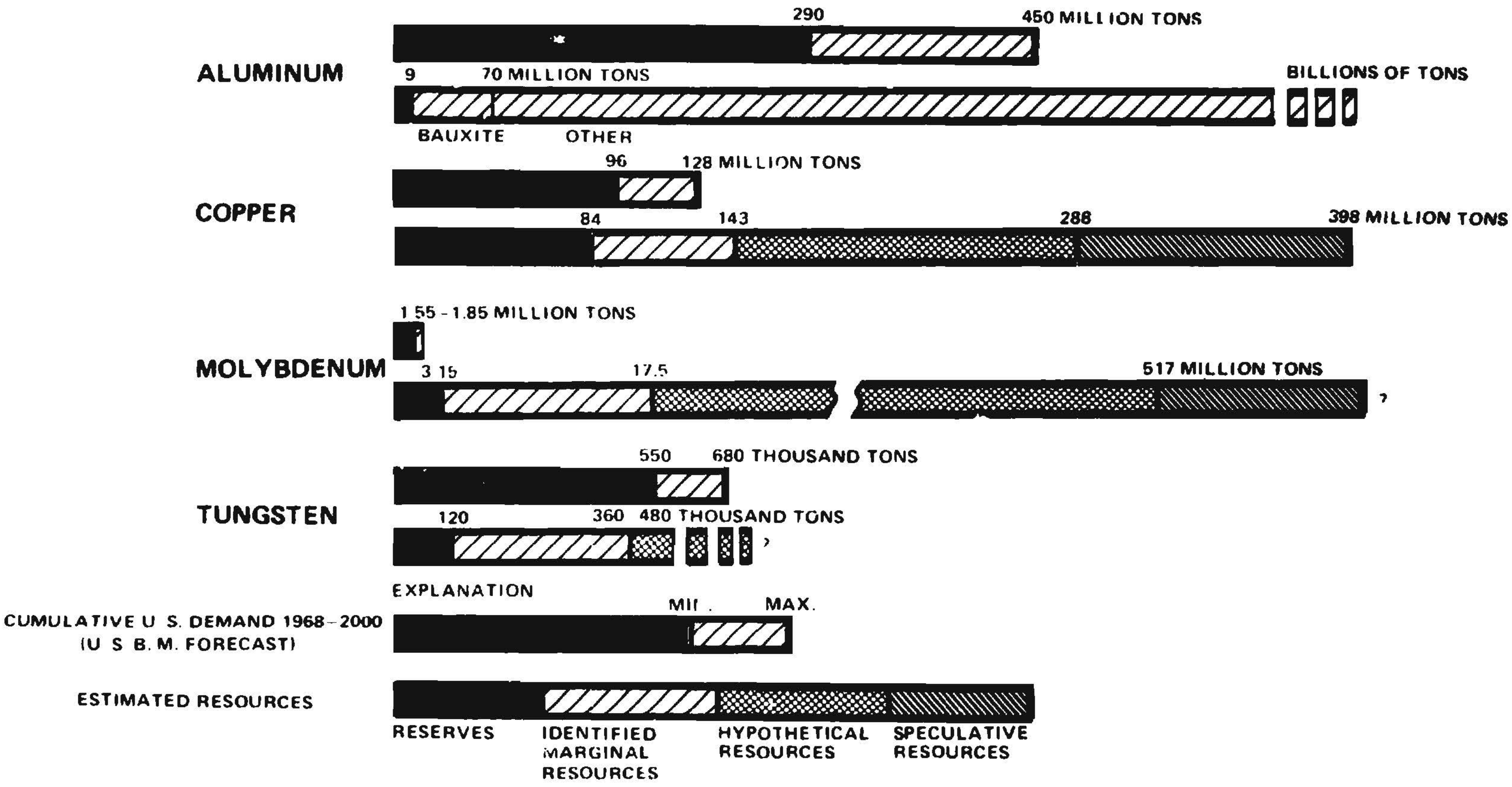


Figure 9-1. United States resources compared to projected demand through the year 2000.

9.2 TRANSPORTATION OPERATIONS

9.2.1 ASSUMPTIONS AND ANALYSIS

The following assumptions were made as guidelines in the transportation analysis and LEO versus GEO assembly trades:

- One 10 GW SPS goes on line per year for 30 years.

- An SPS assembled in LEO is 24 percent heavier than one assembled in GEO because of the orbit transfer engines and propellant.

- A photovoltaic SPS assembled in LEO has 11 percent more solar cells than one assembled in GEO because of solar cell degradation during the LEO to GEO transfer.

- A photovoltaic SPS assembled in GEO requires less attitude control propellant per year due to its smaller size.

- An SPS assembled in GEO requires 88 persons for 330 days and then goes on line. Work on the next SPS begins 35 days later, after the assembly facility and equipment are refurbished.

- An SPS assembled in LEO requires 120 persons for 330 days and then 60 days for self-propelled orbit transfer. During the ascent phase, the assembly facility and equipment are refurbished for 35 days and then work begins on the next SPS.

- The SPS is assembled with the aid of a local space transportation system. This system was not included in the transportation analysis but is described in subsection 12.2.6.

- In the LEO assembled SPS scenario, the POTV delivers an initial crew to the SPS to prepare for operation after the SPS has passed through the radiation zone. A COTV later returns all construction equipment from GEO to LEO.

- The GEO operations crew size is 12 persons per SPS.

- Each person stays in orbit for 1 year, and one-half of each crew is rotated every 6 months.

- Each SPS in GEO has a space station and logistics depot connected to it.

- In LEO, the logistics and OTV engine and propellant depots are in an orbit different from the construction depot so that there are two HLLV launch windows per day.

- The HLLV delivers its payload into an 80 × 435 km orbit. The kick or second stage of the HLLV may circularize, rendezvous, and dock the payload with the appropriate depot, or a LEO tug may perform these functions.

- The COTV retrieves its payload from a LEO depot and delivers it to GEO.

- The shuttle rotates personnel between Earth and a LEO space station.

- The POTV rotates personnel between LEO and GEO.

9.2.2 LOGISTICS EQUIPMENT AND TRANSPORTATION

A transportation analysis has been performed for a 30 year program with either a photovoltaic or thermal SPS assembled in LEO or GEO. The mass of the photovoltaic system assembled in LEO is assumed to be 144.78 × 10^6 kg with logistics resupply requirements of 1.36 × 10^6 kg/year. The photovoltaic system assembled in GEO is assumed to have a mass of 107.48 × 10^6 kg with logistics resupply requirements of 1.16 × 10^6 kg/year. The thermal SPS assembled in LEO is assumed to have a mass of 274.46 × 10^6 kg, while the GEO assembled version has a mass of 221.34 × 10^6 kg. The logistics resupply is 0.85 × 10^6 kg/year for both.

For each SPS case outlined above, nine transportation cases were analyzed as follows:

1. Gas Core Reactor (GCR) POTV
    a. GCR COTV
    b. Single Stage to Orbit (SSTO) HLLV
2. Chemical POTV
    a. GCR COTV
    b. SSTO HLLV

3. Chemical POTV
    a. Chemical COTV
    b. SSTO HLLV
4. GCR POTV
    a. GCR COTV
    b. Two Stage HLLV, Class 4
5. Chemical POTV
    a. GCR COTV
    b. Two Stage HLLV, Class 4
6. Chemical POTV
    a. Chemical COTV
    b. Two Stage HLLV, Class 4
7. GCR POTV
    a. GCR COTV
    b. Two Stage HLLV, Class 5
8. Chemical POTV
    a. GCR COTV
    b. Two Stage HLLV, Class 5
9. Chemical POTV
    a. Chemical COTV
    b. Two Stage HLLV, Class 5

Specific characteristics of the vehicles mentioned are found in Table 9-3. Each case assumes that a shuttle can deliver 68 passengers from Earth to LEO and return.

TABLE 9-3. DESCRIPTION OF VEHICLES

HLLV	COTV	POTV
• Single Stage to Orbit, Class 4 Payload = 204 545 kg Turnaround = 127 h Includes a Kick Stage	• Gas Core Reactor Payload = 363 000 kg Propellant = 159 000 kg Vehicle = 37 000 kg Engine = 32 000 kg Life of Engine = 10 Flights	• Gas Core Reactor Capacity = 6 Persons Propellant = 1512 kg Vehicle = 352 kg Engine = 305 kg Life of Engine = 10 Flights
• Two Stage, Class 4 Payload = 204 545 kg Turnaround = 132 h	• Chemical Payload = 2882 kg Propellant = 22 818 kg Vehicle = 2404 kg Engine = 201 kg Life of Engine = 12 Flights	• Chemical Capacity = 6 Persons Propellant = 27 382 kg Vehicle = 2882 kg Engine = 241 kg Life of Engine = 12 Flights
• Two Stage, Class 5 Payload = 409 091 kg Turnaround = 132 h		

The results of the transportation analysis for the LEO assembled SPS's are given in Figure 9-2; the results for the GEO assembled SPS's are given in Figure 9-3.

Several material depots and personnel space stations are required to support the transportation system and assembly in LEO or GEO of an SPS. In the LEO assembly scenario, a construction depot and space station along with OTV propellant, OTV engine, personnel exchange station, and logistics depots are required. In GEO a logistics depot and space station must be attached to each SPS.

The GEO assembly scenario requires construction, OTV propellant, OTV engine, and logistics depots in addition to a personnel exchange station in LEO. In GEO, there is a requirement for a construction depot with a space station along with a logistics depot and space station attached to each SPS.

Results of depot and space station capacity studies for a LEO or GEO assembled photovoltaic SPS are given in Table 9-4.

9.2.3 TRANSPORTATION LOADING AND VEHICLE OPERATIONS

An analysis was performed to determine the mass of materials that must flow through the launch sight to build one SPS and operate it for 1 year. For this analysis, a photovoltaic SPS assembled in LEO was chosen along with

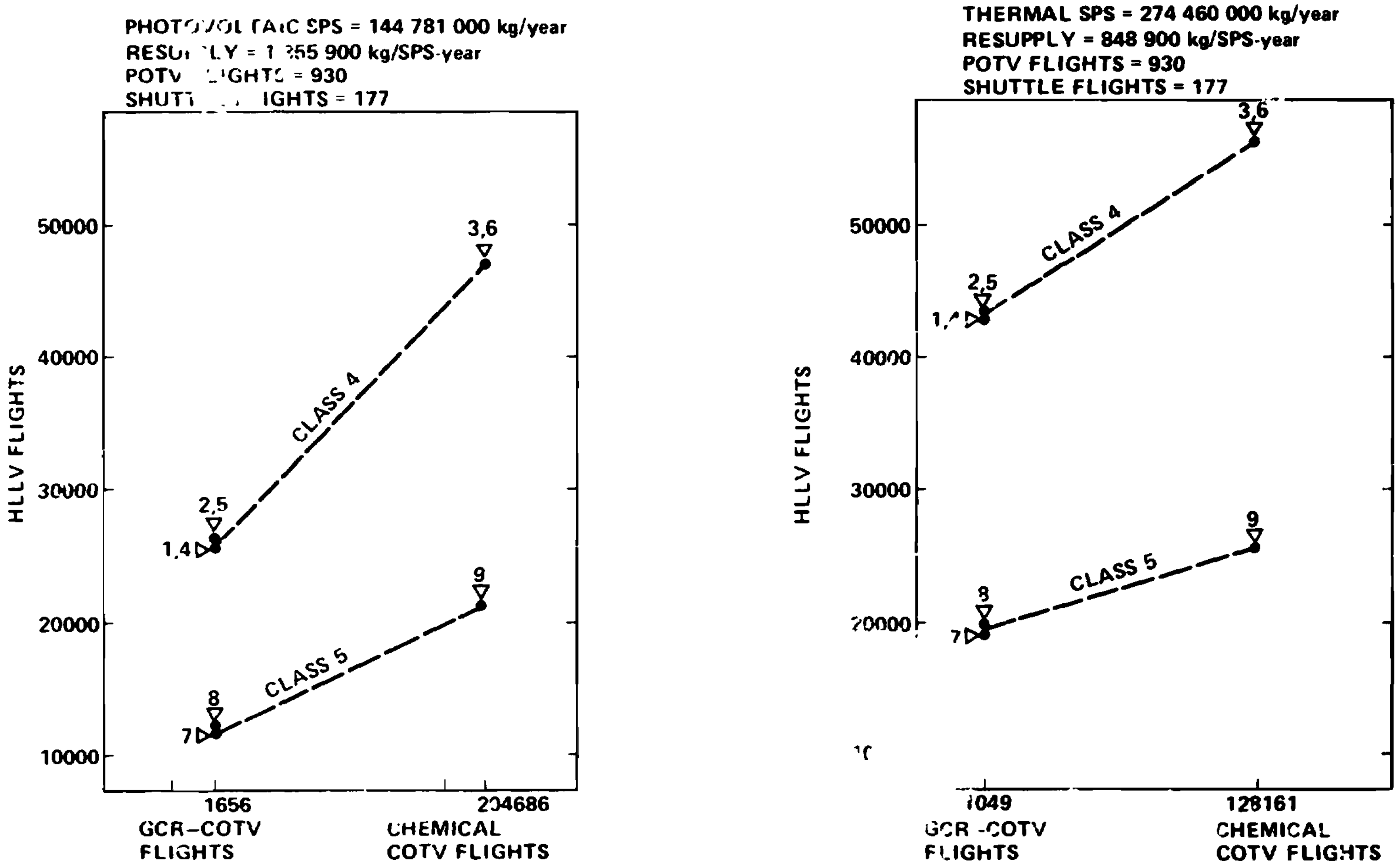


Figure 9-2. LEO assembly and operations and maintenance of the SPS with one on-line per year for 30 years.

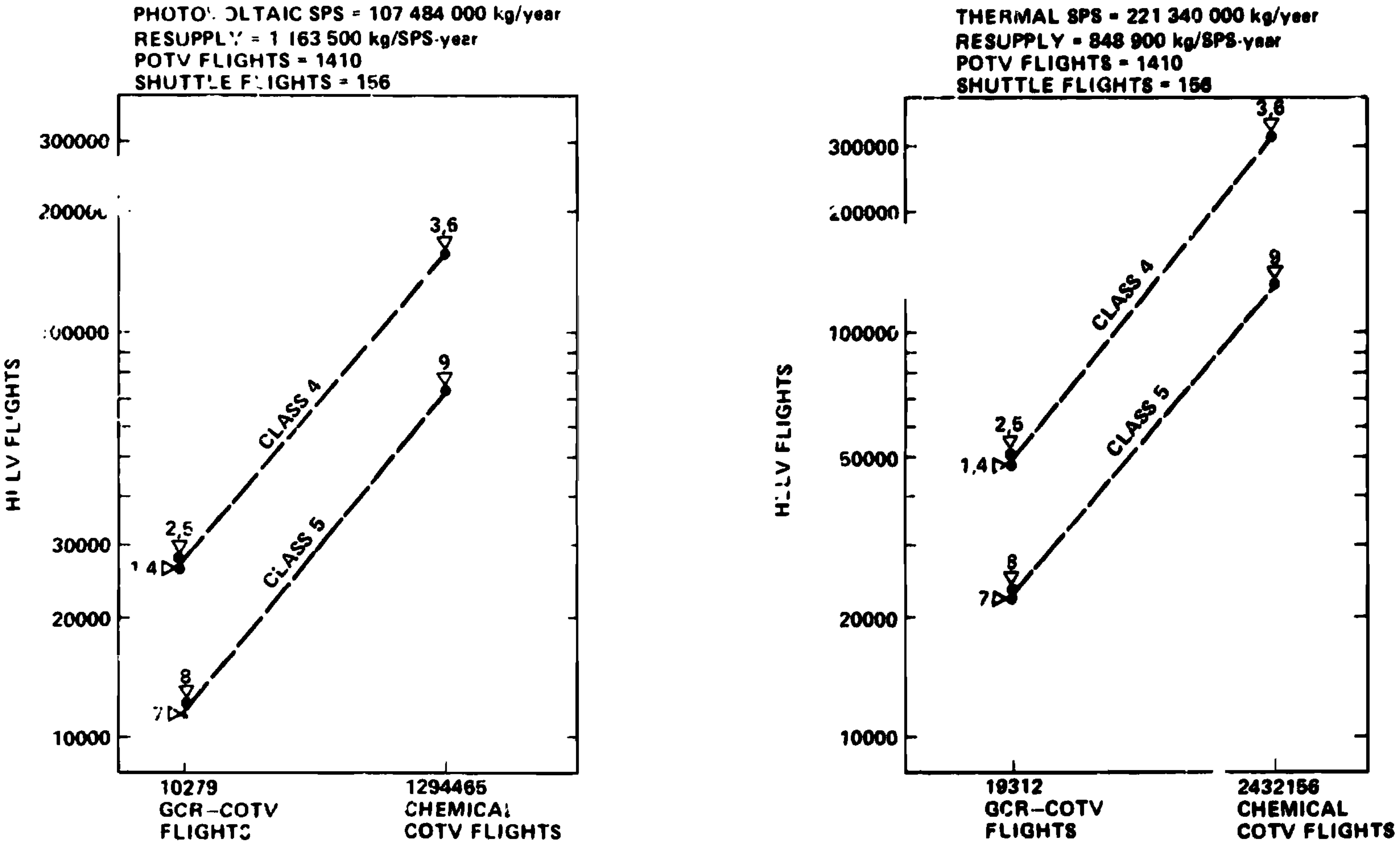


Figure 9-3. GEO assembly and operations and maintenance of the SPS with one on-line per year for 30 years.

TABLE 9-4. DEPOT AND SPACE STATION CAPACITIES FOR PHOTOVOLTAIC SPS

Facility	LEO Assembly		GEO Assembly	
	Number Required	Maximum Capacity	Number Required	Maximum Capacity
LEO Construction Depot	1	60.00×10^6 kg	1	2.50×10^6 kg
LEO Construction Space Station	1	120 Persons	0	—
LEO OTV Support Depot				
COTV Propellant Section	1	2.00×10^6 kg	1	4.00×10^6 kg
COTV Engine Section	1	0.35×10^6 kg	1	0.53×10^6 kg
POTV Propellant Section	1	2.00×10^6 kg	1	3.00×10^6 kg
POTV Engine Section	1	0.22×10 kg	1	0.35×10^6 kg
Logistics Section	1	13.60×10^6 kg	1	13.60×10^6 kg
LEO Personnel Exchange Station	1	68 Persons	1	68 Persons
GEO Construction Depot	0	—	1	55.00×10^6 kg
GEO Construction Space Station	0	—	1	88 Persons
GEO Logistics Depot	1 SPS	5.00×10^6 kg	1/SPS	5.00×10^6 kg
GEO Operations Space Station	1 SPS	12 Persons	1/SPS	12 Persons

a class 4 two stage HLLV, a GCR COTV, a GCR POTV, and a 68 passenger shuttle. (Refer to Table 9-3 for vehicle descriptions.) The results of the analysis for the buildup year and 1 year of operation are contained in Table 9-5. Operational flow diagrams for the HLLV, shuttle, COTV, and POTV are found in Figures 9-4 through 9-6.

TABLE 9-5. MASS FLOW THROUGH LAUNCH SITE FOR ONE PHOTOVOLTAIC SPS

	Buildup Year		Each Year of Operation	
	Mass Flow (kg × 10^6)	Flights	Mass Flow (kg × 10^6)	Flights
HLLV				
Construction	144.78	707.82	0	0
Logistics	0	0	1.36	6.63
OTV Propellant	0.16	0.79	0.60	2.92
OTV Engines	0.38	1.86	0	0
OTV Stage and Engine	0.07	0.34	0	0
Subtotals	145.39	710.81	1.96	9.55
HLLV Propellant	3637.32		48.87	
Shuttle				
Crew Rotation	0.06	3.00	0.005	2.00
Shuttle Propellant	5.14		3.43	
Total Weights	3787.91/SPS Constructed in LEO		54.265/SPS Year in Operation	
COTV Flights	1.00 (Return from GEO to LEO)		3.74/SPS-Year	
POTV Flights	2.00 (Initial Crew Delivery)		2.00/SPS-Year	

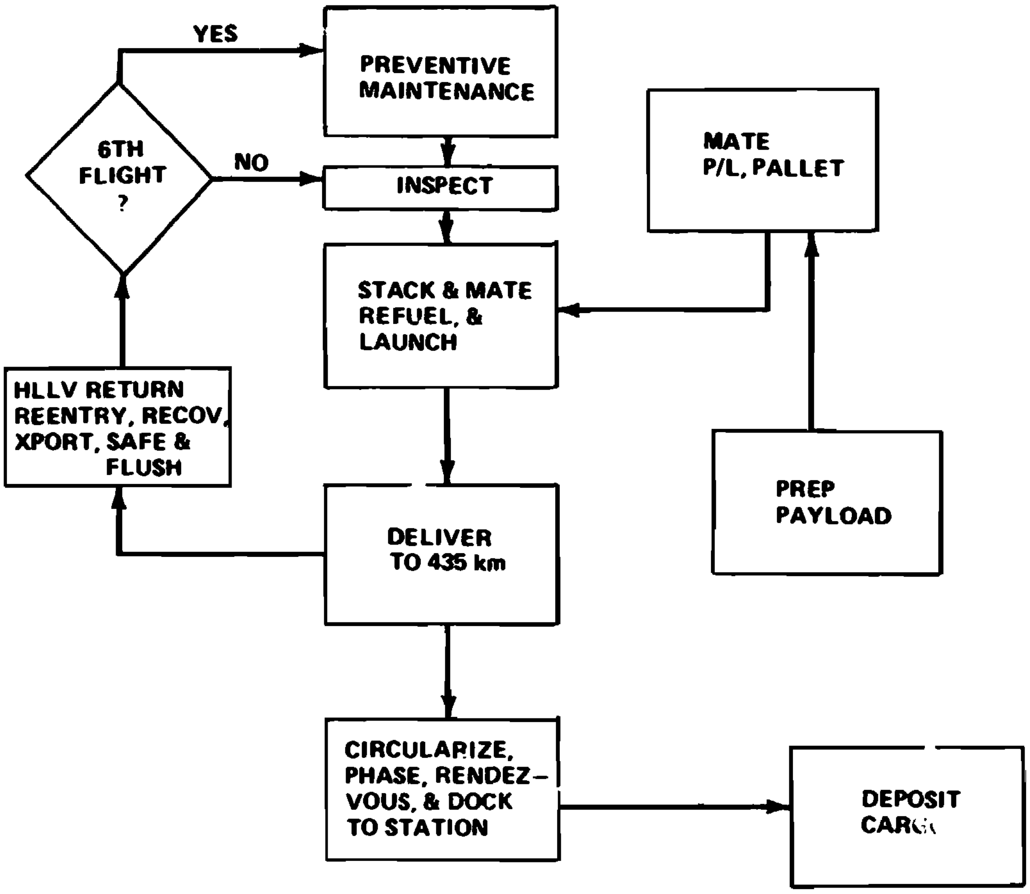


Figure 9-4. HLLV operational flow diagram.

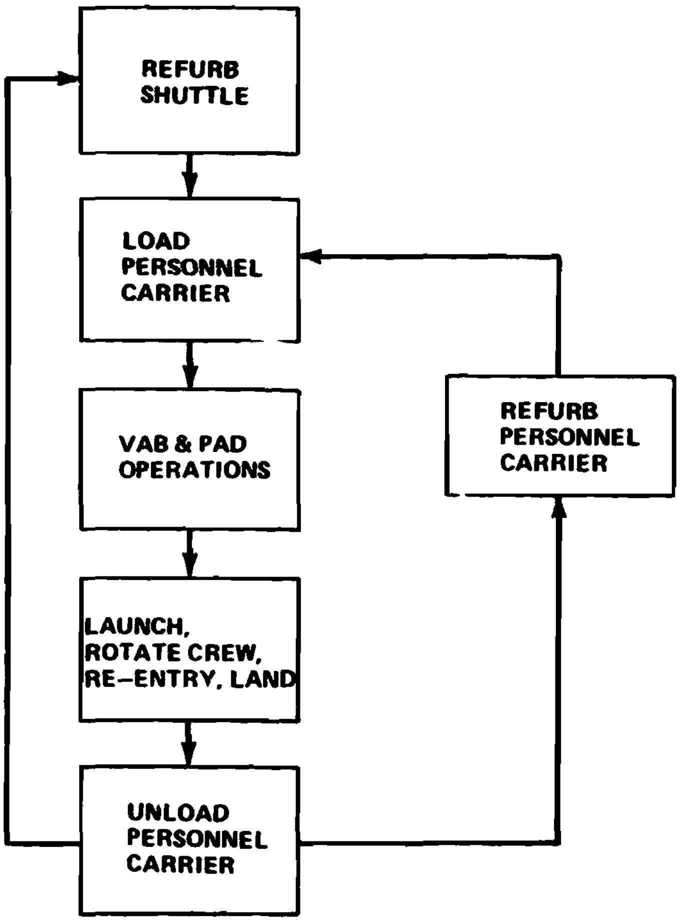


Figure 9-5. Shuttle operational flow diagram.

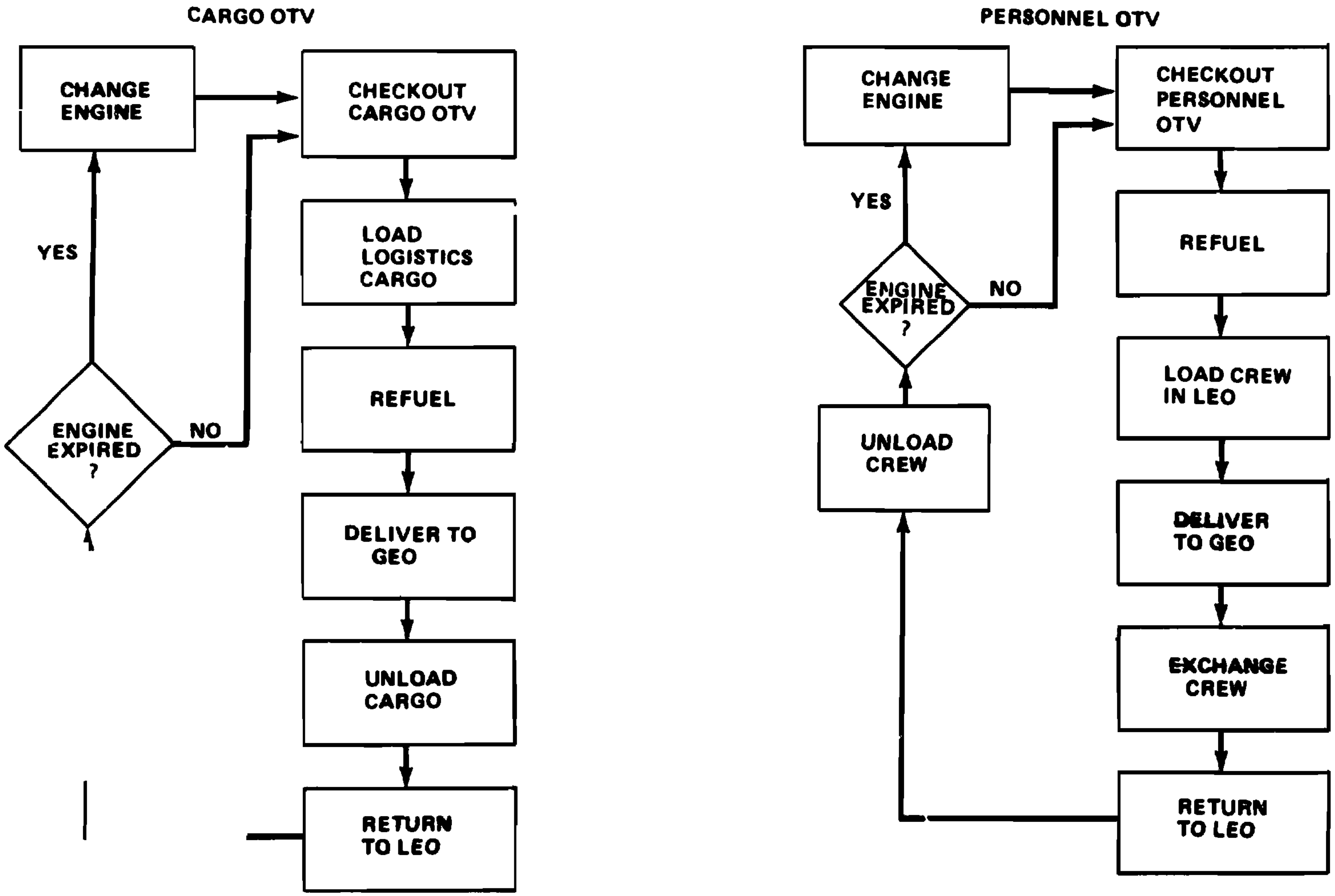


Figure 9-6. OTV operational flow diagrams.

9.2.4 TRAFFIC CONTROL AND INVENTORY MANAGEMENT

Primary traffic around the SPS construction area will be in support of delivery of items of inventory to or from storage. Traffic control, however, must begin on the ground with packaging of materials for delivery to the SPS. With one launch opportunity per day, several HLLV payloads may simultaneously be placed into suitable phasing orbits that may require several days of phasing time before final rendezvous can begin. Thus, as many as a dozen payloads may be in phasing orbits waiting for rendezvous opportunities at any time. Similar situations exist in coordinating operations of shuttle crew delivery, LEO to GEO and return traffic, as well as local delivery of material and products to the SPS assembly sites. Even without a clear concept of the elements involved, functional requirements can be somewhat defined, and they indicate the requirement for a high degree of automatic traffic control, large scale automatic collision avoidance equipment and procedures, and rapid contingency assessment and reaction capability.

9 .5 DISPOSITION OF UNUSABLE ITEMS

During the buildup, construction, and operation of the SPS, there will be many disposable items accumulated on-orbit that must have proper disposal. These items will consist of propellant storage tanks, OTV engines, unusable beams, amplitron hardware, depot facilities, spools, unusable solar blankets, etc. Since these articles must not be allowed to accumulate haphazardly in space, a temporary storage facility must be provided until they can be returned to Earth or disposed of otherwise. Storage and, certainly, return of these items in their present forms would probably be prohibitive. To alleviate this problem, a compacting or smeltering facility should be provided to improve packaging efficiency for both temporary storage and later return to Earth. Another approach to the disposable items would be to build them in such a manner as to have a secondary application. For example, propellant storage tanks might be converted for use as a temporary shelter, or they might be attached to provide some structural element for the SPS.

9.3 MANUFACTURING, CONSTRUCTION, AND MAINTENANCE OPERATIONS

9.3.1 REQUIREMENTS AND GUIDELINES

The prime fabrication and assembly location in LEO is at an altitude of approximately 435 km because of logistics and cost considerations. An HLLV will deliver cargo into the construction orbit, and another vehicle will deliver

the cargo to the construction and fabrication sites of the SPS. All equipment and facilities, except transportation equipment, will remain attached to the SPS throughout construction. Construction equipment and facilities, including the construction facility power sources, will remain in LEO after completing an SPS and will be utilized for subsequent SPS assembly. One SPS will be delivered to GEO each year for 30 years. Figure 9-7 shows a candidate schedule for assembly and delivery of three consecutive SPS's. Allowing 1 month for construction facility refurbishment leaves 330 days for accomplishing assembly. The rates of assembly required to meet this schedule imply that as much ground fabrication be done as is consistent with high density HLLV cargos and that the remainder be fabricated and assembled with automatic equipment in orbit. Technology development should begin very soon to develop cognitive robots for automatic assembly and automatic maintenance.

9.3.2 CONSTRUCTION

A discussion of the construction of a photovoltaic SPS is presented here. Assembly would begin with construction of the center portion of the power conduction mast which will be the backbone of the SPS. Fabrication of the arrays and rotary joint will begin after approximately 40 days. The arrays will be assembled around a clam shell type jig (Fig. 9-8). Beams are assembled around one-half of the jig which rotates the structure into position to be attached to the existing structure. While this half is being attached and outfitted with blanket packages and reflector packages, the other half of the jig rotates out of the previous cell and begins assembly of the next cell. An 8 h jig cycle can assemble the 800 array cells and the 100 dielectric cells in approximately 300 days. Guide wires for later deployment of reflectors and blankets are installed as shown in Figure 9-9. Blankets are in folded packages; reflectors are in rolls and installed at opposite ends to avoid interference. Beam fabrication could be performed either inside the jig or at some central site and transported to the jig. Rotary joint elements are likely to be somewhat unique, lacking a repetitive type of operation and requiring more man involvement either directly or via teleoperators. Antenna fabrication can begin after some progress has been achieved on the rotary joint. Figure 9-10 shows a top level allocation of the 330 days allowed for construction and indicates the quantity of material involved in each task.

Assembly of the antenna structure would be similar to the array assembly in that beams would be assembled around a jig or scaffolding. Alignment tolerances would probably prevent use of a flip-flop jig, and jig movement by smooth linear motion probably would be preferred. Antenna structural beams and subarray units should be fabricated or constructed at a central site of the

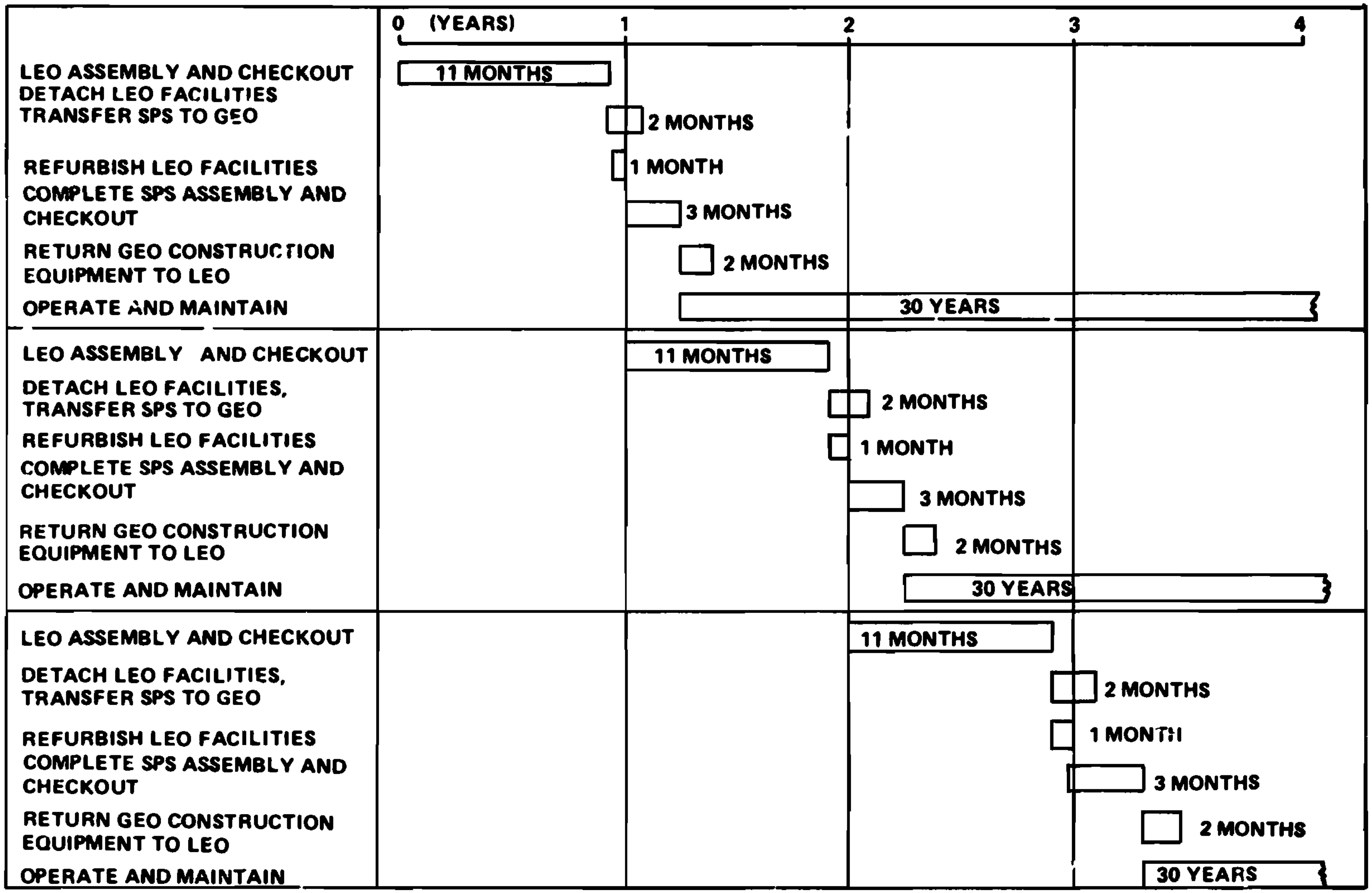


Figure 9-7. SPS construction schedule, one per year for 30 years.

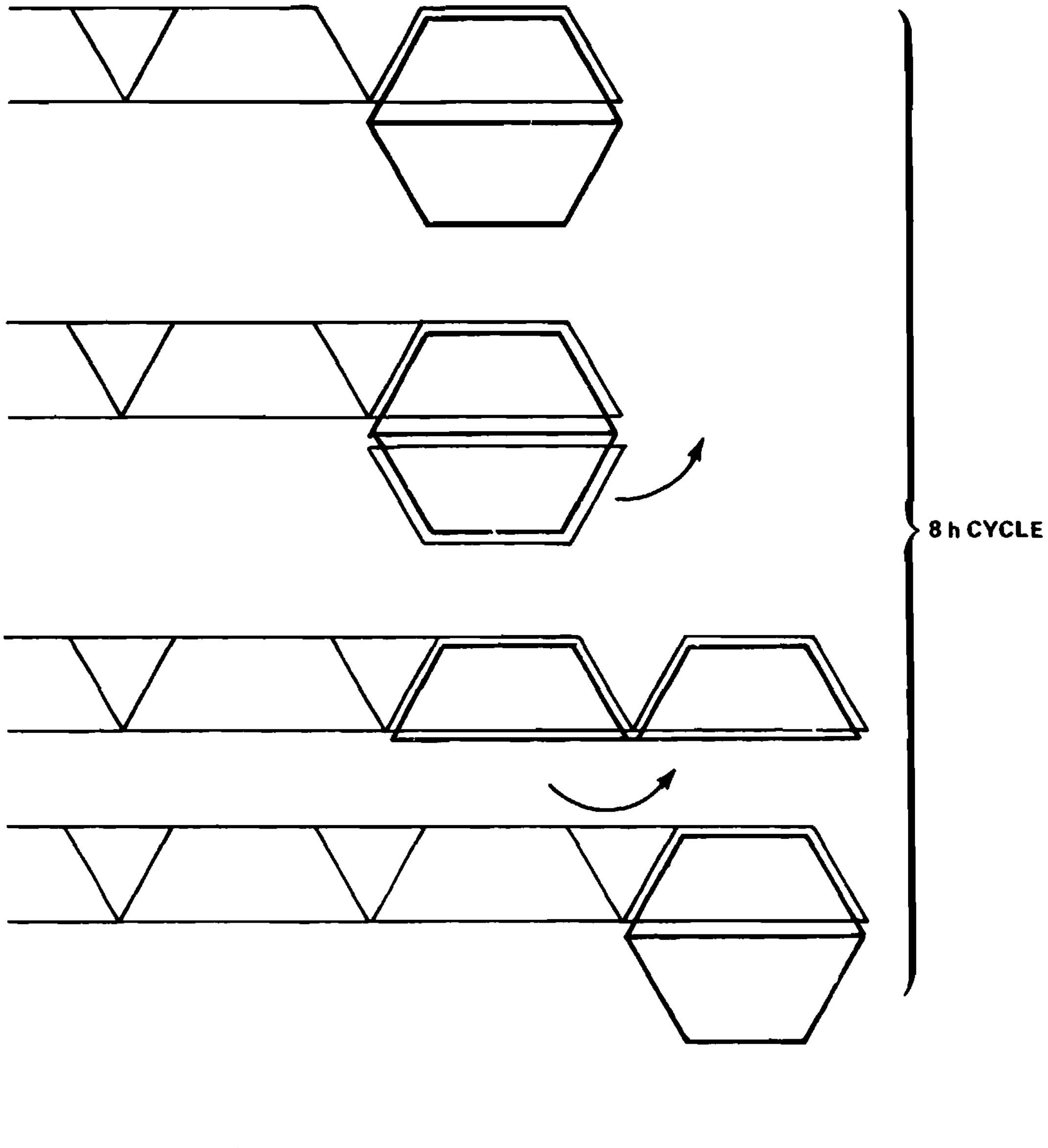


Figure 9-8. Array construction jig sequence.

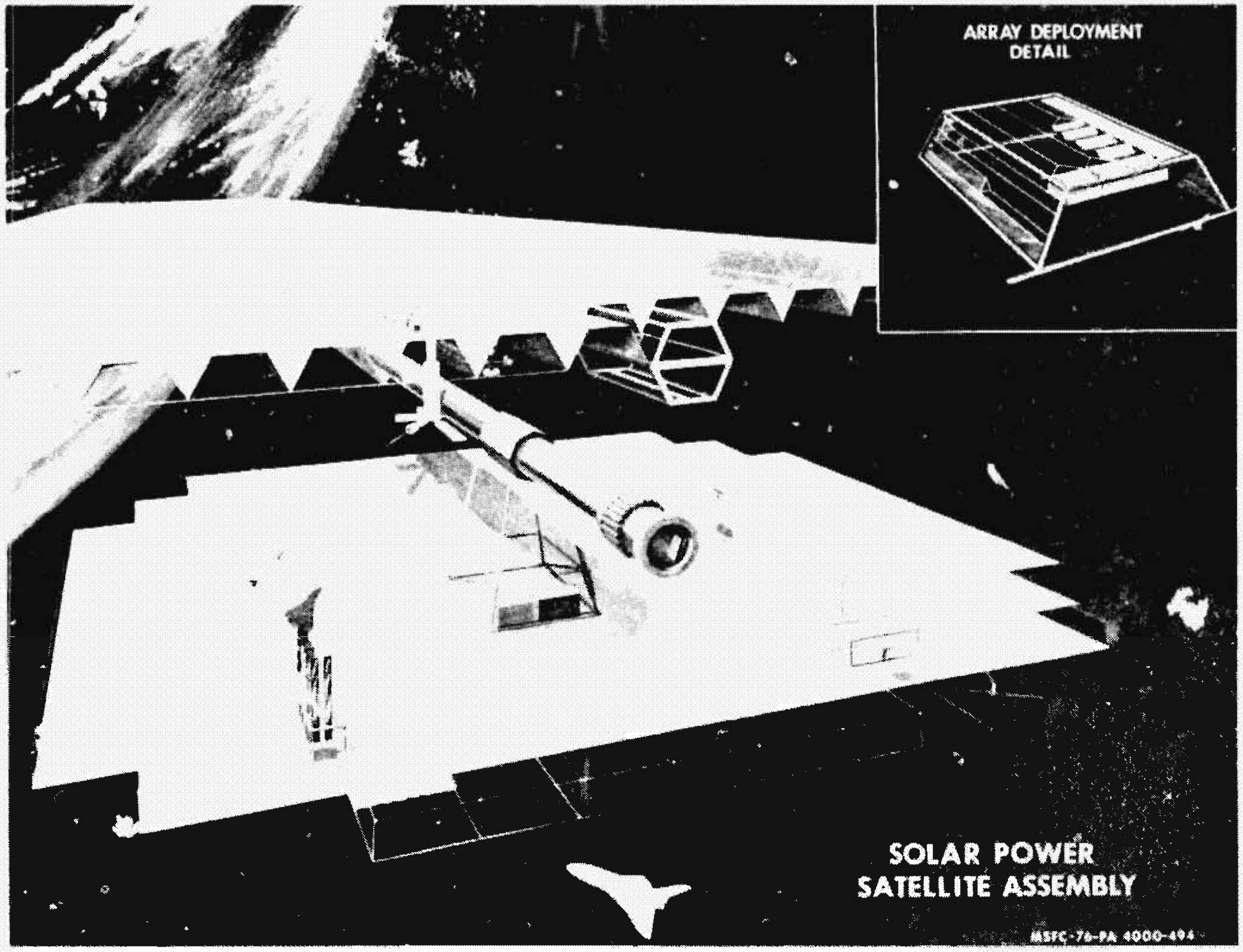


Figure 9-9. SPS assembly.

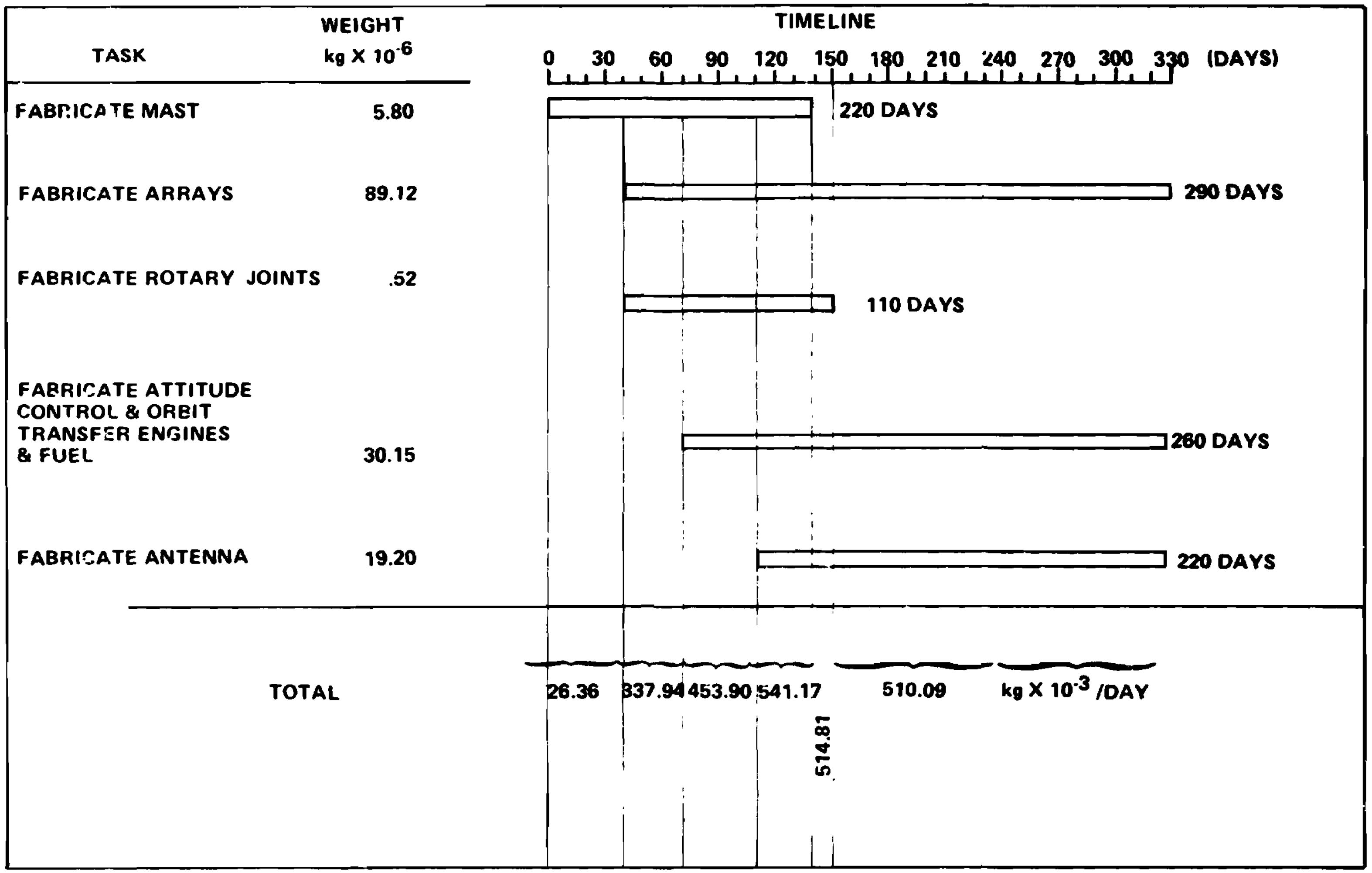


Figure 9-10. SPS assembly timeline.

antenna, probably inside the standoff mast, and transported to the assembly site with some type of LTE. Power for antenna fabrication and assembly may be obtained from deployment of a portion of the solar array, which should be available by the time the antenna assembly begins. After construction has been completed, 22 percent of the arrays will be deployed to provide power to the MPD orbit transfer engine. A GEO space station and control center would be attached to and carried to GEO by the SPS and would receive the initial crew as soon as the SPS had passed the radiation zone. The crew would then begin deployment of the rest of the arrays. Since the blanket and reflector packages, guide wires, deployment cables, and winches have already been installed, this process simply requires activation of the winches and troubleshooting those that do not function properly. The SPS should therefore be ready to begin operation shortly after achieving proper GEO position and orientation.

9.3.3 MANUFACTURING AND CONSTRUCTION EQUIPMENT

Table 9-6 lists the assembly equipment and the facilities, with their estimated weights, required to assemble an SPS, and Figure 9-11 shows some of them. The power mast construction equipment consists of a large ring (approximately 60 m in diameter) containing approximately 20 small beam forming devices for "extruding" stringers and devices for winding and welding a ribbon of skin around the stringers. The skin may be multilayered to accommodate positive and negative current flow. Spools of skin and stringer material may be resupplied by free-flying remote controlled teleoperators.

Structural beams for the solar arrays, dielectric structure, and antenna will be fabricated on-orbit by groups of machines similar to the one shown in Figure 9-11. Beams fabricated by these machines may be further assembled into larger beams by larger automatic machinery. Material for these devices will be delivered on spools or reels.

The structural concept will require a highly automated manufacturing and assembly process. Fully automated manufacturing, delivery, assembly, and checkout systems, all designed to perform with a high degree of precision and reliability, will be required. All systems will be designed with a fail-safe philosophy. Free flyers, end effectors, manipulators, cable layers, etc., should be programmed with sufficient decision-making capability to normally operate without men in the loop. Tools and assembly techniques developed for the SPS assembly should find many applications in other space fabrication endeavors.

Waveguides may have to be fabricated on orbit because of their low density and, if so, will require extremely precise equipment for forming, fastening, and drilling or punching. This equipment should probably be located on the standoff mast near the antenna. The antenna subarray assembly equip-

TABLE 9-6. SPS ASSEMBLY EQUIPMENT AND FACILITIES

Power Mast Construction Equipment	60 000 kg
Assembly Jig	120 000 kg
Solar Array Beam Fabrication Facility	67 000 kg
Dielectric Beam Fabrication Facility	1 000 kg
Waveguide Fabrication Facility	7 000 kg
Antenna Subarray Fabrication Facility	4 000
Antenna Unit Assembly Equipment	10 000 kg
Checkout Equipment	4 000 kg
Local Transportation Equipment	100 000 kg
Maintenance Hangar and Repair Station	87 000 kg
Blar Deployment Equipment	20 000 kg
Reflector Deployment Equipment	20 000 kg
HLLV Cargo Retrieval Vehicles	24 000 kg
Vehicle Refueling Station	1×10^6 kg
Space Station	1×10^6 kg
Cargo Docking and Storage	12 000 kg
Large Assembly Equipment and Checkout Area	12 000 kg
Construction Power Units	25 000 kg
GEO Space Station	100 000 kg
	2 676 000 kg

ment will assemble the waveguides, amplitrons, power conditioners and distribution network, phase control electronics, alignment controls and jacks, instrumentation, etc., into elements that can be delivered to and installed on the antenna structure.

Checkout equipment will be divided into several categories, i.e., those required for solar arrays, the power conduction mast, antenna subarray elements, free-flyer diagnostics, and manufacturing equipment diagnostics. A free-flying package may be required to verify phase front control on the antenna subarrays after installation.

C-3

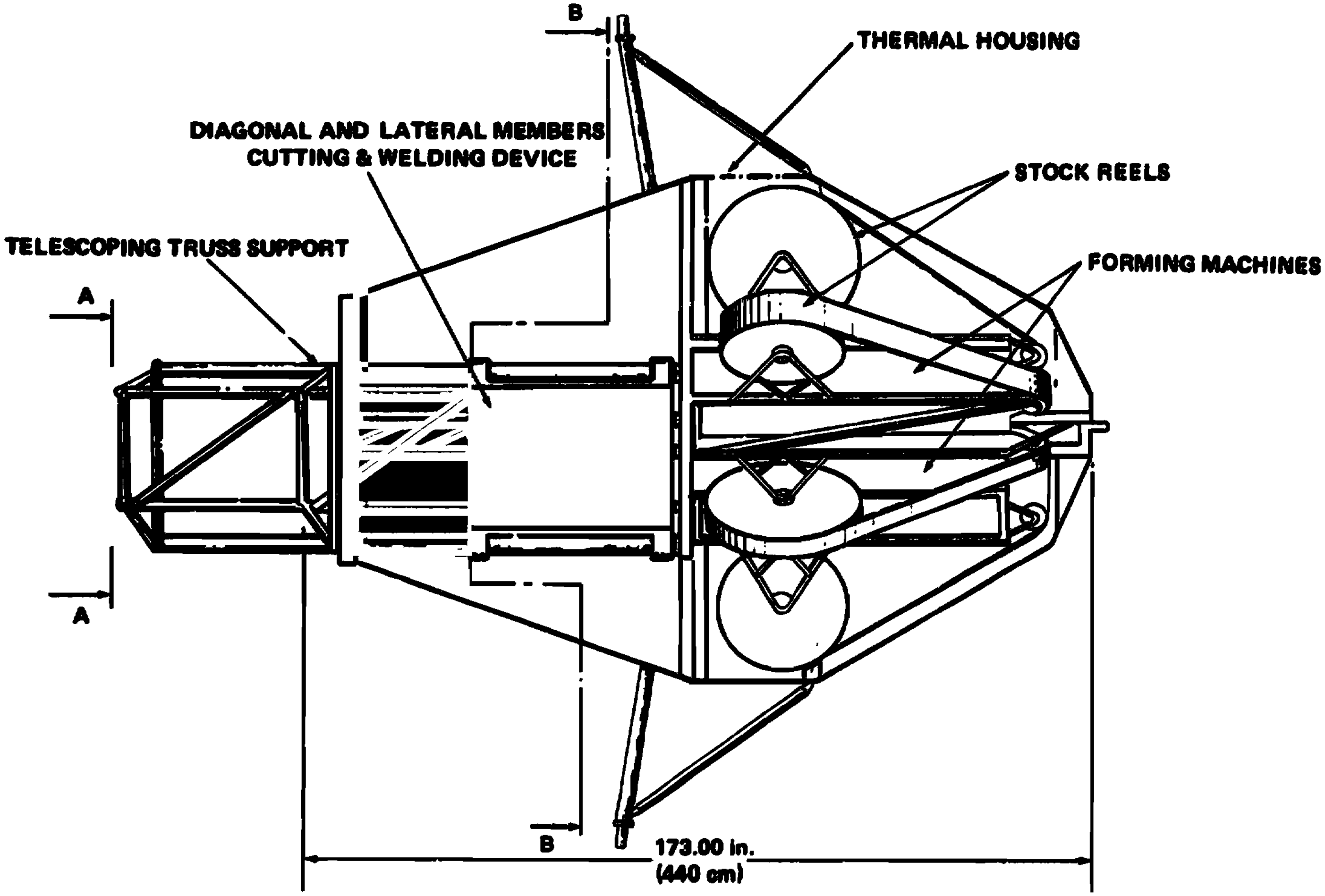


Figure 9-11. Fabrication module design concept.

Local transportation equipment is also divided into several categories and is described in subsection 12.2.6.

The cargo docking and storage areas and vehicle refueling stations were sized from logistics models as described in subsection 9.2.

Manpower for SPS manufacturing and construction is shown in Figure 9-12.

9.3.4 TERMINAL FACILITY

Because of the large amount of supplies involved, a terminal facility or depot must be provided to store and centralize many of the materials used for construction of the SPS. This facility would be capable of storing tanks of propellants, lubricants, cryogenics and other fluids, beams, solar blankets and reflectors, spare OTV engines, etc. Docking ports would be provided on the facility to accommodate both shuttle and tug vehicles. The depot would also be equipped with cranes, mainpulator arms, elevators, rail systems, etc., for storage and retrieval of supplies and materials. No attitude control system is needed for the depot, since it is attached to the space station. Table 9-4 shows the capacities of the various depot departments for both LEO and GEO facilities.

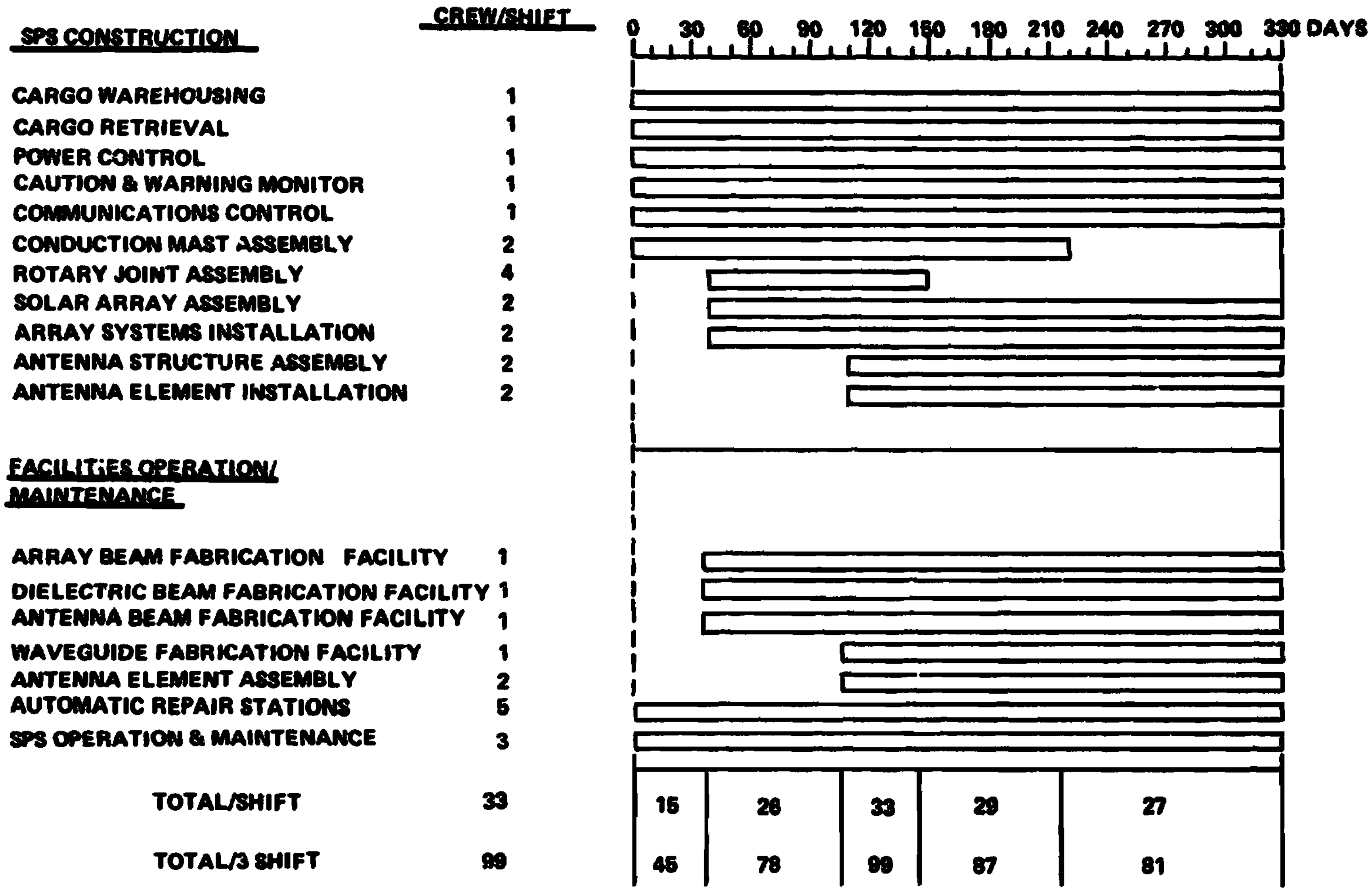


Figure 9-12. SPS assembly crew.

10.0 POWER MANAGEMENT [20-22]

The objective of the power management function is the central control, coordination, and integration of SPS status, operational maintenance base activities, SPS logistics, and ground receiving and distribution management required to sustain an operational SPS system.

10.1 REQUIREMENTS AND ANALYSIS

The following requirements are established for power management:

- Continuous status management of all operational SPS's
- Management of all operational maintenance bases for SPS
- Management of all operational logistics
- Management of the ground receiving and distribution system
- Interfacing with SPS manufacturing and construction management.

10.2 MANAGEMENT CONCEPTS AND FUNCTIONS

Management concepts depend on certain criteria that can be derived from overall program requirements (Table 10-1). The Grumman "level of authority" organizational pyramid model has been assumed as fulfilling the requirements for power management. The levels of management, their structure, and power management functions can be identified as follows (Fig. 10-1):

- SPS status management involves a complete display and monitoring of all vital operational functions of each SPS, the operation and maintenance of communication with SPS data and information management, and the coordination with the operational maintenance base for actions to maintain operational status of SPS.

- Operational maintenance base management covers the establishment of priorities and maintenance schedules for both SPS and base maintenance, the operation and maintenance of communication with the ground, data and information management, and the coordination with the SPS logistics management for required materials and supplies.

TABLE 10-1. PROGRAM REQUIREMENTS AND MANAGEMENT CRITERIA

Program Requirements	Management Criteria
1. SPS operational program elements require numerous diverse and concurrent activities.	Multidisciplinary action ties
2. Ground location of management, administrative, and technical personnel; GEO located maintenance personnel; ground-based and space-based logistics personnel.	Diverse location of personnel
3. Diverse, complex, and interrelated activities require single authority.	Unity of command
4. Complexity and multitude of systems and operations require capability of shifting resources to unscheduled events and activities.	Flexibility

- SPS logistics management covers the tasks of SPS maintenance, crew rotation, and the provision of crew logistics and material supplies for maintenance.

- Ground receiving and distribution management includes the monitoring and control of the receiving antenna and the utility interface system, their maintenance and logistics, and the systems safety provisions.

The overall approach in power management is shown in Figure 10-2. Each SPS has its individual level of management that feeds all vital information to the central power management from where, in turn, required actions will originate.

10.3 CONTROL MODEL

The objective of the control model is to satisfy the energy demand at the ground receiving and distribution sites according to established SPS performance criteria.

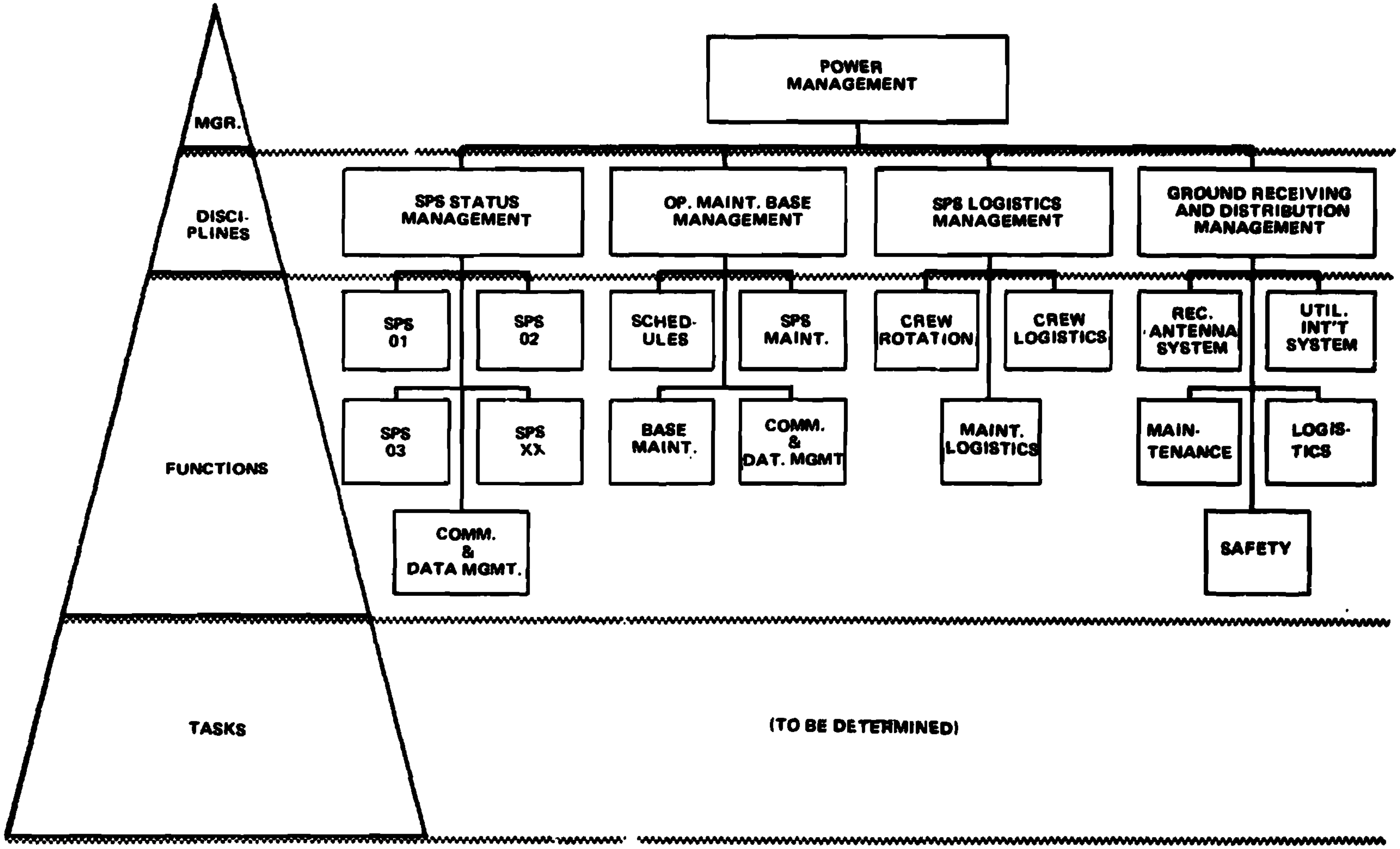


Figure 10-1. SPS power management structure.

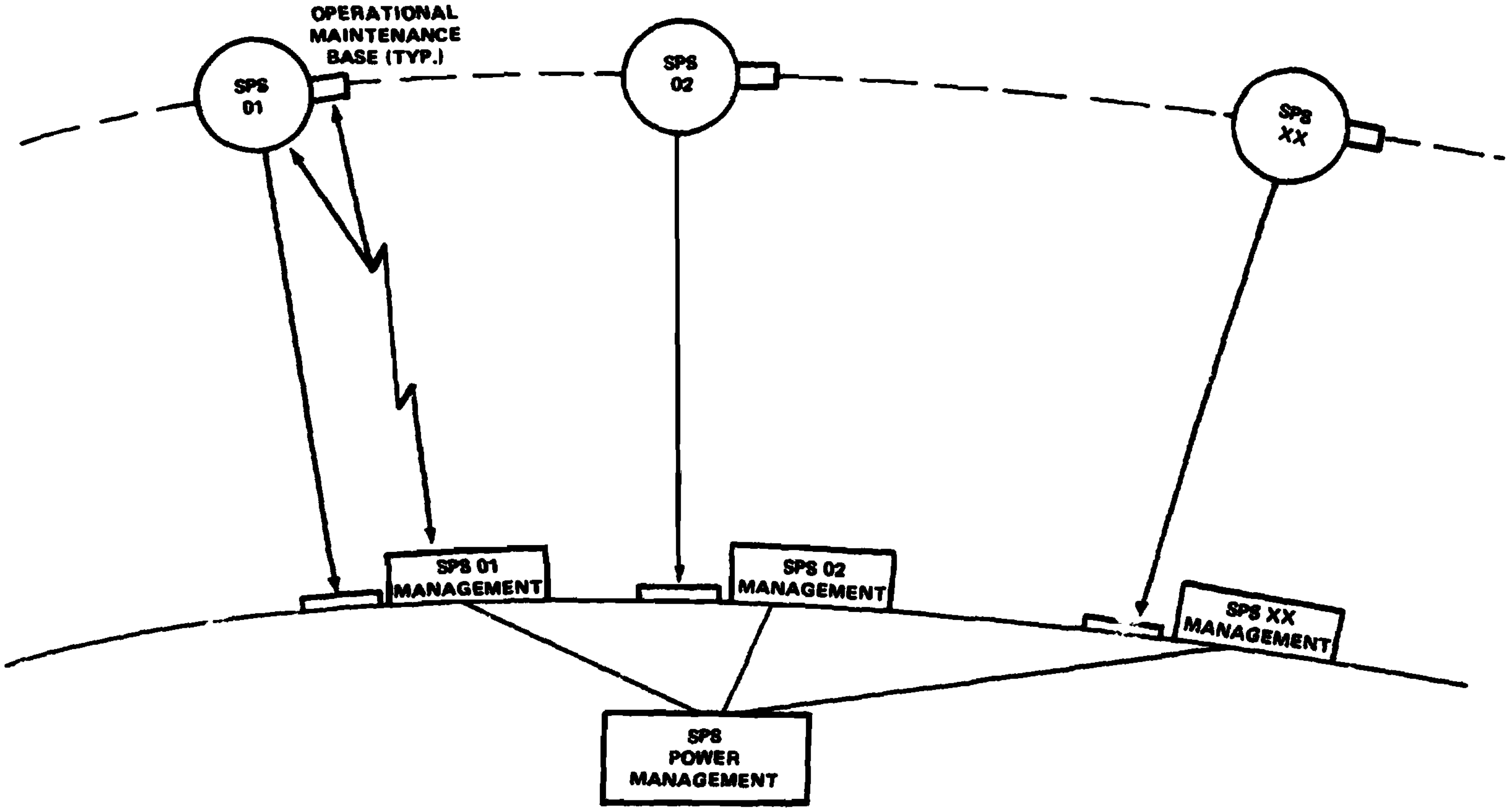


Figure 10-2. Approach to power management.

To that end, a control model will include the following elements:

- SPS Orbit Control (Ground-Based)
    - Coordinate Systems and Time References
    - SPS Tracking
    - Data Processing
    - Computation
    - Optimum Correction Maneuvers
    - Commands and Communications
- SPS Performance Control
    - Energy Collection Control
        - SPS Originated Variations
        - Maintenance Shutdown Variations
        - Solar Annual Variations
        - Solar Cycle Variations
    - Energy Reception Control
        - Meteorologically Caused Variations
        - Earth Precession and Nutation Caused Variations
        - Maintenance Shutdown Variations
    - Performance Compensation Control
        - Grid Power Shift
        - Satellite Power Shift

- SPS Malfunction Control
    - Diagnostics
    - Communications
- SPS Logistics Control
    - Spares Planning
    - Transportation Planning
    - Schedule Planning
    - Communications

The systematic development of the control elements and their interrelations will be the subject of future studies.

11.0 SUMMARY BASELINE DEFINITION

As the satellite power systems are studied, more knowledge will be obtained as to the interactions, limitations, and impacts of such systems. Along with this study activity, many baseline concepts will be developed to aid in the understanding of the SPS program. These baselines will naturally change and evolve as study iterations are made. The final SPS design will probably be significantly different than can currently be envisioned. When a concept is defined and studied, problem areas can be identified and techniques to improve the design will be noted. Since the baselines of this study were established, a great deal of analysis has been completed and several areas of potential improvement have been noted. The next iteration baselines will be forthcoming when additional analysis is completed.

11.1 PHOTOVOLTAIC

One photovoltaic concept that was used in this study was a 10 GW SPS of elliptical planform with two 1 km diameter end-mounted antennas. Further study of this configuration identified some unfavorable traits, such as the elliptical configuration is not best suited for end-mounted antennas both from a structural and power distribution standpoint. It was noted much earlier in the study activity that for an SPS with a center-mounted antenna, the elliptical configuration offered several advantages over the rectangular configuration. These included less structure and power distribution mass and less attitude control propellant. However, these advantages do not apply to the end-mounted antenna. It appears that the rectangle is better suited for the end-mounted antenna. For the center-mounted antenna the diamond configuration may be the most favorable for the power distribution and attitude control requirements. The impact on the structure and assembly has yet to be completed.

Most of the previous SPS study activity assumed that the SPS solar array would be maintained perpendicular to the Sun (Z-solar). After a trade study was completed which included both mass and cost analyses, it was concluded that from a cost standpoint there was only a slight difference in cost for Z-solar versus pointing perpendicular to the orbital plane (X-POP). The initial cost for the X-POP concept was slightly greater than that for Z-solar because the larger solar array is required to offset not always being pointed directly at the Sun. The total cost for the X-POP concept was slightly lower than Z-solar primarily because of the lower propellant requirements for the 30 year lifetime. The rotary joint requirements are also less severe for the X-POP orientation since a two-axis rather than a three-axis gimbal is required. Therefore, the next baseline which is established may be oriented X-POP rather than Z-solar.

Changes in the concentration schemes and concentration ratios have been studied. Preliminary analyses indicate that the overall mass of the SPS could be reduced significantly by using the reflectors as passive radiators for the trough concentration concept. This would require using blanket widths of approximately 1 cm to dissipate the heat rather than the current width of 200 m. There are still some areas where analysis is incomplete. The impact of adding secondary structure to support the small width troughs is currently under investigation. The effect on orbital assembly operations and the impact for orientation perpendicular to the orbital plane has yet to be completed. There are several other concentration schemes also under investigation, some of which are parabolic and four-sided reflector configurations. However, the impact of X-POP orientation on the concepts has not been completed.

One of the original baseline concepts studied included one 10 GW antenna. It was noted that this resulted in large microwave power density levels of 50 to 60 mW/cm^2 near the rectenna. The power density level at which microwave beam ionosphere interactions are expected to occur (approximately 20 mW/cm^2) was then used as an upper limit in the design of the microwave system. This limited the output power level at the power interface to approximately 5 GW for each antenna. Recent analyses have been conducted comparing the mass of two 5 GW photovoltaic SPS's to that of one 10 GW SPS if both are equipped with 5 GW antennas. This comparison assumed that the antenna and rectenna designs were identical for both power levels and each 5 GW SPS had one center-mounted antenna and the 10 GW SPS had two center-mounted antennas, each pointing to a different rectenna. The dry weight of the 10 GW system with two antennas was 7 percent heavier primarily because of the dc power distribution efficiency losses. In addition, the attitude control propellant requirements for the 10 GW SPS were approximately twice that of the two 5 GW SPS's. The result of this analysis, which indicated some merit for the 5 GW concept over the 10 GW concept, should not be interpreted as assuming that four 2.5 GW or three 3.3 GW SPS's would be better than two 5 GW SPS's. The additional antenna and rectenna cost would probably affect any additional savings in power distribution efficiency and attitude control propellant. However, there are plans to study various output power levels to determine the best design point. Also, the limitation of available space in synchronous orbit and its impact on the upper bounds of total program output for various SPS power levels is to be studied.

In some of the completed sensitivity analyses it was noted that the total SPS mass could be reduced by approximately 12 percent if the dc power distribution voltage was increased from 20 to 40 kV. The impact on the amplitron design and plasma interactions on the solar array are under study. Also, the previously described comparison of 5 to 10 GW SPS's might change if 40 kV rather than 20 kV was used for power distribution.

A new photovoltaic baseline will be established when the previously mentioned analyses are complete. Included will be the interaction of various elements of the SPS in order to minimize the overall mass and cost of the SPS program.

11.2 SOLAR THERMAL

Most of the study activity for the solar thermal concept assumed 544 individual absorber/turbomachine modules. Preliminary analyses and trade studies have been completed. They indicate that the total SPS mass could be reduced by approximately 20 percent if the number of modules was decreased to approximately 50. This was primarily due to the mass of the turbomachines. Other analyses indicate potential mass savings if the radiators were placed near the absorber rather than included on the back surface of the concentrator. Additional analysis is underway in other areas and, when completed, a new baseline will be generated.

11.3 NUCLEAR

The study activity for the nuclear concept has switched to a UF_6/Brayton concept because of limitations in the molten salt breeder reactor which required a low temperature radiator. This caused this type of satellite to be the most massive of any concept studied.

12.0 SPS SUPPORTING PROGRAMS

12.1 SPACE CONSTRUCTION BASE (SPACE STATION)

A space construction base will have the facilities to accommodate various manning levels; for erecting large structures; for the conduct of scientific experiments and for application operations such as materials processing, pharmaceuticals, and public service communications; and for basing and servicing transport vehicles and free-flying spacecraft. The early space construction base at LEO will have the facilities to provide limited capability in these areas for advanced technology verification, development, and demonstration. Limited production and operational capabilities may be provided in the area of materials processing and communications. The facility will provide for permanent occupancy by man in Earth orbit, for continuous development and use of space for industrial purposes, and for advanced development of space science and applications. The facility will provide a space laboratory environment for final systems development, assembly, modification, and verification beyond the capabilities of Earth-based facilities. The early construction base will be shuttle compatible and will provide for modular buildup. Larger diameter module elements may be required to support a larger capability construction base in the 1990's and beyond.

12.1.1 SCENARIOS AND CONCEPTS

Various scenarios and basic concepts have been developed in response to the requirements for an early construction base at various levels of involvement. Emphasis has been placed on requirements during the 1983 through 1987 time period. In-house activities included tasks for development of basic space station module configurations including habitability and subsystems, mission support, and logistics elements. Preliminary concepts and configurations are being developed for a number of dedicated elements that may have production and/or commercial capability, such as material processing.

Beam manufacturing and assembly modules, material storage modules, and construction platform concepts and preliminary configurations were developed in support of SPS verification and demonstration programs. Figures 12-1 through 12-4 are representative of an early space station construction base program and a basic module element. Preliminary analysis of a modular station buildup to accommodate up to 20 men could support construction of a subscale SPS and a public service platform program.

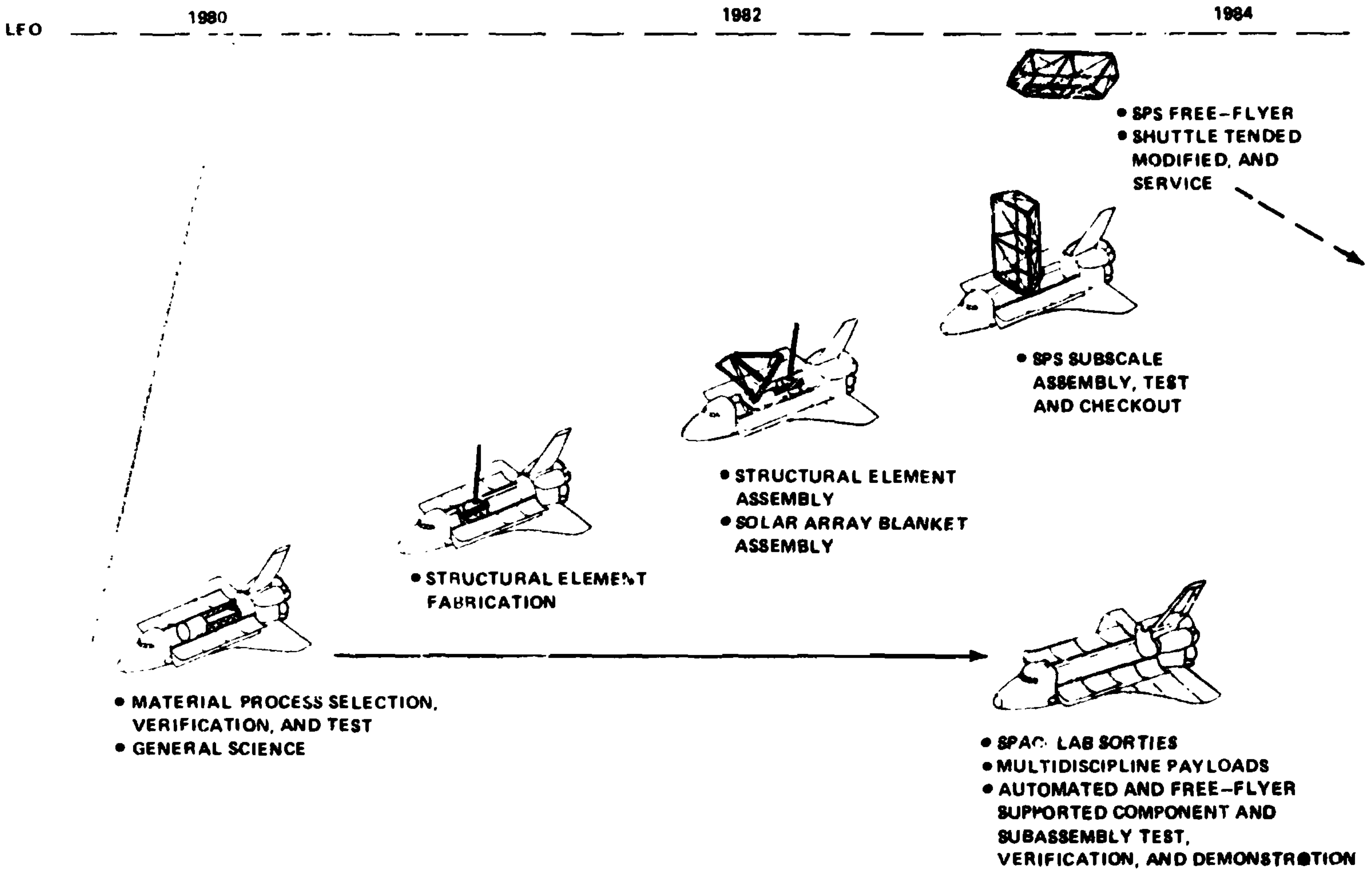


Figure 12-1. Typical pre-space station scenario, 1980 through 1984.

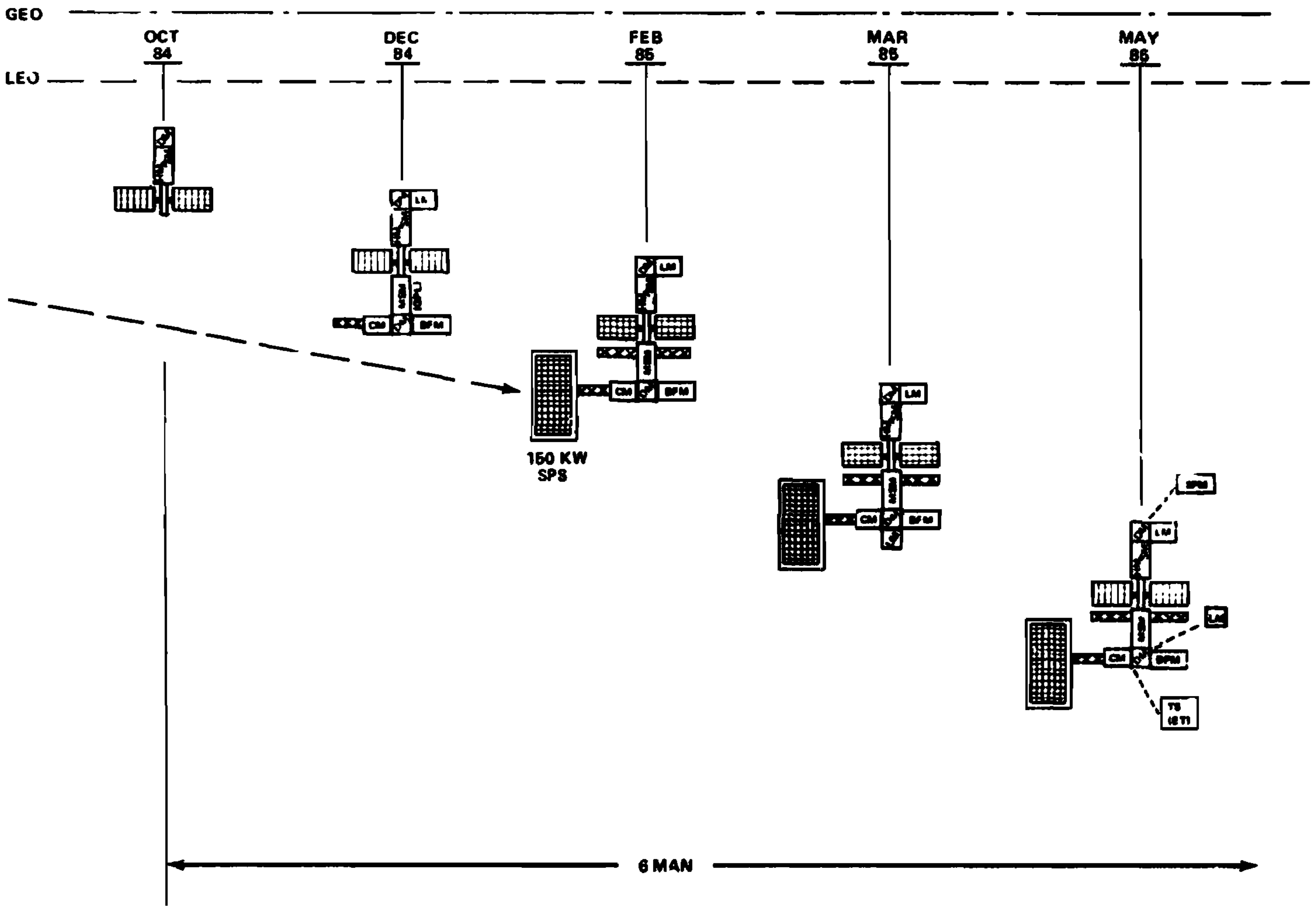


Figure 12-2. Typical space station program option.

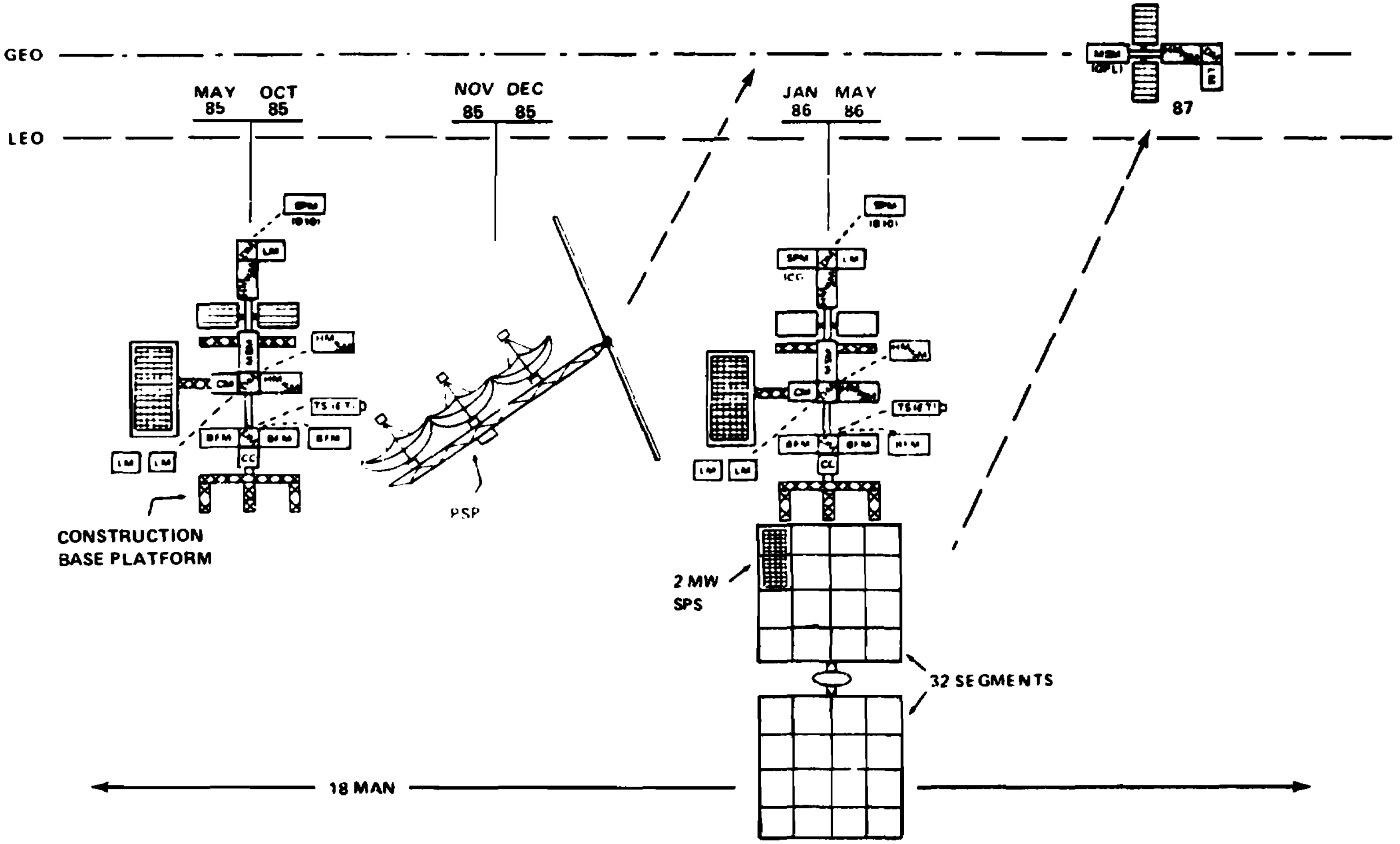


Figure 12-3. Typical space station program option.

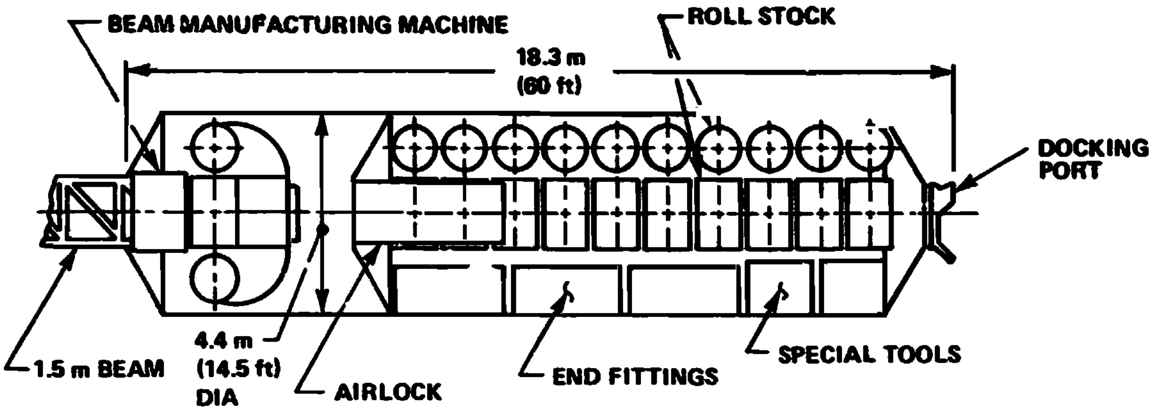


Figure 12-4. Space station beam manufacturing.

12.1.2 SUPPORT SERVICES

The space station construction base provides the basic support for multidiscipline science and applications development and operations. The support includes accommodations for crew and subsystems services such as environmental control and life support, power, communications data processing and instrumentation, attitude control system, and stabilization. Basic concepts and modular configurations were developed to accommodate a four to six man space station buildup that provides these services. These services coupled with a fabrication and assembly capability and an earlier developed power satellite provide a base for construction of a subscale SPS. Continued modular buildup and minor modifications to subsystems will significantly increase the construction base capability to accommodate dedicated elements for large scale commercial test, manufacture, and operations.

12.2 TRANSPORTATION

12.2.1 REQUIREMENTS AND ANALYSIS

The transportation requirements to support the SPS during the buildup in LEO and transfer to GEO, and the SPS operational time frame dictate a number of special purpose transportation systems. Earth-surface-to-LEO transportation requirements dictate two basic systems: (1) heavy lift launch vehicles for SPS cargo, and (2) personnel launch vehicles for construction/maintenance base(s) crew rotation and critical logistics support. Orbital transportation basically falls into the same two categories; however, there are a number of issues yet to be resolved that exaggerate the number of options for propulsion systems that are candidates for orbital transportation systems.

The basic issue, from an orbital transportation viewpoint, is SPS assembly location (LEO versus GEO). Propulsive options that exist for SPS transportation when assembly is in LEO are low thrust electric propulsion devices that can capitalize on the electric power generation capability of an assembled SPS or SPS module. Electric propulsion is also desirable for the transfer of SPS's or assemblies from LEO to GEO, since the lightweight structure for SPS requires very low acceleration ($< 10^{-3}$ Earth gravity). When electric propulsion is employed as the cargo orbital transfer system, personnel transportation between LEO and GEO is accomplished via a high thrust system with trip times to GEO of hours rather than the months dictated by low thrust options. If geosynchronous orbit assembly is assumed, the nature of the orbital transfer system changes because of the lack of a "free" power source provided by the payload. GEO assembly does not eliminate electric propulsion as a viable option; however, the need for an independent power source and the trip times associated with low thrust propulsion make high thrust propulsion a viable candidate. Further, when high thrust propulsion is employed for cargo orbital transportation, the need for personnel and critical logistics transportation, as well as cargo, can be met with the same system.

The SPS assembly orbit location selection is dependent on a large number of factors other than the transportation issue and will only be resolved with significant attention over the next several years. Since this is the case, this section of the report will discuss transportation systems that meet both requirements.

12.2.2 HEAVY LIFT LAUNCH VEHICLE SYSTEM

The HLLV concepts described here are sized to deliver payloads to a 500 km circular orbit at 28.5° inclination. Two candidate HLLV's are described. Configuration details are given for a two-stage ballistic vehicle, followed by a description of a ballistic single stage-to-orbit (SSTO) vehicle. Cost comparisons for each vehicle are also provided.

The two-stage configuration, shown in Figure 12-5, has two fully recoverable ballistic stages. The upper stage uses seven space shuttle main engines (SSME). The lower stage has nine new LOX/RP-1 high chamber pressure engines. A 8 896 442 N thrust engine was selected to be compatible with current test stand capability.

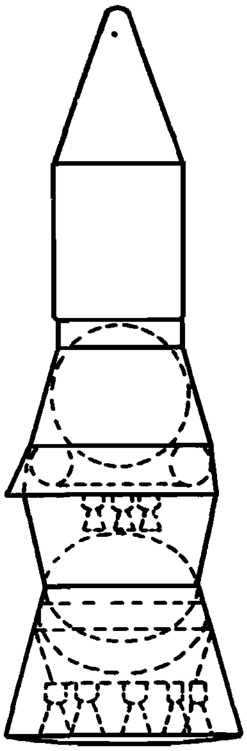


Figure 12-5. Two-stage ballistic HLLV.

Payload capability is approximately 270 000 kg. The payload shroud shown is sized for a density of 50 kg/m^3. The payload is cantilevered from the forward face of the second stage.

The first stage burns for 148 s. After separation, doors are closed over engine openings in the base heat shield and the stage is rotated heat shield forward for reentry. The stage is decelerated using retromotors for landing. It is towed 313 km back to the launch area, refurbished, and prepared for launch.

The second stage terminates at perigee of a 92.6 × 500 km orbit. At apogee circularization delta V is provided by orbit maneuvering system (OMS) motors. Following payload deployment, the second stage deorbits, reenters, retros, and lands on a prepared landing site near the launch site where it is refurbished and prepared for reuse.

The OMS function is performed by RL-10 engines. OMS LOX is stored in independent spherical containers. OMS LH_2 is in a separate container within the main hydrogen tank.

An asymmetric aft skirt panel provides directional control during entry. Control is required to overcome flight dispersions so that the stage can return to the vicinity of the launch site. During entry the stage is rotated slowly on its axis. If lift is required, rotation is stopped and the aft skirt is located away from the desired direction. A 0.25 lift to drag ratio can be developed.

During entry as the stage reaches approximately 25 000 m and Mach 2, it is rotated and a ballute is deployed. The ballute stabilizes and slows the stage. In the subsonic region three parachutes are deployed. At approximately 150 m altitude, two SSME's are ignited for final deceleration. The engines, which can be throttled, settle the stage to a soft landing on its landing legs.

The booster is a sea landing ballistic entry stage. The most forward component is an interstage forward skirt. The space forward of the LOX tank houses the upper stage engines. The LOX and RP-1 tanks share a common bulkhead. The aft shell structure is primarily an aerodynamic fairing. The thrust structure distributes thrust loads into the external shell and provides the load path from the external shell to the hold-down posts. The hold-down posts support the stage on the ground and are the terminus for the heat shield radial beam structure.

The base heat shield consists of two skins separated by 12 cm high stiffeners. Each inner skin is an assembly of two sheets rolled together. The inner of these has expanded ducts for water flow. Formed spray holes are drilled in the skin. When needed during boost and entry, water is sprayed onto the exterior skin. The steam is vented overboard through a metering orifice.

The ballistic single stage to orbit vehicle, shown in Figure 12-6, is a dual fuel fully recoverable launch vehicle with a 230 000 kg payload. At liftoff its 48 engines provide a thrust to weight ratio of 1.3. Twenty-four engines utilize LOX/RP-1 propellants; the other 24 are LOX/LH_2 engines. The engines supply thrust vector control by differential throttling. Both engines represent a new engine development. The engines do not have protective doors, instead a steam system provides active engine cooling during entry.

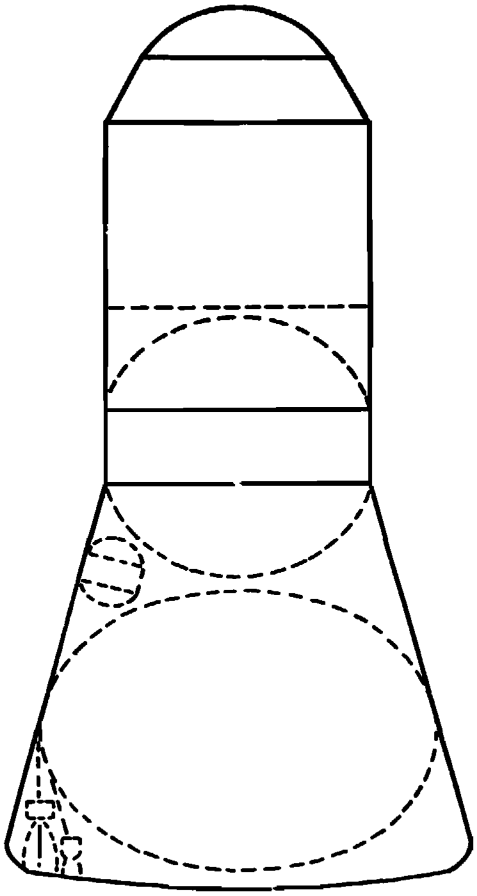


Figure 12-6. Ballistic single stage concept.

The stage structure in the base region is similar to that described for the ballistic first stage. The base heat shield is water cooled and is supported on a space frame structure. The materials used are also similar to the first-stage ballistic case. An exception is the forward skirt and payload bay which use a René 41 honeycomb skin to enable the fully recoverable payload bay to withstand reentry heating.

After 127.4 s of flight time the LOX/RP-1 auxiliary engines are shut down. The 24 LOX/LH_2 engines continue to accelerate the vehicle until a 92.6 × 500 km orbit is reached. The vehicle then deploys the payload and a kick stage. The kick stage circularizes the payload orbit at 500 km. The SSTO is deorbited and reenters. As it approaches the landing site, 16 LOX/RP-1 engines are ignited to give a thrust to weight ratio near 3. The SSTO is decelerated to near zero velocity. All but four of the engines are shut down, and the stage settles to a water landing in a prepared landing basin. The landing site is a 5 km lake adjacent to the launch site. A dredged basin is used in preference to open sea to reduce the effect of sea state.

In one concept the 900 000 kg empty SSTO is towed out of the landing basin through a canal to the launch area. At the pad area it is positioned over the launcher and anchored. A lock gate is closed, and the water is pumped out. The SSTO is then ready for on-pad servicing and launch. In a second concept, the launch pads are located around the periphery of the basin.

Cost estimates were made for development, production, facilities, and operations of both the two-stage and SSTO vehicles. For a launch rate of 500 per year, the development and facilities costs total approximately $11 billion for each concept. Production costs to support a 15 year program are greater for the SSTO.

Figure 12-7 gives cost per flight for each of the configurations over a range of activity levels. For activity requiring 500 launches per year, operations cost is broken into three elements. The shaded areas, which represent expendable hardware, are a significant portion of the operations costs. The shroud costs can vary over the range given by the angled line as payload density varies over a possible range required to support SPS. The significant cost effect of expending even relatively inexpensive hardware suggests continuing work to find ways to recover shrouds and the kick stage, or develop other uses in space for them.

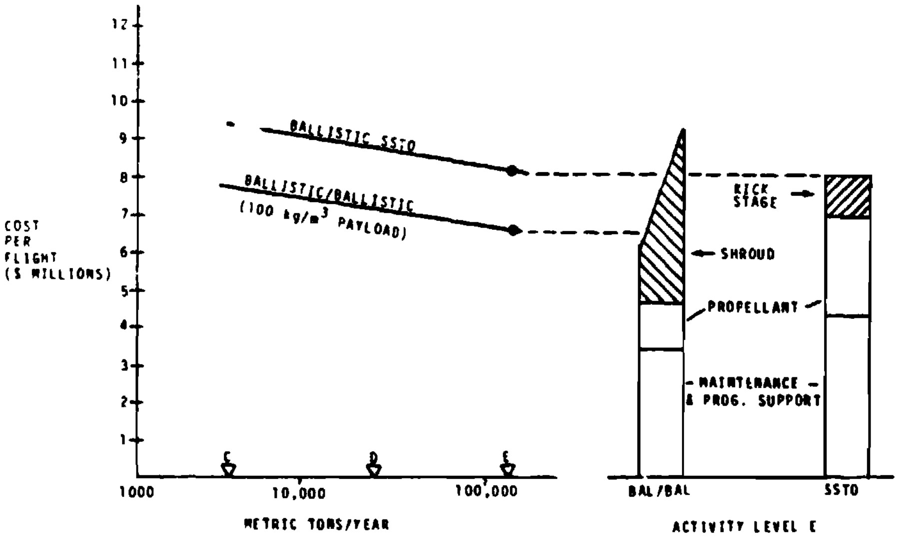


Figure 12-7. HLLV cost per flight comparison.

Propellant costs for the SSTO are twice the two-stage value because of the required larger flow and the greater amount of LOX/LH_2 used.

Maintenance and program support costs are major cost elements for both vehicles. Stage maintenance cost is slightly higher for the SSTO than for the two-stage vehicle, because maintenance cost is a function of stage cost which is weight dependent. The SSTO is heavier; therefore, its stage maintenance cost is higher. SSTO engine maintenance cost is much higher than the two-stage engine cost, because the total installed thrust is higher and the number of engines is greater. Maintenance and overhaul ground rules were patterned after airframe and engine overhaul specifications for shuttle, with some modifications to differentiate between water and land landing stages.

Ground operations and program support costs have shuttle as their basis. Modifications were made to allow for the absence of crew and payload related functions. Operations and support costs for HLLV use approximately the same number of people located at Kennedy Space Center per flight as shuttle, but the man-hours were adjusted linearly with the HLLV turnaround time compared to shuttle.

Major technology requirements are those associated with minimizing cost and improving performance. Cost is minimized by eliminating expendable hardware such as kick stages, shrouds, or ablative heat shields. Payload improvements may be made by eliminating an actively cooled heat shield in favor of a passive system that will withstand the high heating rates and yet be salt water compatible. Each of the vehicles requires the development of new engines. A large LOX/LH_2 engine and new LOX/hydrocarbon engines are required for the SSTO. A new high chamber pressure LOX/hydrocarbon engine is used for the two-stage booster.

Landing modes for both land surface and water need development. These should include implications of surface atmospheric environment. The influence of salt water on the hot structure or engines must be determined. These investigations should include both parent metal effects and those of coatings and finishes.

The evaluation of the two concepts provides the following observations:

1. The development schedule and cost for each are about equal.

2. Operationally, the SSTO has shorter recovery and processing time and requires fewer launch pads.

3. Maintenance costs are lower for the two-stage vehicle, but its expendable shroud is a cost disadvantage.

4. Cost per flight for the two-stage vehicle is slightly lower if dense payload shrouds are used.

5. Weight growth performance sensitivity is much lower for the two-stage vehicle.

12.2.3 PERSONNEL LAUNCH VEHICLE SYSTEMS

The personnel launch vehicle system is assumed to be the shuttle. A variety of booster/external tank options have been evaluated in the MSFC Shuttle Growth Study under contract to Rockwell International Corporation and in MSFC in-house transportation systems studies. The baseline shuttle (Fig. 12-8) uses a solid rocket booster and an expendable LOX/LH_2 external tank to supply propellants to the orbiter's main engine (SSME). Alternate configurations are shown in Figures 12-9 and 12-10 that use reusable liquid boosters and reusable boosters/external tank, respectively. The range of cost per flight achievable with these systems is from approximately $8 million (reusable boosters and external tank) to $15 million (baseline).

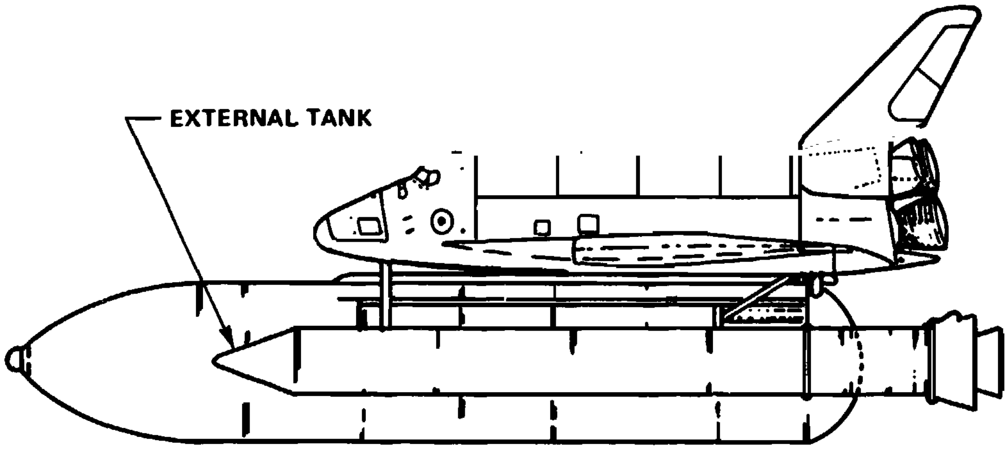


Figure 12-8. Baseline shuttle configuration.

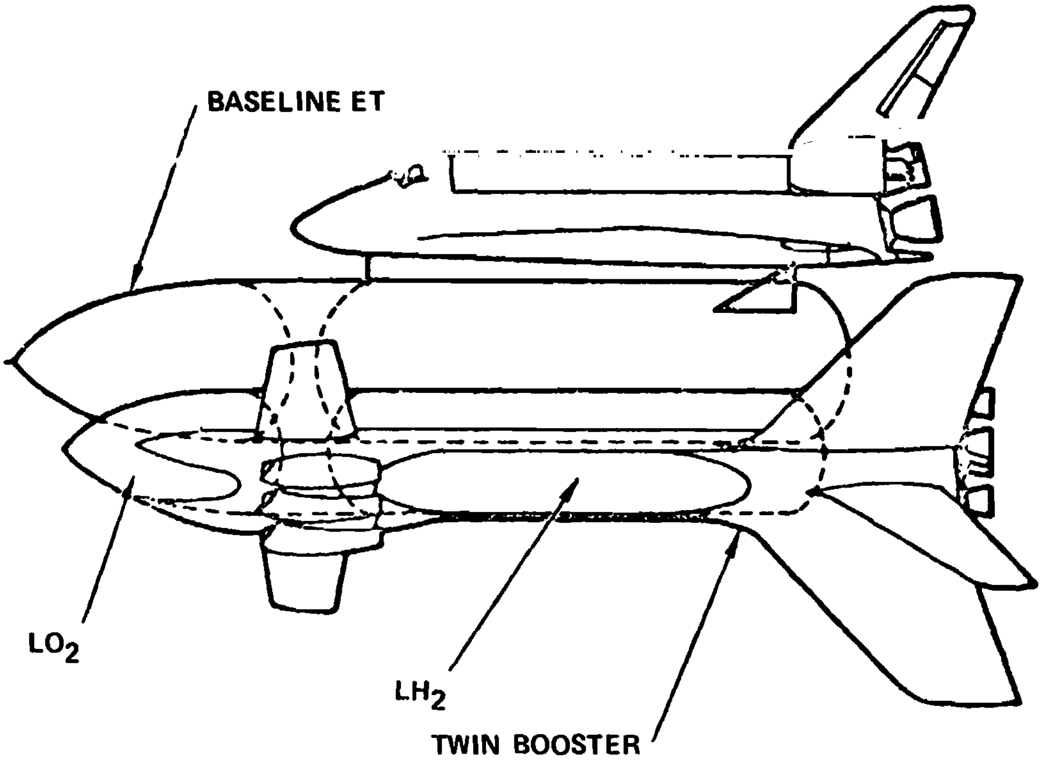


Figure 12-9. Shuttle with liquid booster.

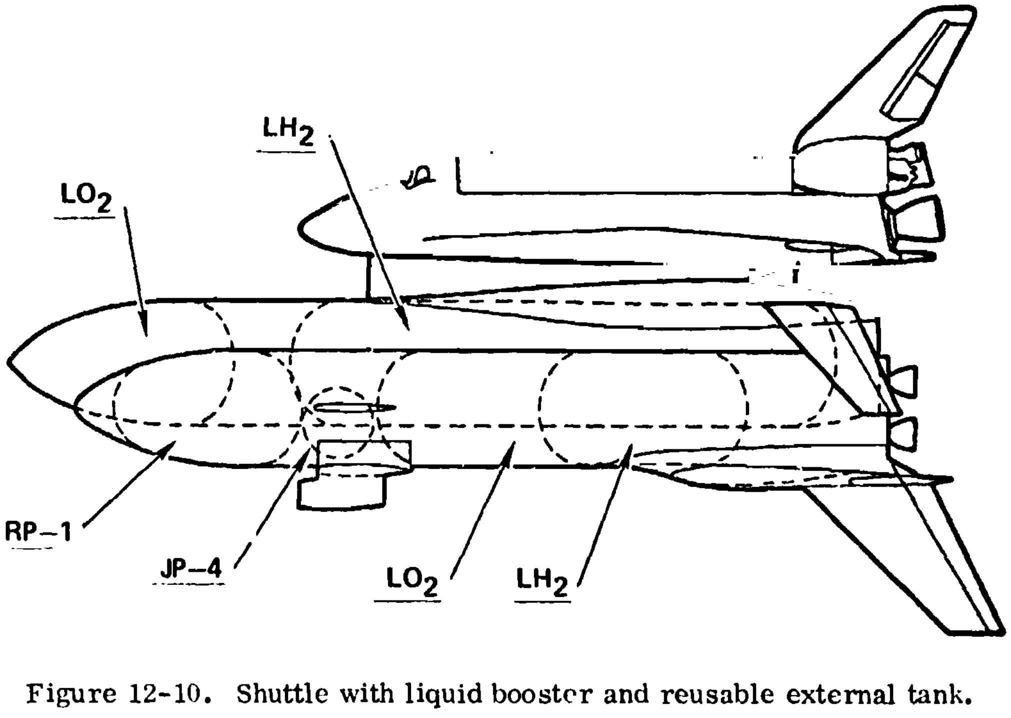


Figure 12-10. Shuttle with liquid booster and reusable external tank.

The propulsion options that are currently being evaluated for a liquid booster for shuttle are shown in Figure 12-11. Prime candidates for booster engines are the space shuttle main engine with a shortened skirt (expansion ratio of 35:1 optimized for booster application) and a modified SSME (dubbed SSBE for space shuttle booster engine) that can use LOX/RP-1 propellants with hydrogen being used for engine cooling. The use of the F-1 and the HiPc (O_2/RP-1) for this application has been deemphasized because of the weight, performance, and thrust levels (or cost to provide deep throttling capability) on the F-1 and development costs for the HiPc (O_2/RP-1) engine (estimated to be approximately $800 million).

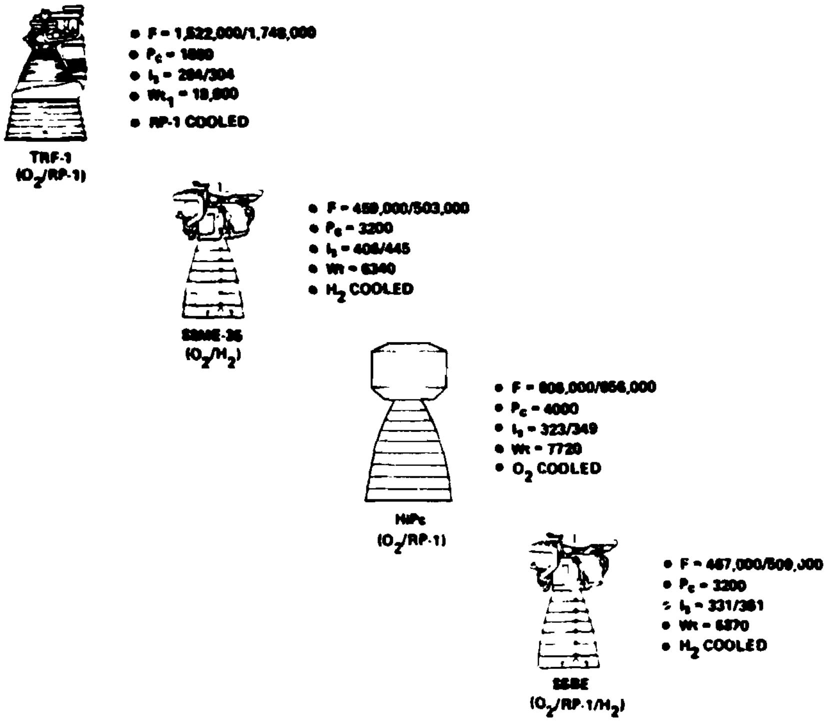


Figure 12-11. Liquid rocket booster engine candidates.

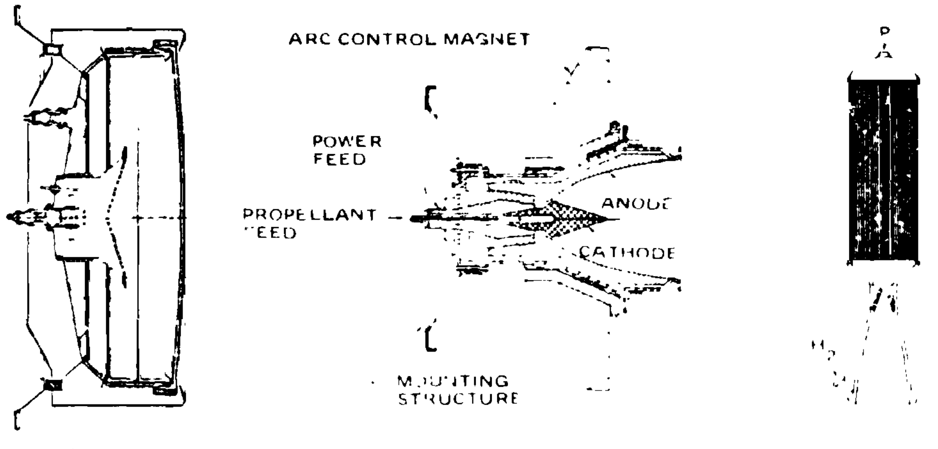
The shuttle capabilities for delivering a payload to the assembly orbit of 270 km at an inclination of 28.5° (due east launch from Kennedy Space Center) are from 25 500 kg for the baseline shuttle with one OMS kit to 29 500 kg with a liquid booster replacement for the solid rocket booster and integral OMS. The OMS kit is added to the orbiter and is internal to the payload bay (requires 2.75 m of payload bay length). The use of the orbiter within the SPS program is primarily for personnel transfer to and from LEO and possibly for critical logistics resupplies. The number of passengers that can be launched in the shuttle is more a function of cargo bay length available than weight lifting capability. The baseline shuttle (requiring one OMS kit) has 15.6 m of cargo bay length available, and it is estimated that a personnel module would contain 55 people. If the OMS kit was not required, approximately 70 people per flight could be accommodated in the shuttle cargo bay.

12.2.4 CARGO ORBITAL TRANSFER VEHICLE SYSTEM

As described in subsection 12.2.1, the orbital cargo transfer system options can be divided into two basic categories: (1) LEO assembly/payload powered systems, and (2) GEO assembly/independently powered systems. A discussion of each of these systems follows.

12.2.4.1 LEO ASSEMBLY/PAYLOAD POWERED SYSTEMS

Since electric power generation is inherent to the SPS or SPS assemblies, electric propulsion is a natural choice for at least initial consideration. There exists a large variety of electric propulsion types, but for the purpose of examining the range of capabilities available, three thruster types are evaluated: (1) ion, (2) magnetoplasmadynamic, and (3) resistojet. Characteristics of these thrusters are shown in Figure 12-12.



THRUST: 3.77 N	THRUST: 9.81 N	THRUST: 1670 N
WEIGHT: 100 kg	WEIGHT: 40.0 kg	WEIGHT: 4200 kg
I_{SP}: 18 000 s	I_{SP}: 10 000 s	I_{SP}: 814 s
POWER: 398 kW	POWER: 587 kW	POWER: 7.5 MW
(1) ION	(2) MPD	(3) RESISTOJET

Figure 12-12. Candidate electric thrusters.

The ion thruster is a demonstrated technology for low power systems (kilowatts), and extrapolations can easily be made from this base (30 cm thruster) to a more appropriate ion thruster (100 cm) for SPS cargo orbital transfer. The ion system, in addition to being state-of-art, has the advantages of high efficiency, long life, and wide operating range. Power processing for the ion thruster, however, makes this system expensive and heavy per kilowatt input to the thruster. Future analyses can profitably be applied to methods of returning the ion thruster assemblies back to LEO to lower operating costs through reuse of these long life and relatively expensive components. Argon is used as the propellant for this study.

Magnetoplasmadynamic thruster concepts are relatively speculative. A significant amount of additional technology work is needed to specify configurations, performance characteristics, and costs. Conceptually, however, the MPD is a relatively simple device with low power processing requirements and, consequently, low hardware costs. An additional advantage of the MPD thruster is that the thrust density (power density) is significantly greater than that of the ion engine; therefore, fewer thrusters are required to perform the SPS orbital transfer to GEO. Argon is the propellant assumed here.

Resistojets are basically the electrical resistance parallel of the nuclear (solid core) engine, NERVA. The performance characteristics of such an engine are similar, i.e., specific impulse of approximately 800 s. Resistojets are essentially state-of-art, but are at significantly lower power levels than will be required for the SPS application. In addition to the relatively low specific impulse (in comparison the ion and MPD, specific impulse is in the 2 000 to 20 000 s range), the propellant is liquid hydrogen and long term (several months) storage of LH_2 is substantially more difficult than storing argon.

From the candidate electric thruster discussed and a relative transportation performance/cost comparison shown in Figure 12-13, the MPD thruster warrants increased technology attention. The relative performance/cost comparisons are based upon total costs from Earth surface to GEO. The results of the HLLV study indicate \$20 per pound is easily achieved with the fully reusable system; therefore, this is used for the Earth launch cost. The performance of the ion system is indicated by the launch cost contribution, and the ion system cost (expendable) pushes it to the highest value of the three options evaluated. The MPD and resistojet have relatively low cost hardware and, even though the launch costs predominate (especially for the resistojet), the totals are less.

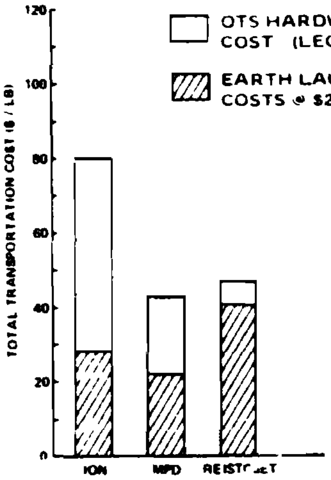


Figure 12-13. Electrically powered concepts comparison.

12.2.4.2 GEO ASSEMBLY/INDEPENDENTLY POWERED SYSTEMS

The nature of the orbit transfer function changes when assembly of the large lightweight SPS is in GEO. The acceleration limit is orders of magnitude greater. Consideration for orbit transfer propulsion in this option is given to both high thrust (chemical and nuclear) and low thrust (nuclear electric with MPD thrusters). Chemical (LOX/LH_2) stages are state-of-art systems and low cost per unit hardware. Since the specific impulse of the LOX/LH_2 systems is relatively low, staging either of tanks or of total stages (reusable) is mandatory. The solid core nuclear stage is an option that offers almost double the specific impulse (825 s) of the LOX/LH_2 system, but the mass of the nuclear (solid core reactor) engine degrades most of the specific impulse gain. The solid core nuclear engine is a mature technology. A more speculative nuclear engine that offers substantially greater specific impulse than either LOX/LH_2 or the solid core nuclear system uses a gaseous core reactor. The specific impulse of this system is estimated to range from 1800 to more than 5000 s, and thrust levels on the order of 400 000 N are achievable. These high thrust options have been evaluated and orbit transfer performance/cost comparisons are shown in Figure 12-14 (including the nuclear electric system with MPD thrusters).

While the chemical system (the range covers staging options and expendable systems) and solid core nuclear system are existing or mature technologies, the gaseous core nuclear system and the nuclear electric system offer substantial performance/cost margins. Once again, trip times from LEO

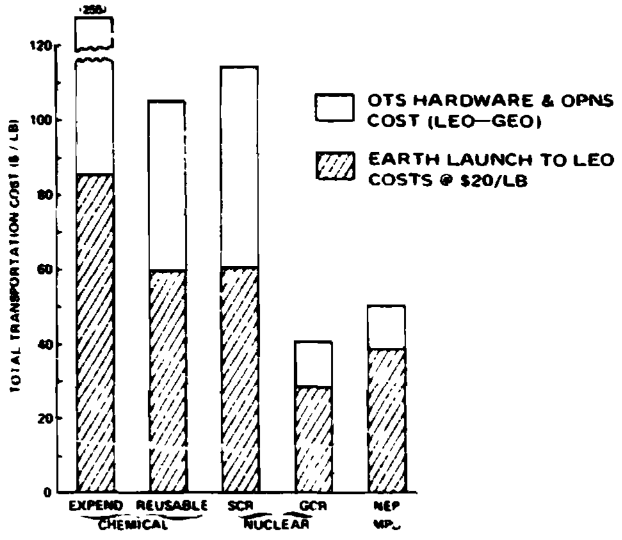


Figure 12-14. Independently powered orbit transfer systems comparison.

to GEO are a factor. The trip times for the nuclear electric systems are on the order of several months, whereas the high thrust gaseous core nuclear system trip time is less than 1 day. Economic trades will be required to make a more definitive conclusion on this issue.

As discussed in subsection 12.2.1, the high thrust orbit transfer systems can potentially perform the dual function of cargo and personnel transfers from LEO to GEO. The use of nuclear systems for personnel transfer will require more extensive studies of the radiation environment where crew modules would be located.

12.2.5 PERSONNEL ORBITAL TRANSFER VEHICLE SYSTEMS

Personnel transfer vehicle options included in this discussion are sized for personnel and critical logistics resupply transfers between LEO and GEO. The only propulsive option evaluated is LOX/LH_2. All propulsive (all maneuvers of the transfer mission) and aeromaneuvering (aerodynamic braking on the return from GEO) vehicles are described. Figure 12-15 shows a typical configuration for an all propulsive orbit transfer vehicle. From a cost and orbital debris point of view, single and two-stage fully reusable transfer vehicles are preferred over the stage and one-half and two and one-half stage approaches.

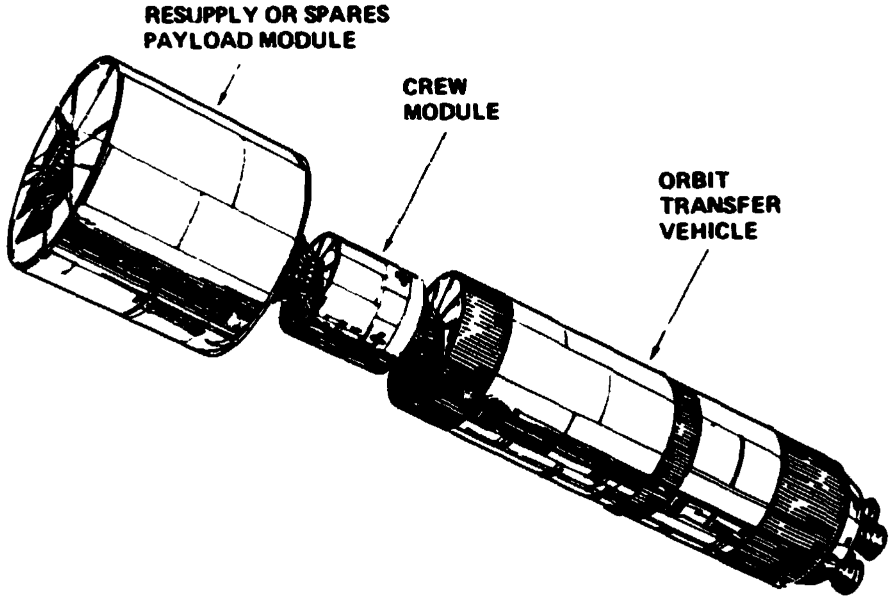


Figure 12-15. All-propulsive personnel and critical logistics orbital transfer vehicle.

Aerodynamic braking/maneuvering to reduce the mission propulsive velocity requirements have shown a significant benefit in propellant mass requirements for a personnel/logistics mission. The aerodynamic braking maneuver reduces the total mission propulsive velocity requirements by approximately 2.3 km/s. The resulting performance gain is most significant on a round trip or retrieval mission, the mission type required for personnel transfers, exchanges, and return from GEO to LEO. A typical aeromaneuvering orbit transfer vehicle is shown in Figure 12-16, and a performance comparison between all propulsive and aeromaneuvering systems is shown in the Table 12-1. The performance comparisons are based on shuttle compatible systems.

12.2.6 LOCAL SPACE TRANSPORTATION SYSTEMS

A variety of types of local transportation equipment will be required to accommodate such activities as cargo receiving and handling, solar array and conduction mast material delivery, structural beam transportation, antenna subarray delivery and retrieval, and damage surveillance. Cargo receiving and handling will probably involve such transportation equipment as elevators, trolleys, long manipulator arms, and possibly conveyor belts. Array and mast material delivery and beam delivery will probably involve some type of automatic free-flyers, since the distances involved may frequently exceed 5 km,

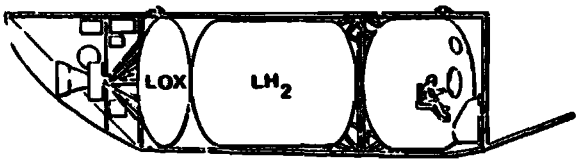


Figure 12-16. Typical aeromaneuvering orbital transfer vehicle concept.

TABLE 12-1. PERFORMANCE COMPARISON OF AEROMANEUVERING VERSUS ALL PROPULSIVE ORBIT TRANSFER VEHICLES

	Performance (kg)	
	Delivery	Round Trip
One Stage Aeromaneuvering	5 200	2 860
One Stage All Propulsive	3 600	1 100
Two Stage Aeromaneuvering	15 000	8 200
Two Stage All Propulsive	11 800	3 600

unless a rapid trolley can be designed to operate on foil tracks. Some of this problem can be eliminated if an HLLV cargo depot and a beam fabrication facility is located on the jig, but this may not be compatible with rotations required by the flip-flop type of jig. For transportation of antenna materials from the depot to and across the rotary joint and to the antenna fabrication facilities, an elevator or cable car concept would probably suffice. Movement of subarrays across the antenna back surface should use the same type of trolley that will be routinely utilized to replace failed elements in the operational phase. Local transportation of personnel will be required for monitoring, damage surveillance, special extravehicular activity construction, alignment, or maintenance.

13.0 ENVIRONMENTAL EFFECTS ASSESSMENT

There are profound mutual effects envisioned between SPS elements and the respective environment in which interactions would take place. The analysis, testing, and definition of resulting impacts on the SPS systems design are a major topic of present and future efforts.

A summary review of SPS environmental interactions covers the range from Earth surface to geosynchronous orbit. For convenience this range can be divided into the terrestrial environment (Earth surface through stratophere) and the space environment (ionosphere, thermosphere, magnetosphere, and the geosynchronous distance)

The general approach toward assessing the multitude of interactions between SPS program elements and the various environments is through careful analytical modeling of environmental parameters and the introduction of the applicable program elements. The resulting interactions will be evaluated against acceptable tolerance limits and introduced into the overall SPS systems definition efforts as design guidelines and constraints. The definition of acceptability will have to rely on various political processes.

Certain environmental interactions that are of fundamental importance and that possibly create global consequences will have first priority. These are listed in Table 13-1. A summary of the major areas of potential SPS environmental interactions is given in Figure 13-1. A special discussion on the SPS collision probability with other satellites and possible effects on maintenance activities is given in subsection 7.1.7.1.

Tasks are being initiated in the following areas:

- Definition of the environment at geosynchronous orbit
- Definition of the physical processes that cause spacecraft charging and investigation of sheath perturbations around high altitude spacecraft
- Quasi-static modeling of the environment interacting with SPS including effects such as:
  - Differential charging
  - Variable geometry

TABLE 13-1. FIRST PRIORITY ENVIRONMENTAL EFFECTS

Physical Effects of Stratosphere Ozone Depletion
Expected Acoustic Launch Effects
Vehicle Failure Effects
Land Usage Effects
Ionospheric Microwave Interactions
Biological Effects of Microwave Beam
Communication, Meteorological, and Astronomy Effects of Microwave Beam
Biological Effects of Space Radiation
Magnetosphere Plasma SPS Interactions
SPS Collision Probability with Other Satellites

  - Secondary particle emission
  - Magnetic field/shadowing effects
  - Iterating models to include results learned from Scatha benchmark experiments

- Definition of in situ benchmark experiments for Scatha relative to SPS and data analysis of ATS-6
- Experimental modeling of aspects of SPS charging phenomena and data analysis equipment
- Quasi-empirical modeling of charging phenomena affecting large space structures (e.g., SPS), including:
  - Environmental effects that could uniquely affect large active space structures
  - Perturbations to the ambient environment created by large active space structures
  - Surface discharge characteristics based on available statistics of spacecraft malfunctions.

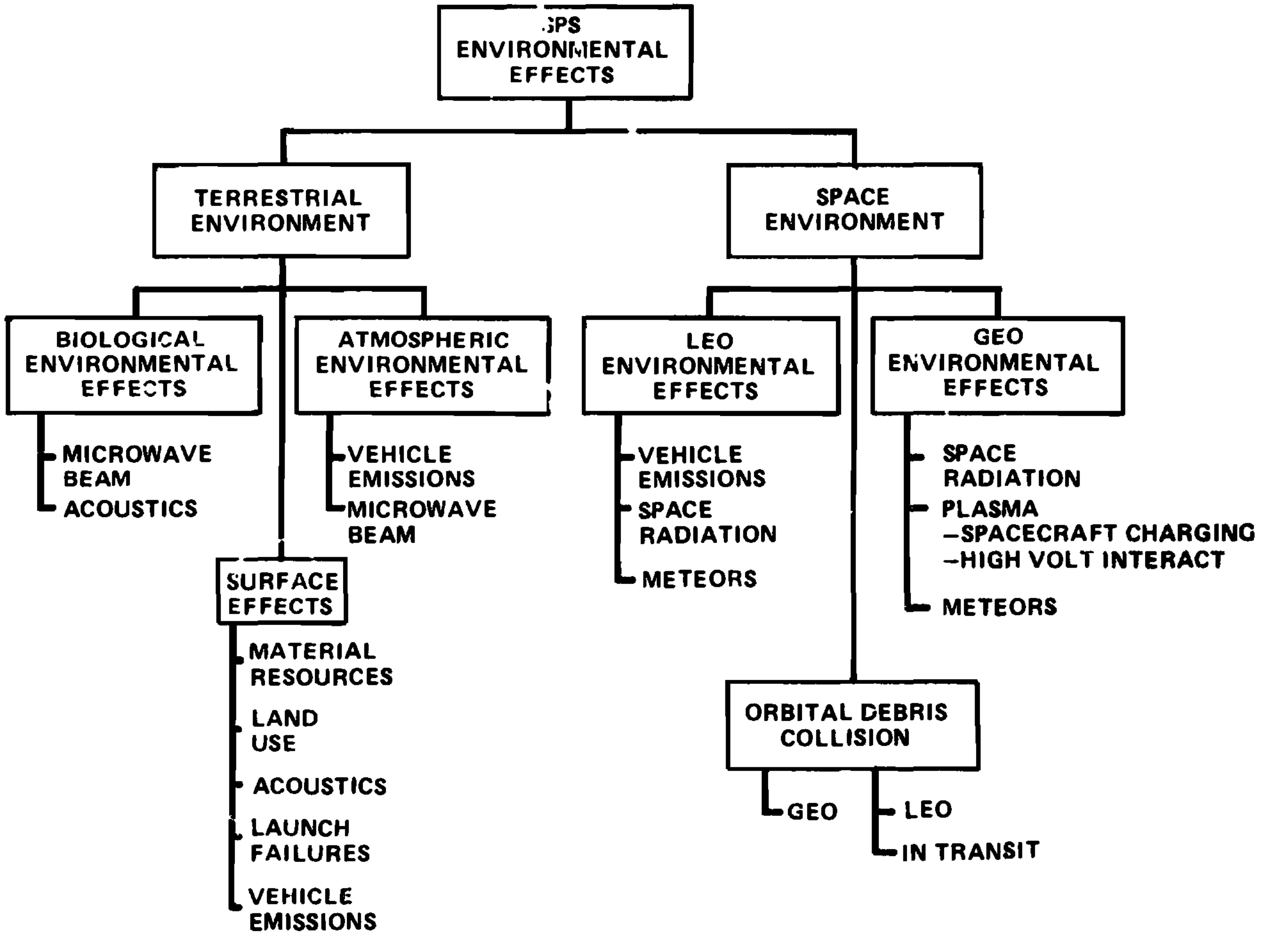


Figure 13-1. SPS environmental effects.

14.0 PROGRAM COST AND ECONOMICS

14.1 INTRODUCTION

A cost and economic analysis has been performed on a typical SPS concept using the MSFC SPS photovoltaic design and ground rules as a baseline. The results to date, when compared to similar studies of existing and new technology power plants for the 1990's, definitely place SPS in contention as a potential competitive candidate for economic reasons as well as other considerations. While the SPS is a complex, expensive, technologically advanced undertaking, all of the alternate methods for producing power have serious technical, environmental, social, or economic problems as well. The results of this particular study show that if cost and technical goals can be met in a few key areas, an SPS program could cleanly produce and safely deliver electrical energy from space to anywhere on the Earth's surface at economically competitive rates.

14.2 WORK BREAKDOWN STRUCTURE

In order to promote equitable, complete, and understandable comparisons of SPS concepts, a work breakdown structure (WBS) has been developed[2] that provides a standard format for the reporting of SPS costs. The WBS successively subdivides the SPS program into the lowest divisions and elements for which costs can practically be estimated, collected, and compared. Each division and element is defined by a WBS dictionary.

The WBS for the SPS program differs somewhat from the typical WBS for government aerospace programs in that it has been developed to accommodate the financial involvement of the private sector. Entries for taxes and insurance have been provided, and distinctions have been made between capital expenditures (which are recoverable by annual depreciation charges and are not deductible as expenses) and operation and maintenance charges against income (which are deductible as expenses in the year incurred). As shown in Figure 14-1, costs have been divided into five major divisions:

1. Design, development, test, and evaluation costs are the costs of the engineering analyses and testing necessary to convert the system performance specifications into a validated design.

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[2] "Satellite Power System Work Breakdown Structure Dictionary," IN-PP03-76-1 published by the Engineering Cost Group (PP03), Marshall Space Flight Center.

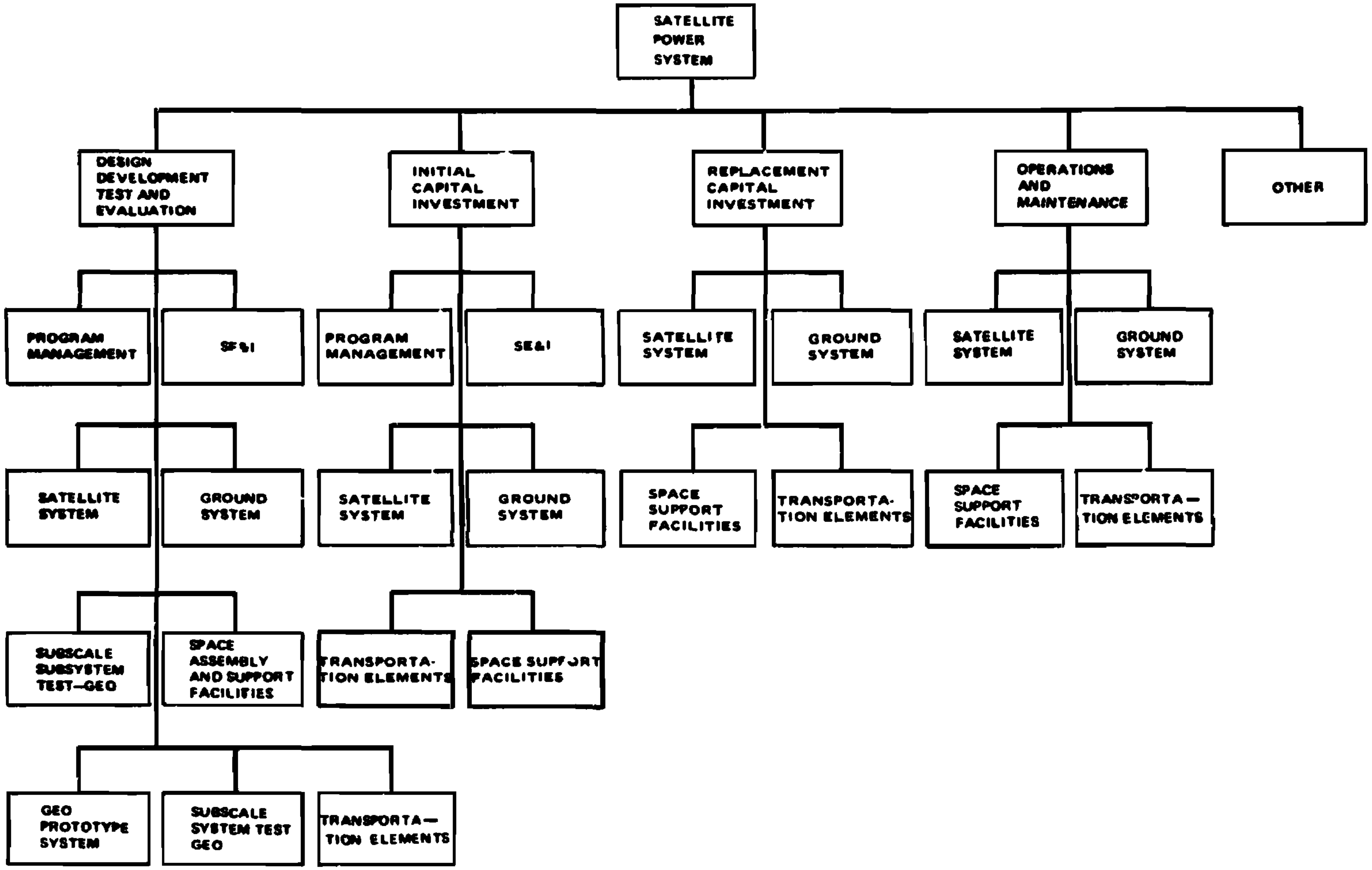

Figure 14-1. Work breakdown structure.

2. Initial capital investment costs are the costs associated with initial procurement and emplacement of the plant and equipment.

3. Replacement capital investment costs are the costs associated with capital asset replacements over the operating life of the SPS (i.e., subsystem spare parts, overhauls, maintenance equipment, etc.).

4. Operations and maintenance costs are the costs of expendables (i.e., argon fuel for the MPD thrusters), minor maintenance, crew rotation, etc.

5. Other costs include the costs of taxes and insurance.

Although not completely shown in Figure 14-1, each of the five major cost divisions is then subdivided into the elements (such as systems, subsystems, components, etc.) at which level the cost analysis is performed.

14.3 GROUND RULES AND ASSUMPTIONS

The SPS program cost estimate and economic analysis were based on the following ground rules and assumptions:

- The cost estimate is in constant mid-1976 prices.
- The estimate includes the cost of a 30 percent satellite weight contingency.
- The estimate includes an across-the-board 15 percent cost contingency.
- A plant load factor of 85 percent is assumed.
- A space transportation to LEO load factor of 80 percent is assumed.
- A typical MSFC photovoltaic 10 GW satellite configuration (116×10^6 kg) is assumed.
- Energy Research and Development Administration (ERDA)/Electric Power Research Institute (EPRI) economic methodology[3] is utilized where applicable.

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[3] Refer to The Cost of Energy from Utility-Owned Solar Electric Systems. ERDA/Jet Propulsion Laboratory 1012-76/3, dated June 1976.

- SPS initial operational capability is assumed to be 1995.

- An SPS operational lifetime of 30 years is assumed.

- A program of thirty or sixty 10 GW stations is assumed. (This report contains cost results of a sixty 10 GW station program analysis.)

14.4 METHODOLOGY

14.4.1 COST MODELING AND ESTIMATING METHODOLOGY

The cost estimates presented in this report were developed from the interim MSFC SPS cost model[4] utilizing parametric cost estimating techniques. Cost estimating relationships (CER) in this model were derived from historical and projected cost, technical, and programmatic data applicable to the SPS cost elements. Transportation and space construction base costs were derived in other MSFC studies and were included in the SPS cost estimate as throughputs.

The interim MSFC SPS cost model is, as the name implies, at an early stage of development and will be updated next year by a more comprehensive model. Although the costs are considered to be the best possible estimate at this time, the CER's have been applied broadly and generally to SPS subsystems which are themselves in the early stages of definition. Because of these uncertainties, an overall system cost contingency of 15 percent has been included. In addition, other contingency factors including weight contingency, subsystem contingency, etc., have been used as appropriate and are so noted. These additional contingencies should reduce cost growth due to unknowns and increase the validity of the estimate. Additional cost data are now being collected, normalized, and documented so that a more accurate cost model will be available for future SPS cost estimates.

14.4.2 ECONOMIC METHODOLOGY

The primary criterion for economic evaluation of the SPS was the generation cost of electrical energy at the ground bus bar in mills per kilowatt-hour (a mill is one-tenth of one cent). As stated, generation cost refers to the cost of energy at the ground bus bar and not the ultimate cost of the energy to the consumer. (A rule of thumb is that transmission and distribution costs are

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[4] "Interim Satellite Power System Cost Model," IN-PP03-76-2 published by the Engineering Cost Group (PP03), Marshall Space Flight Center.

approximately equal to generation costs and, therefore, consumer cost is double the generation cost. However, it is generally not necessary to include transmission and generation costs in a comparative analysis of alternate power plants, since these components of cost would normally be the same for all alternates.)

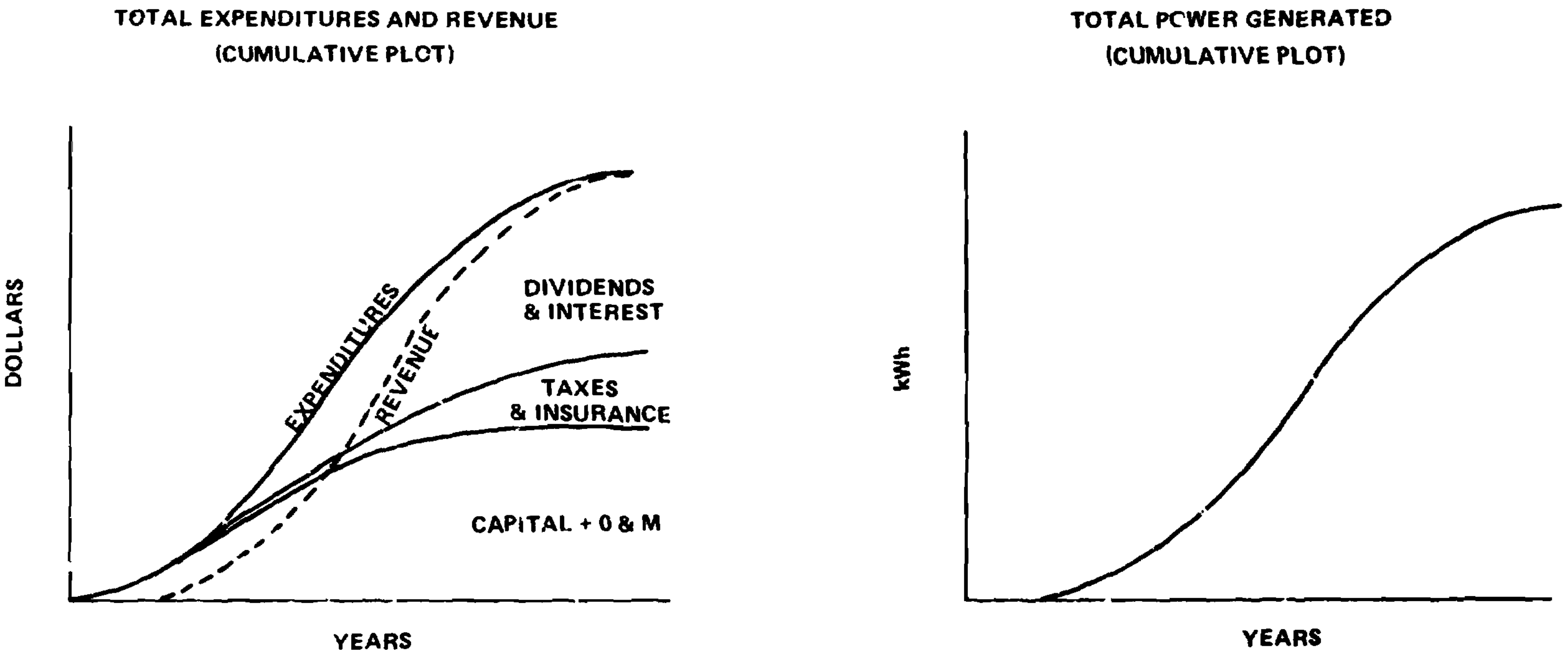
The generation cost for the electricity produced by an SPS is simply that price which must be charged for each kilowatt-hour of energy such that the total money investment in the systems has been recovered at the end of its useful life. If the time value of money did not have to be considered (resulting from the fact that investors in the system require a return on their investments), the price would simply be the total money invested (in mills) divided by the total power generated (in kilowatt-hours). However, since investors (stockholders, bondholders, and noteholders) require a return before they will invest their money, these dividend and interest payments must be passed on and added into the generation cost.

Thus, as shown in Figure 14-2, the generation cost can be calculated by dividing the total expenditures, including the time value of money charges, by the total power generated. Realize that the resulting generation cost automatically includes profit, because the dividends paid to the stockholders are the profits. That is, revenue over and above that used to pay for the operating costs, to repair the facilities, to pay executive salaries, to buy new equipment, to pay debts, etc., is distributed to the shareholders.

In actual practice, however, there are more sophisticated economic techniques available to facilitate the calculation of the generation costs than the method described. Specifically, present value analysis (discounting) was employed in a manner consistent with the capital budgeting methodology that a private utility might use to determine the generation cost of its power plants. Although there are minor differences in execution, the technique employed here is essentially identical in approach and result to the methodology reported in Footnote 3.

14.5 COST ESTIMATES

Table 14-1 and Figure 14-3 show the estimated average unit life cycle cost for one SPS. Costs were estimated at the lowest practicable level, and a breakdown of the major SPS WBS elements with their respective costs are shown in the table. The costs are further subdivided into four program cost divisions, which were described in subsection 14.2.



$$\text{GENERATION COST} = \frac{\text{TOTAL EXPENDITURES (TOTAL REVENUE)}}{\text{TOTAL POWER GENERATED}}$$

Figure 14-2. Generation costs.

TABLE 14-1. SPS AVERAGE UNIT LIFE CYCLE COST (MILLIONS OF 1976 DOLLARS)

	Initial Capital Investment	Replacement Capital Investment	Operations and Maintenance	Other	Total
Satellite System	17 923	478	13		18 414
Structure	1 312	38			1 350
Power Source	1 230				1 230
Antenna	38				38
Rotary Joint	44	38			82
Power Source	12 711	347			13 058
Blankets	11 270	157			11 427
Concentrators	296	1			297
Power Distribution and Conditioning	1 145	189			1 334
Microwave Transmission	2 200	81			2 281
RF Generation	1 280	76			1 356
Waveguides	920	5			925
Propulsion	1 500		13		1 513
Avionics	200	12			212
Ground Station System	3 878	429			4 307
Rectenna	3 118	315			3 433
Utility Interface	760	114			874
Space Construction Base	227	340			567
GEO Space Station	502	753			1 255
Assembly and Support Equipment	84	127			211

TABLE 14-1. (Concluded)

	Initial Capital Investment	Replacement Capital Investment	Operations and Maintenance	Other	Total
Transportation Systems	479	202			681
HLLV	479	148			627
COTV		17			17
POTV		37			37
Transportation Operations	9 570	933	2583		13 086
HLLV	9 570	915	1800		12 285
Shuttle			138		138
COTV		18	180		198
POTV			465		465
Satellite Operations	1 914	183	420		2 517
Taxes and Insurance				18 167	18 167
Program Management and Systems Engineering and Integration	1 729	150	151		2 030
Program Contingency	5 446	471	475	2 725	9 117
Total	41 752	4066	3642	20 892	70 352

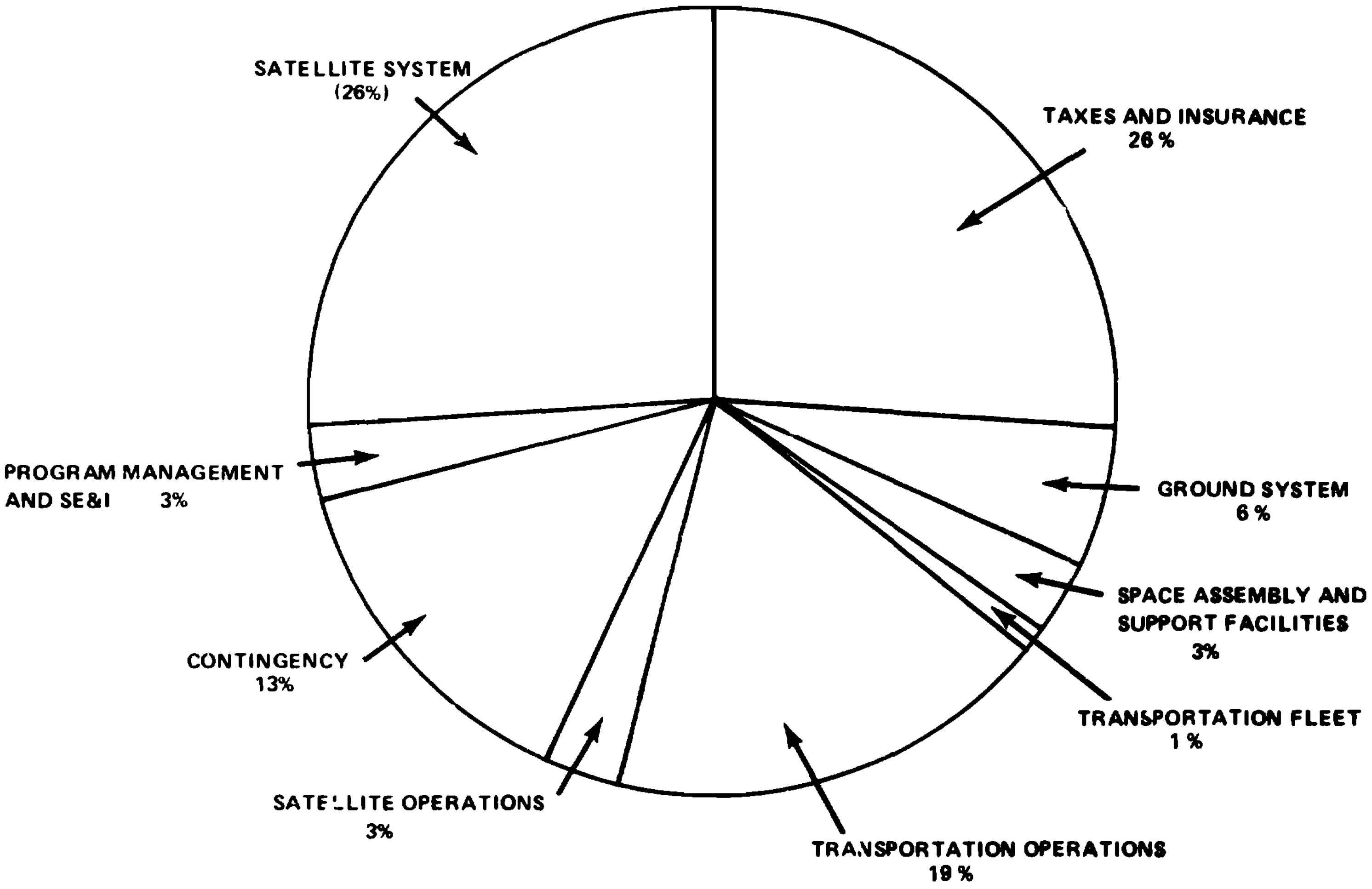


Figure 14-3. SPS average unit life cycle cost.

The feasibility of SPS as a cost competitive energy producer depends on significant cost reductions in a few major areas. For example, Figure 14-4 shows the effect of solar cell cost on the electrical generation cost. Since current solar blanket prices for spacecraft applications range from $5000 per m^2 to $20 000 per m^2, a cost reduction of two orders of magnitude is required. ERDA has set a goal of $64 per m^2 to be attained by 1985. This is is an optimistic goal and is considered to be the low end of the range of 1985 solar blanket prices. The MSFC solar blanket cost shown in Table 14-1 is therefore based on $128 per m^2 or double the ERDA goal to account for space rating and to add a margin of contingency. Although this estimate is still a significant reduction from current prices, it appears attainable when considering the combined effects of projected technology advancements and of mass production and learning.

Another element considered to be a major cost driver is transportation. Figure 14-5 shows the effect of launch to LEO cost on generation cost. It can be seen that approximately one order of magnitude reduction in the current shuttle launch cost is required by the proposed HLLV to provide a competitive generation cost.

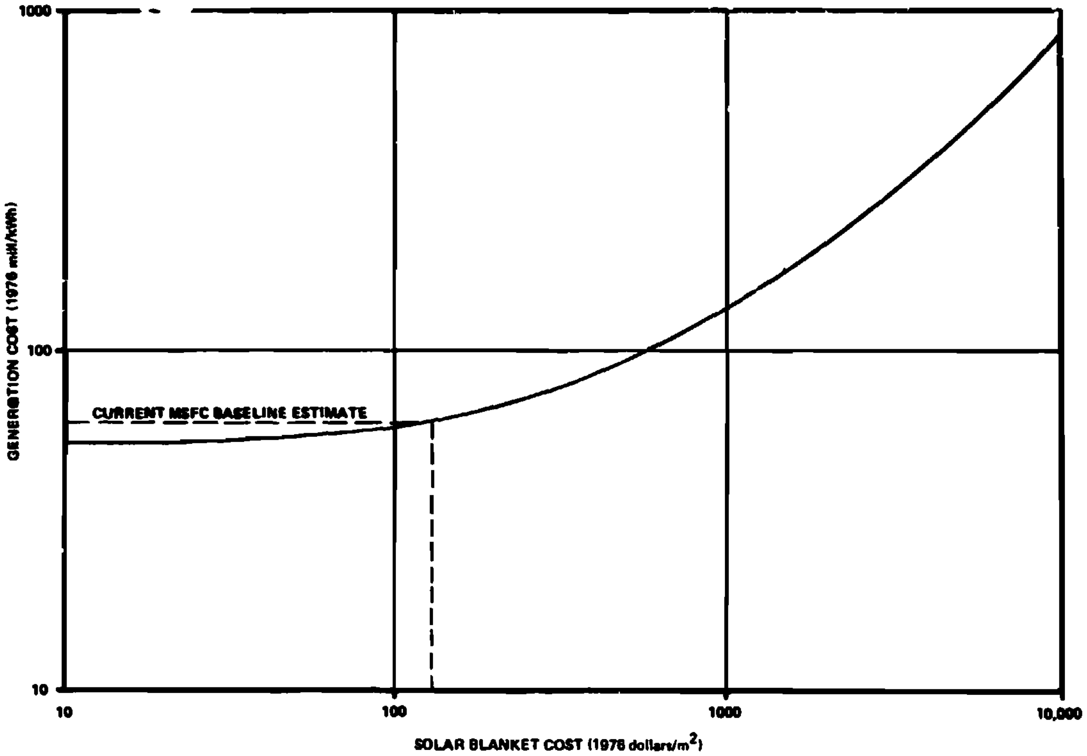


Figure 14-4. Effects of solar blanket cost on SPS economics.

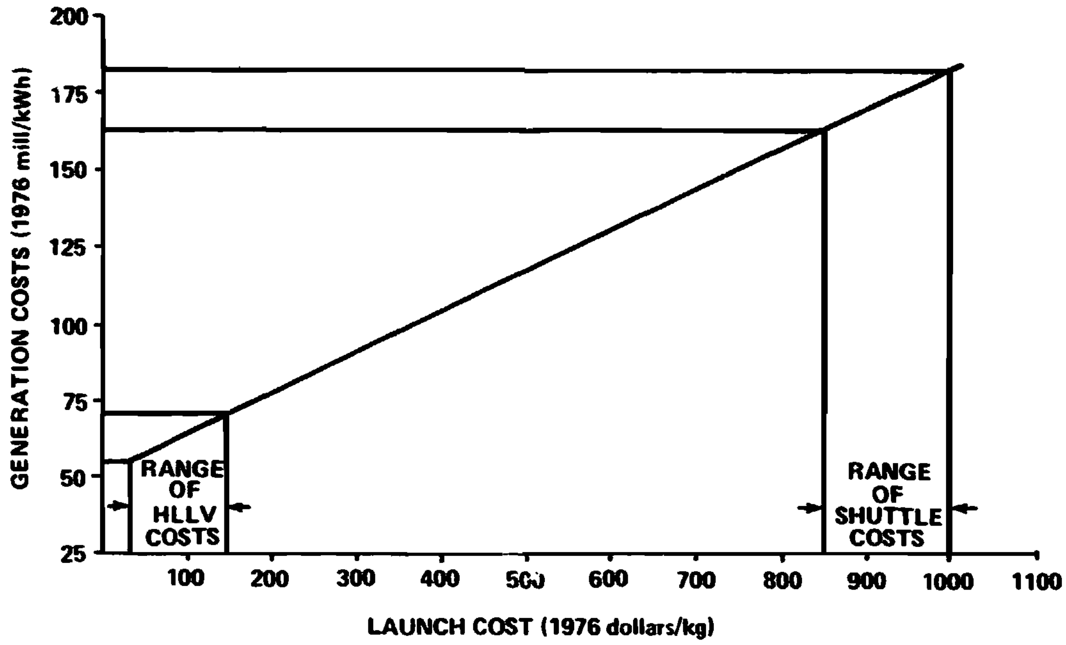


Figure 14-5. Effects of launch to LEO cost on SPS economics.

The HLLV target cost range shown on Figure 14-5 is $30 per kg to $150 per kg; however, current HLLV studies indicate that future transportation costs would more likely range between $60 per kg and $200 per kg depending on the class (size) of vehicle and the annual flight rate. Launch costs for a vehicle of the approximately 225 000 kg payload class with a launch rate of 600 per year are actually estimated at $60 per kg. Assuming SPS would require an HLLV with at least that payload capability and an annual flight rate of 600 to 800, the transportation operations cost was estimated at $66 per kg to include a 10 percent or more growth contingency. While this estimate represents a significant reduction from current launch costs, the HLLV concept of large payload capability, maximum reusability, and short turnaround time is expected to establish the required low cost transportation for the SPS program.

The sensitivity of generation cost to other SPS cost elements was also determined and found to be less sensitive than those mentioned previously. Figure 14-6 shows the effect of structure cost on generation cost. Although the SPS economic feasibility is not dependent on a significant cost reduction in structural members, it could depend on the ability to develop fast and reliable structural assembly techniques.

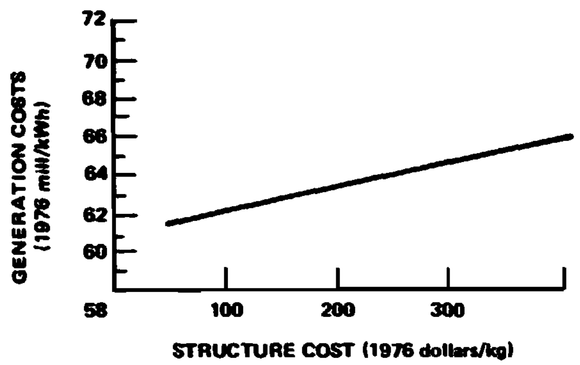


Figure 14-6. Effects of structure cost on SPS economics.

Cost trades have been performed in other areas such as the solar concentration ratio and the Z-solar versus X-POP array orientation. See subsection 7.1.6.5 for results of these trades.

The design, development, test, and engineering costs of the unique SPS elements plus the other major hardware programs that support the SPS (e.g., the space construction base and the various new transportation elements) are shown in Table 14-2. It is not yet known which of these support programs costs are to be attributed solely or in part to the SPS program.

TABLE 14-2. DESIGN, DEVELOPMENT, TEST, AND ENGINEERING (MILLIONS OF 1976 DOLLARS)

Satellite	5 150	
• Solar Array		4 340
• Microwave Antenna		810
Rectenna	470	
Support Equipment	4 230	
• Space Construction Base		2 890
• Fabrication and Assembly Equipment		1 340
Transportation	16 590	
• HLLV		8 030
• Support Tugs		470
• Cargo Orbital Transfer		3 540
• Satellite Orbital Transfer		4 550
SPS Flight Demonstrations	TBD	

14.6 ECONOMICS

14.6.1 SPS CONTRIBUTION TO THE ELECTRIC POWER MARKET

The installed peak electrical capacity of the United States is approaching 500 GW. Although the rate of growth has slowed recently from the lagging effects of the recent recession, it is generally forecasted that the demand for electrical power will increase sharply throughout the remainder of this century and probably beyond. For the purposes of this portion of the study, an SPS power capacity, initial operating capability, and buildup rate were assumed in order to investigate the potential market capture of an SPS program. The capacity chosen was 10 GW per SPS, corresponding to the MSFC baseline concept. Incidentally, 10 GW is as large as power plants go today. A typical large nuclear generating plant is on the order of 3.5 GW (made up of, possibly, three generating units). However, preliminary discussions with representatives of the industry have uncovered no serious problems with future power plants of this size from the standpoint of integration into the utility grid (down time due to shadow occultation, for example, seems not to be a problem as long as it is precisely predictable, as it would be for SPS).

The year of initial operating capability was chosen to be 1995, which seems consistent with an aggressive and orderly technology development program. A buildup rate was hypothesized that sought to maximize the market capture while not overstepping the bounds of reasonableness from technical, logistical, and capital availability standpoints. (This last subject, capital availability, is discussed in more detail later.) It was assumed that between 1995 and 2025, sixty 10 GW stations would be brought on-line at the rate of one per year for the first 10 years, two per year for the second 10 years, and three per year for the third 10 years.

The installed SPS capacity resulting from such a program is graphed in Figure 14-7. Shown also is a forecast of total United States installed capacity requirements over the same period. It can be seen that by the end of the first quarter of the 21st century, the SPS program could be providing 20 percent of our total electrical capacity needs. Considering the fact that we satisfy an increasing part of our total energy needs with electricity and also the fact that SPS would operate as a base load system and would thus provide more than 20 percent of the total electrical energy generated, SPS could make a significant contribution to the projected electrical energy demands.

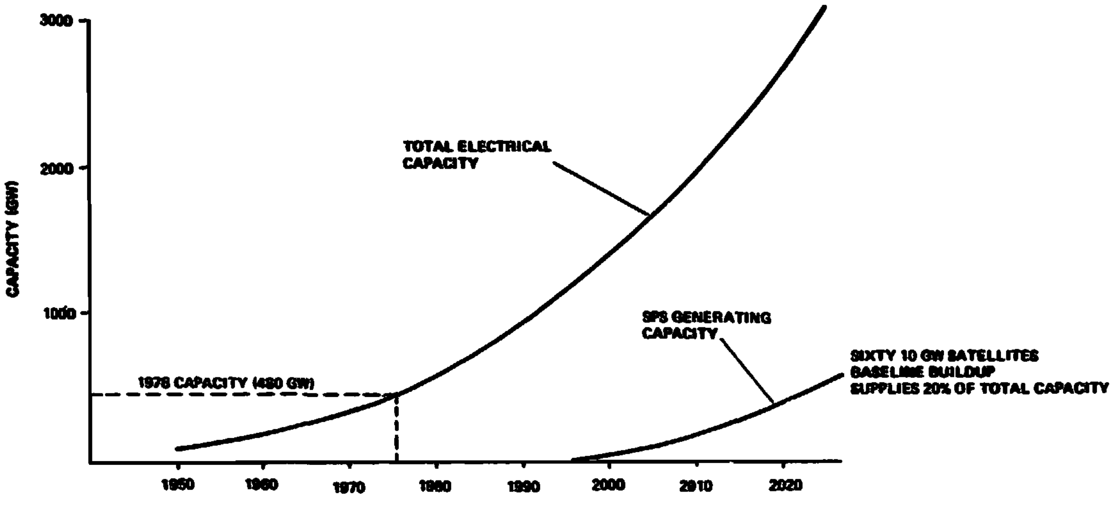


Figure 14-7. Contribution of SPS to United States installed electrical capacity.

14.6.2 PRICE LEVELS AND THE COST OF CAPITAL

To exercise the economic methodology discussed in subsection 14.4.2 and calculate the generation cost of the SPS program, it is first necessary to resolve several issues. Two of the more important assumptions that must be made regard prices (choosing price inflation rates) and the cost of capital (choosing an interest rate to use for discounting).

Prices in the energy sector of the economy have been rising at a rate higher than general price inflation. Power plant capital costs have been rising because of environmental and safety concerns, longer construction lead times, higher interest rates, etc., while operating costs have been rising primarily because of the scarcity of fuel resources. When comparing different types of power plants it is necessary, of course, to take these price changes over time into account. However, it is only necessary to account for the differential energy sector price changes. There is no reason (with the exception of budget planning purposes) to impact the cost estimates with changes in the general price level, because any such changes will affect SPS and other types of power plants equally and thus have no relative effect. Thus, as shown in Figure 14-8, price changes are broken down into two components, those attributable to general price changes and those resulting from energy sector price changes. In this study, general price changes have been ignored (except for interest rate effects), and all costs are in constant dollars of mid-1976 purchasing power.

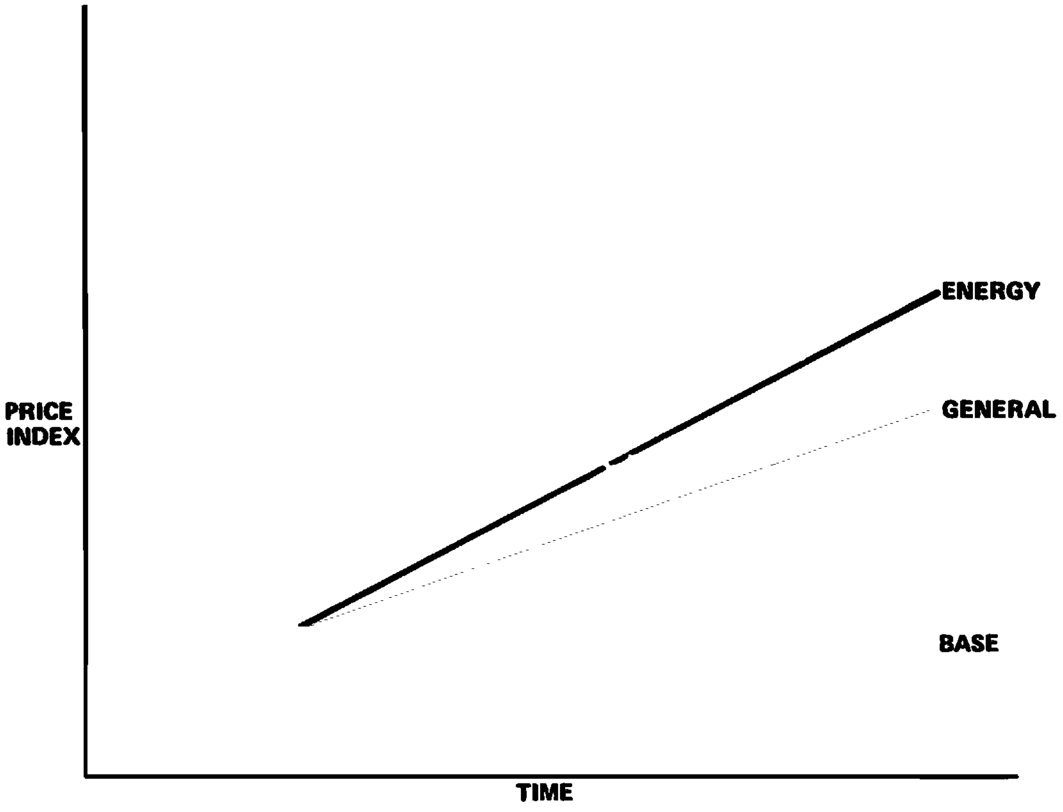


Figure 14-8. Energy price rise relative to general prices.

The difficulty, then, is to estimate the differential price changes of the various power plants being compared. There are many such estimates available in the literature for current and new technology power plants, and this study has used a composite of such estimates to compile ranges of likely costs for power plants other than SPS. For the SPS, sufficient analyses have not been completed to date to quantify any components of differential price inflation beyond those effects of general price changes. Whatever these components may turn out to be (they are currently under study), they will almost surely be less than those that affect the terrestrial systems which rely on depletable fuel resources. Therefore, while it may be argued that certain factors might cause the capital and operating costs of SPS to rise relative to general prices, the current results of this study are based on the assumption that SPS prices will rise at a rate equal to, but no greater than, general price inflation.

As mentioned elsewhere, this economic analysis assumes substantial involvement of the private utility industry in the SPS program and that a large portion of the capital funds will be supplied from private sources. For utility companies, the external money market is reached primarily through the sale of stocks (equity financing), bonds, and short term borrowing (debt financing). Profit from the sale of electricity is used to pay stockholder dividends and to pay the debt plus interest (i.e., the cost of capital). The portion of total funds needed that are raised from these various sources differs among the utilities. Likewise dividends and interest rates paid may vary widely. However, Table 14-3 can be regarded as depicting typical assumptions for the utility industry.

TABLE 14-3. INTEREST RATE

Source of Funds	Market Rate of Return (%)	-	General Price Escalation (%)	=	Real Rate of Return (%)	×	Capitalization Ratio (%)	=	Weighted Rate of Return (%)
Bond and Note Debt	8		5		3		55		1.65
Common Stock	12		5		7		35		2.45
Preferred Stock	8		5		3		10		0.30
									4.4

The interpretation of Table 14-3 is as follows. Utilities raise approximately 55 percent of their external capital needs from the sale of bonds and by borrowing, for which they incur 8 percent annual market interest charges. However, in the face of an assumed 5 percent price escalation, the real interest amounts to only 3 percent because the utility enjoys the opportunity to pay their debt with money that each year is declining in value by 5 percent. Similar interpretations hold for stock financing, such that a weighted rate of return of approximately 4.4 percent is a good approximation of the overall real rate of return for capital investment in the industry. (At 5 percent annual price inflation, this corresponds to a market interest rate of 9.4 percent.) For this evaluation, a 7.5 percent real rate of discount (12.5 percent market rate) has been used, even though for the regulated utility industry such a rate of return is most likely high. This discount rate has been purposely chosen for conservatism, to reflect economic risk, and to account for a certain amount of government involvement and the corresponding 10 percent real rate of discount often required by the Office of Management and Budget (OMB) for public investment projects.

14.6.3 INSURANCE AND TAXES

In the area of insurance and taxes, this analysis departs from the methodology used in other SPS economic evaluations. That methodology consists of applying factors against capital cost to arrive at an annual cost to account for insurance and taxes. The various SPS concepts are very different in their capital intensiveness, and a single factor applied against the capital cost would not take this into account. With a computerized model it is just as convenient to calculate the insurance and taxes more precisely. The results reflect ground system property insurance and property taxes based on average rates being paid by the utility industry. Federal income taxes were estimated by applying corporate tax rates against the taxable revenue after taking the allowable deductions (depreciation, operating expense, interest, and taxes).

In addition, it was assumed that a certain percentage of launches would not successfully place the payload in orbit, and/or the launch vehicle would not be recovered. These losses could conceivably be insured in the same manner as commercial communications satellite launches are insured today. Of course, the underwriter for such insurance would charge a premium based on the long term probability of successful launches such that he would recover the expected payments plus make a profit. Alternately, it can be assumed that the financial burden and risk of a program as large as SPS can only be borne by a joint enterprise arrangement between an aggregate of private companies (possibly including the federal government) and that the resulting joint enterprise would be large enough to absorb launch losses. Theoretically, for a program of the magnitude of SPS with many launches over time where the statistics of infinite processes approximately hold, the only difference between insuring and absorbing launch losses is the profit of the insurance underwriter (there is no tax difference because either losses or insurance premiums are deductible for income tax purposes). The study results reflect the assumption that the company absorbs launch losses.

Finally, in the area of insurance, no provision for third party liability insurance has been included for two reasons. First, the definition of a premium is complicated by the fact that the insured event has a low probability with potentially large claims. Second, current rates for commercial satellites seem quite reasonable, and linear extrapolations to SPS yield premium costs that are essentially negligible (less than 1 million dollars).

14.6.4 CAPITAL AVAILABILITY

As discussed elsewhere in this report, the SPS program has the potential to produce power at rates economically competitive with other types of power plants projected for the 1990's. This is true even though the SPS program is very capital intensive (i. e., the major portion of the expense of the SPS program is the procurement and installation of plant and equipment). The SPS is able to economically compete with less capital intensive power plants only because the SPS is much less expensive in the operational phase of the program. However, since the funds available in the capital market are limited, the large SPS capital requirements could hamper the implementation of the program.

It is somewhat analogous to the situation of the space agency in the early 1970's when comparing various space transportation options. Cost benefit analyses showed that a partially reusable space shuttle was more economical than expendable rockets and a fully reusable space shuttle was even more economical. Even though the fully reusable version was the most economical, the initial cost was simply too high for the agency budget, and the partially reusable version was chosen as a compromise.

In the 1990's an SPS will probably require two to four times the initial plant and equipment expense of other types of plants. Already, even without SPS, the utility industry is concerned about the availability of funds to sustain the required growth in electrical capacity. However, the utility industry is accustomed to large capital requirements and, in fact, is the most capital intensive of all United States industries, accounting for approximately one-fifth of all United States industry plant investment. Many potential solutions to the financing problem are being investigated. These include increasing the attractiveness of external investment by such plans as tax free dividend reinvestment and increasing internal cash generation by increasing the investment tax credit, more rapid depreciation rates, allowing a tax deduction for dividends, etc. At any rate, the problem of power plant financing is not peculiar to the SPS; to supply the future electrical power needs of the nation, by whatever means, requires that ways be found to meet large capital requirements.

14.6.5 GENERATION COST

The SPS generation cost is given in Table 14-4 and Figure 14-9 broken out by major WBS cost elements. For the typical SPS concept studied, the 1995 total generation cost in 1976 prices is estimated to be 62 mill/kWh. The largest single contributor to generation cost is the satellite itself which corresponds to 21.9 mill/kWh or 35 percent of the total. Although not shown,

TABLE 14-4. SPS AVERAGE UNIT GENERATION COST (1976 mill/kWh)

Cost Element	Initial Capital	Replacement Capital	Operations and Maintenance	Other	Total
Satellite System	21.6	0.3			21.9
Ground System	4.7	0.3			5.0
Space Assembly and Support Facilities	1.2	0.4			1.6
Transportation Fleet	0.7	0.1			0.8
Transportation Operations	10.9	0.5	1.6		13.0
Satellite Operations	2.2	0.1	0.3		2.6
Taxes and Insurance				6.6	6.6
Program Management and Systems Engineering and Integration	2.1	0.1	0.1		2.3
Contingency	6.6	0.3	0.3	1.0	8.2
Total	50.0	2.1	2.3	7.6	62.0

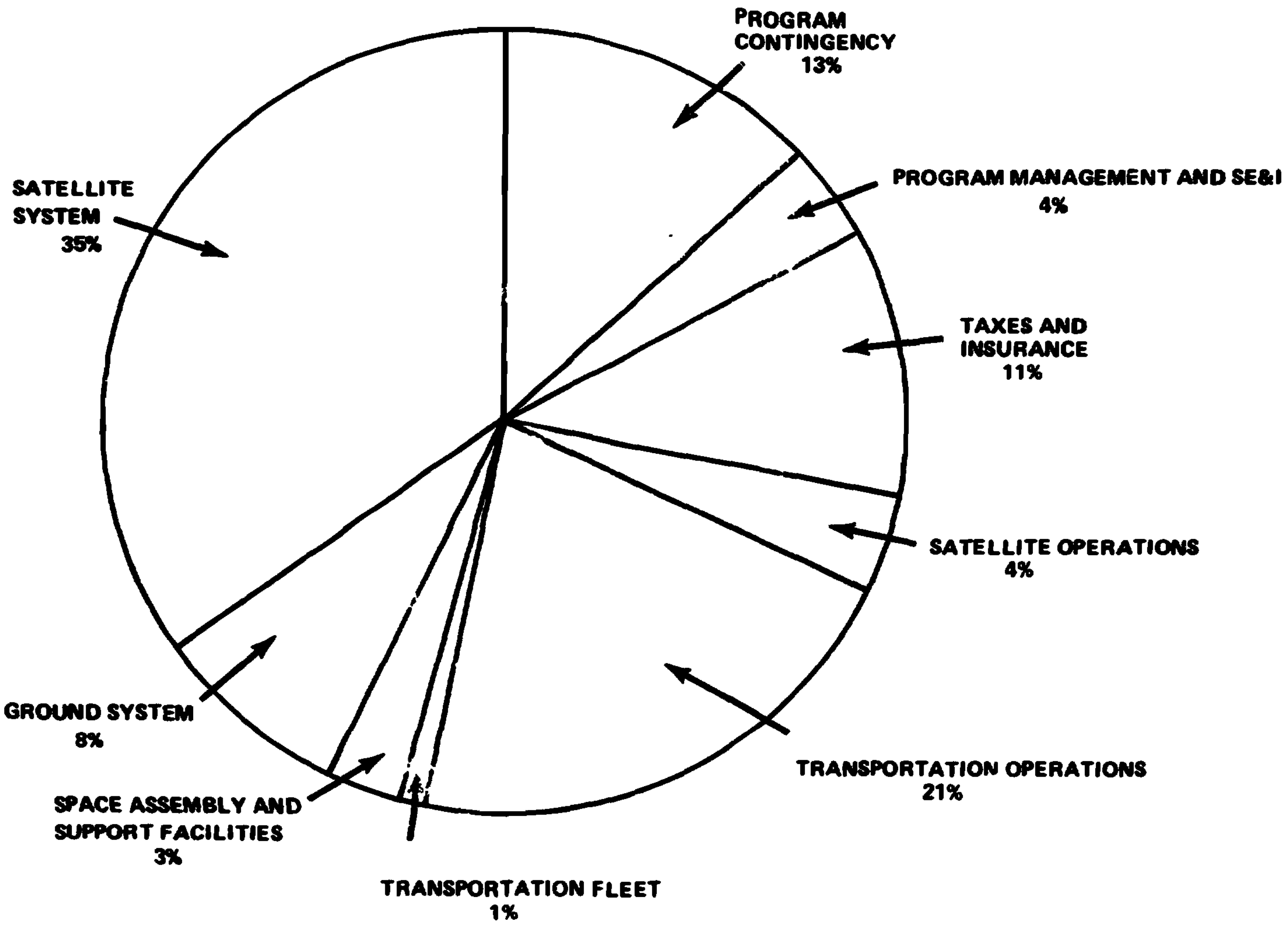


Figure 14-9. SPS average unit generation cost.

nearly two-thirds of this cost is due to the solar blankets. Total space transportation cost (fleet and operations) is the second most important contributor at 13.3 mill/kWh or 22 percent.

It is interesting to compare these results with Table 14-1 and Figure 14-3, which listed the life cycle cost of the SPS by major contributor. Such a comparison reveals a shifting perspective of the major cost drivers of the program. For instance, Figure 14-3 shows that in terms of life cycle cost, the cost of the satellite and the cost of taxes and insurance each contribute 26 percent of the total. However, in terms of generation cost, Figure 14-9 shows that the satellite accounts for 35 percent of the total while taxes and insurance are down to 11 percent. This is because the generation cost takes into account the time value of money and gives a more realistic comparison of the relative importance between cost elements. The satellite is primarily an initial capital cost item and therefore incurs high financial charges. The taxes and insurance, on the other hand, are costs which are incurred at a later point in time and are therefore affected less by the time value of money.

As stated previously, the generation cost of 62 mill/kWh is a realistic estimate for the MSFC baseline concept including a number of contingencies and based on factors discussed throughout this report. In subsection 14.5, economic sensitivities to certain of these factors were discussed (launch costs, for example). In Figure 14-10 the 62 mill/kWh baseline generation cost has been expanded to a range of values or error bars, which correspond to a range of reasonable assumptions. This figure compares the expected range of the SPS generation costs to the range of expected costs taken from various sources for other types of power plants for the 1990's. The conclusion is that, while costs are uncertain for all of those power plants, the SPS is in the competitive range of generation costs predictions.

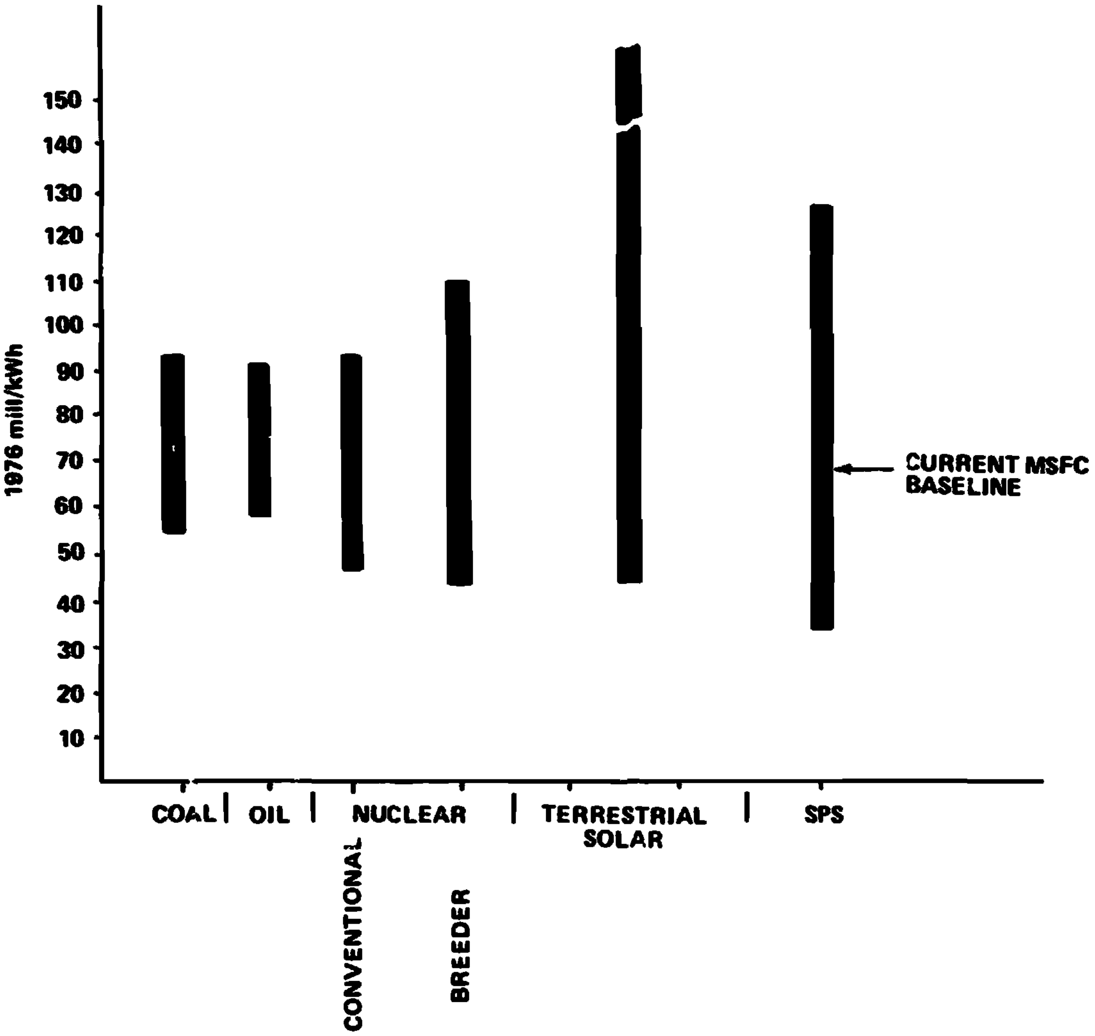


Figure 14-10. Year 1995 generation cost comparison.

15.0 PROGRAM PLAN, TECHNOLOGY ADVANCEMENT PLAN

15.1 PROGRAM PLAN

The program plan for developing the SPS consists of four specific phases: feasibility assessment, technology advancement, development, and operational capability. The feasibility assessment phase of the plan began approximately in 1975 with the formation of a team of engineers with representation from each NASA center to evaluate the potential of the SPS concept. The results of the team activity were a recommendation for further study of the SPS concept and an identification of key discipline technology drivers. The feasibility assessment phase is to end in 1980. The primary purpose of this phase of development is to compare major competing concepts for the SPS and, at the feasibility level, resolve the major technology questions concerning each concept. At the end of this phase the number of concepts should be narrowed to approximately two and hopefully to a single preferred concept, since at this time large ground and flight test programs are envisioned that will be expensive to conduct on a parallel and competitive basis.

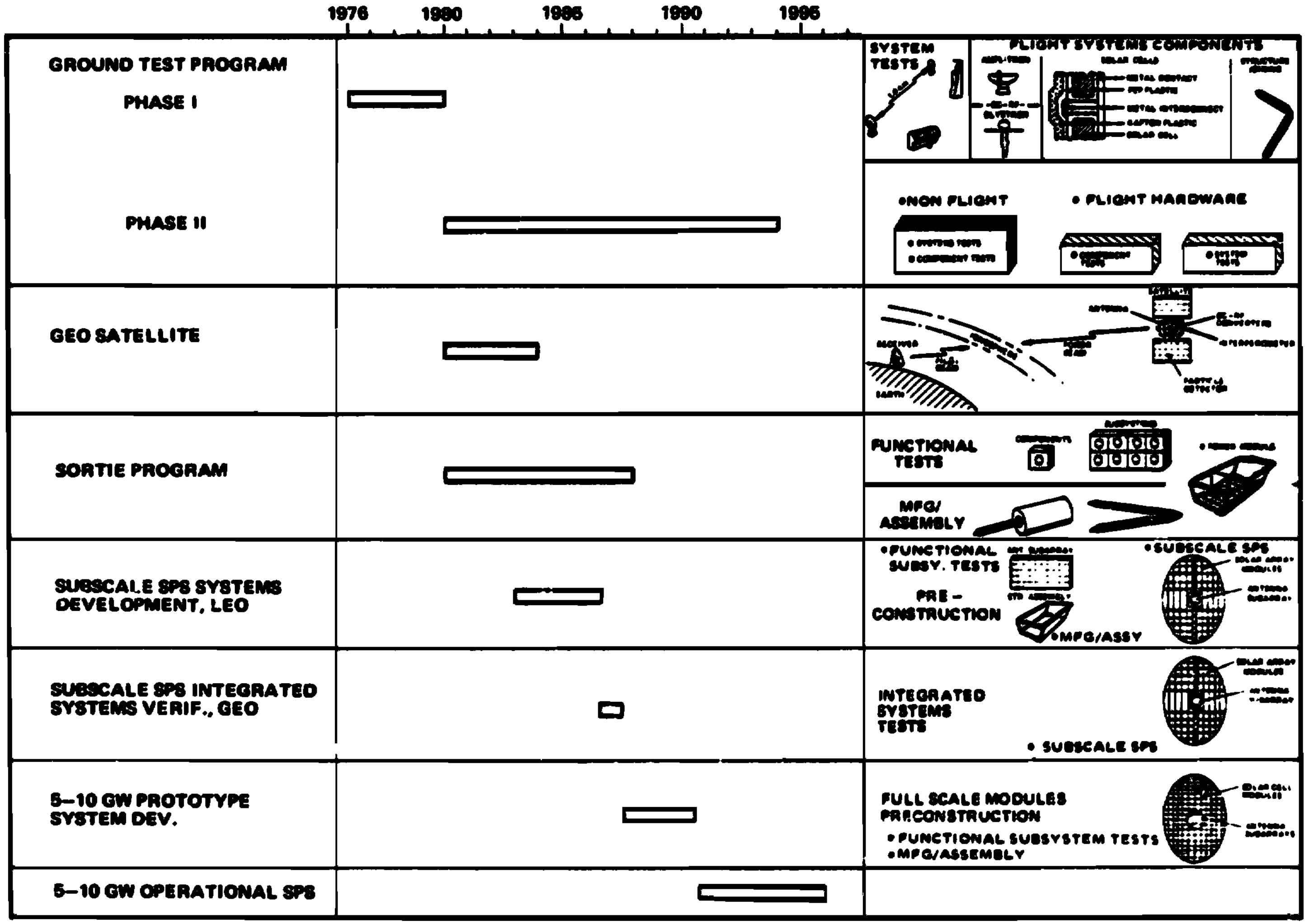
The technology advancement phase of SPS development is to begin in 1980 and proceed to 1987. This phase of development is to begin with a continuation of ground technology work initiated in the previous feasibility assessment phase, is to have ground test work in support of a flight test program, is to feature sortie and space station support flight tests, and is to be marked with the milestones of a 150 kW power module in the 1982 through 1984 time frame and a 2 MW subscale SPS in the 1985 through 1986 time period.

To begin the development phase a commitment to development of the SPS, to the required ground-to-orbit launch vehicle, and to the orbit-to-orbit launch vehicles will be necessary. Within the development phase, the design and construction in orbit of a prototype or first unit SPS will be concluded and the development of SPS production capability will occur. The production of as many as 100 to 125 SPS units is a possibility.

The operational capability phase of the SPS is envisioned as beginning in 1995. During this phase as many as 10 SPS units could foreseeably become operational per year.

The SPS technology development plan consistent with the program plan just outlined is shown in Table 15-1.

TABLE 15-1. DEVELOPMENT/VERIFICATION PROGRAM OUTLINE



15.2 TECHNOLOGY DEVELOPMENT PLAN

The Phase I ground test program of the technology development plan is to consist of selected computer simulations and component tests in each discipline related area of the SPS, such as the microwave power transmission, power conversion, power distribution, attitude control, structures, assembly, thruster, and environmental impact areas. To be included are overall system tests of microwave power transmission. Phase II of the ground test program is to include continued component tests in each discipline of the SPS, subsystems and systems tests, and tests in support of the flight test program. This second phase of the ground test program is expected to continue until completion of the SPS development and operational status is achieved in the 1995 time period.

A small GEO satellite is an envisioned part of the technology development plan. This satellite, which is to be operational prior to 1984, is to be deliverable with the shuttle in an upper stage to geosynchronous orbit. Results to be derived from this test are all systems operation and compatibility data that will reflect an acceptable selection of materials, thermal coatings, system locations, etc.; selected long duration component test data, such as on the power amplifiers; spacecraft charging data, microwave beam control, attenuation, and scintillation; and RF interference characteristics in the environmental impact area.

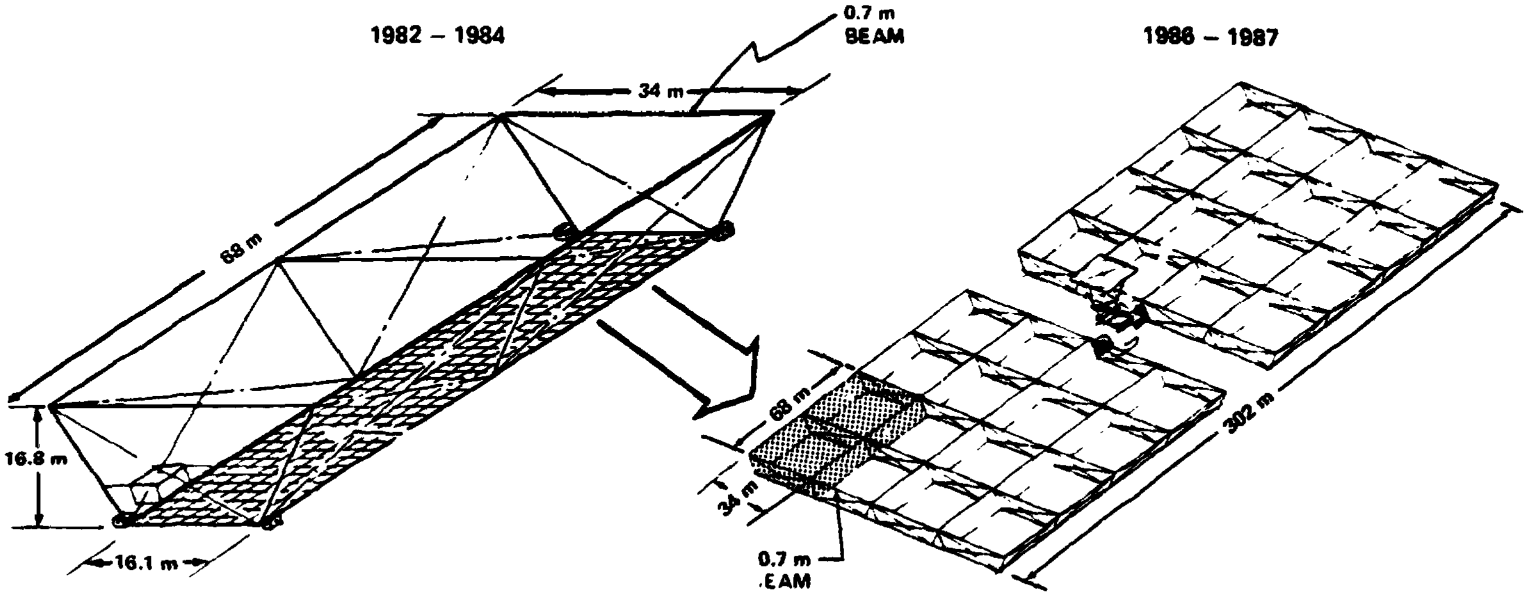
A vigorous shuttle supported sortie mission program is included within the technology development plan. This sortie program is to be required through 1987. The major features of the sortie program are component and subsystem tests in support of each SPS discipline area. Preconstruction tests are to be included in the sortie program to initiate learning on assembly of structure in orbit and installing subsystems. Sortie missions using a long duration exposure facility or equivalent carrier structure are envisioned for support of long duration test programs on materials, material coatings, systems requiring continuous exposure to the space environment, certain continuous cyclic operations, etc., where manned interface is not directly required for obtaining test data. The sortie mission program results are to be focused with the construction of a 150 kW power module from the shuttle. This module is to be capable of supporting and supplying power for low orbit or GEO space industrialization, to provide power to the space station, and to provide a technology base for construction of the subscale SPS. Potentially the power module could be constructed as subelements of the subscale SPS.

The subscale systems development portion of the technology development plan is to be initiated in early 1983. The major hardware objective of this effort is to be a subscale SPS, potentially of the 2 MW ground output size.

At least arguments can be made for a subscale SPS this size from the point of view of the microwave system. For the 2 MW size system, but not for a smaller power level, the microwave antenna would be a subelement of the full scale system and this subelement can have power taper as a test variable. The subscale systems development effort is seen as having four major chronological parts: systems environmental and performance program, which is to provide tests of systems in which long duration and manned interface are combined and to provide qualification tests for subscale SPS subsystems; preconstruction techniques development, which is to be a continuation of preconstruction techniques development from the sortie program and to be the effort that finalizes the procedures for fabrication and subsystem installation for the subscale SPS; construction base/site preparation, which is the buildup and organization of material and equipment to accomplish the subscale SPS construction; and subscale construction, which is to be the actual construction of the subscale SPS. The subscale SPS is to provide the basic technology reference from which to construct the full scale SPS. The intent is that primarily the preconstruction effort for the full scale SPS be characterized by modifying subscale construction technology for the larger scale systems. In the subscale program the techniques and machines for manufacturing the basic structure are to be used for manufacturing the basic beam elements of the full scale SPS. Free-flyer handling and joining of beams in the subscale program will be applicable to the full scale program with modifications to procedures for beam size. The subscale SPS is to be operated in LEO to verify performance of the system and is to be transferred to GEO for further performance tests. The space station is to be the control center for operating the subscale SPS in LEO or GEO. A potential concept for the 2 MW subscale SPS is shown in Figure 15-1 for the photovoltaic power conversion concept. A concept for the 150 kW power module evolving from the 2 MW subscale system is also included in Figure 15-1.

Following the subscale SPS development, the development of a full scale SPS prototype is seen as the next major milestone in the technology development plan. Prototype development is to be a single and first unit development to distinguish it from the operational SPS, although the prototype system will be operationally functional.

The final item of the technology development plan is development of the capability to produce SPS's at the rate of 1 to 10 per year. This effort is to be completed by 1995.



150 kW
SOLAR ARRAY

(LEO)

CHARACTERISTICS

NOMINAL POWER	130 kW (LIGHT & DARK)
SOLAR PANEL OUTPUT	260 kW
SOLAR PANEL AREA	1100 m^2

2 MW
SPS SUBSCALE

(LEO/GEO)

CHARACTERISTICS

POWER LEVEL	2 MW GROUND OUTPUT
OVERALL LENGTH	302 m
WIDTH	136 m
ANTENNA DIAMETER	19 X 20 m

Figure 15-1. SPS feasibility demonstration facilities.

16.0 CONCLUSIONS

The large number of system and subsystem elements under continued study required extensive tradeoffs and comparisons of many options in each area. These efforts are far from being complete and require further extensive in-depth analyses and optimizations; therefore, only the following tentative conclusions can be drawn at this time.

- This study effort has improved and expanded the understanding and identification of critical technology areas and of required technology advancement and verification programs.
- Progress has been made in further definition of the key issues associated with each program element.
- Improved cost and economic analyses have shown that SPS remains cost competitive with other energy sources for the planned implementation period.
- All study results indicate that further program definition efforts are warranted.

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APPENDIX A

SPACE-BASED SOLAR POWER CONVERSION AND DELIVERY SYSTEMS STUDY[1]

A study of space-based solar power conversion and delivery systems was initiated by NASA, George C. Marshall Space Flight Center, on February 1, 1975, with ECON Inc., as prime contractor and with Grumman Aerospace Corporation, Arthur D. Little, Inc., and Raytheon Company as subcontractors to ECON. The initial study effort ended November 30, 1975, and resulted in an interim report released March 31, 1976. This phase of the study examined potential concepts for a photovoltaic satellite solar power system (SPS) (focusing on power levels of 5000 and 10 000 MW) and a power relay satellite, and studied certain aspects of the economics of these systems. An artist's concept of the 5000 MW SPS is shown in Figure A-1.

The initial phase of the study addressed the economic feasibility of an SPS. Potentially possible, but futuristic goals were set on the various technologies necessary for an SPS and the system capital and operation and maintenance costs estimated based on the assumption that all the goals are achieved. This effort led to an SPS capital cost estimate of $7.6 billion and an annual operation and maintenance cost estimate of $136 million for the 5000 MW system. This system was economically compared with terrestrial fossil fuel systems (coal and oil) over the period 1995-2025 as shown in Figure A-2. The line R in Figure A-2 relates the generation cost of electric power to the capital cost of an SPS assuming this operation and maintenance cost. Thus, given a price of power, this figure provides the allowable unit of cost of an SPS.

Projections of the cost of power generation for coal and oil systems were made based on three scenarios:

1. Relative fuel prices[2] remain constant (C_O, O_O)

2. The relative prices of coal increase by 2.6 percent per year, and the relative price of oil increases by 0.67 percent (C_A, O_A)

3. The relative prices of coal and oil increase by 5.0 percent per year (C_B, O_B).

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[1] Synopsis — NASA/MSFC Contract No. NAS8-31308.

[2] "Relative prices" refer to the price relationship of all goods and services to each other. The usual practice is to consider one good as the baseline and calculate all prices relative to it. Obviously, generalized inflation would not affect relative prices.

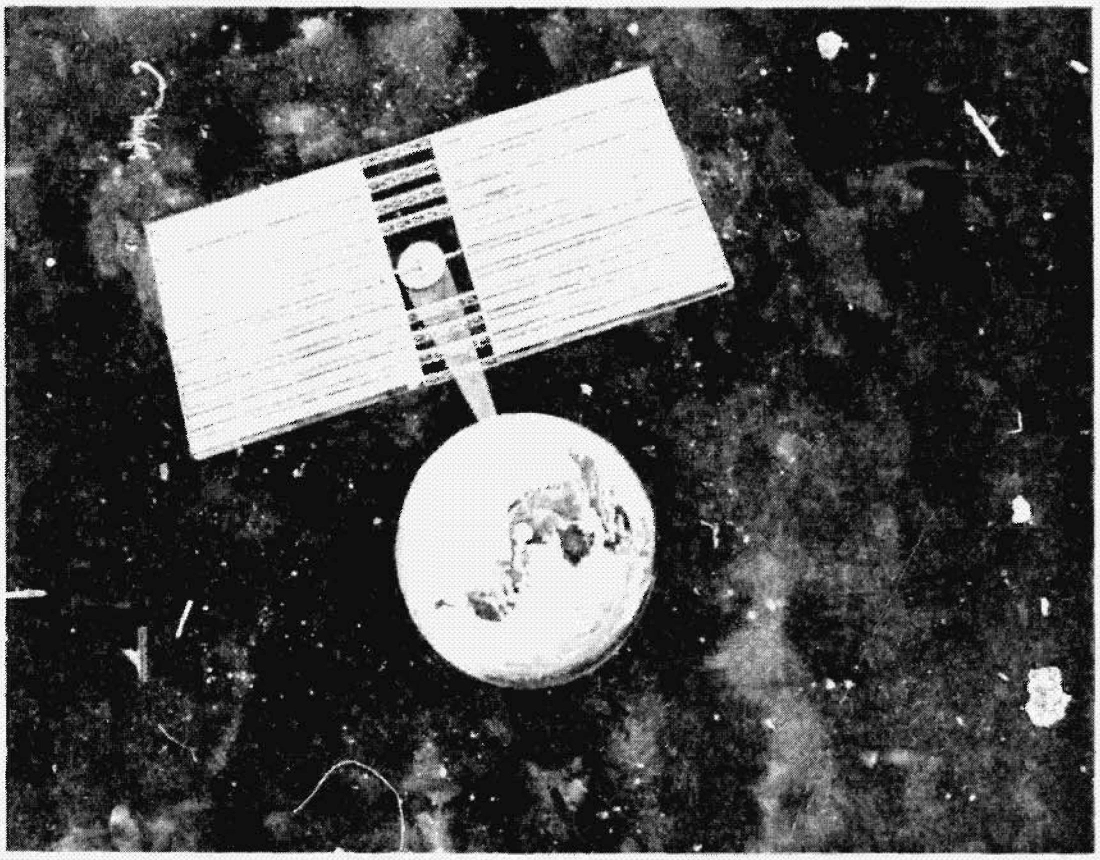

Figure A-1. Artist's concept of a 5000 MW SPS.

Given the results summarized in Figure A-2, the conclusions of the first study phase are that, given appropriate technological advances and continued increases in the real cost of generating electrical power by terrestrial systems, SPS might become economically viable by the mid-to-late 1990's and that it is unlikely that the power relay satellite will become economically viable at any time over the study period — through 2025.

The second study phase, conducted during the period February 1 to June 30, 1976, examined in greater depth the technical and economic aspects of SPS with a focus on the current configuration 5000 MW system. In this phase of the study, the following efforts were requested:

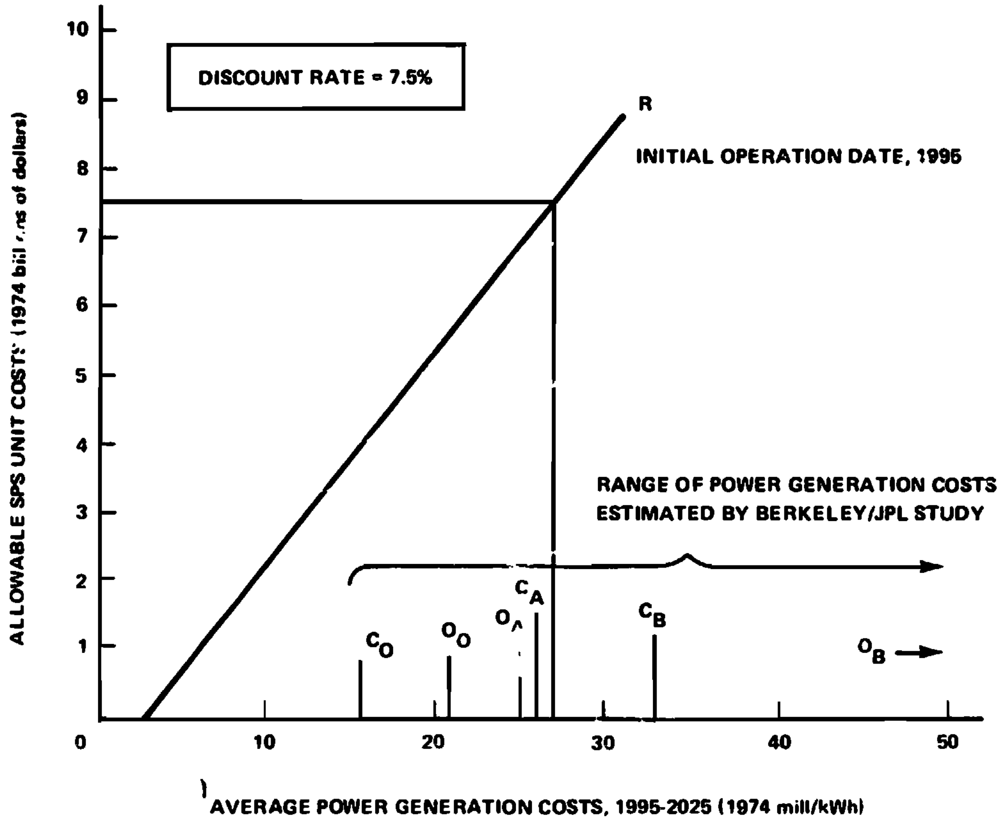

Figure A-2. Economic comparison of a 5000 MW SPS operating over the period 1995-2025 with terrestrial fossil fuel plants.

1. Additional engineering studies of the current configuration focusing in the following areas:

   a. Orbital system structures

   b. Control and station keeping analysis

   c. Flight mechanics and orbit transfer stresses.

2. An analysis of alternative program plans focusing on the economics of low Earth orbit and geosynchronous orbit test satellites.

3. An identification of critical issues and technologies relevant to the construction and operation of a photovoltaic STS.

4. A preliminary identification and analysis of utility interface issues and problems.

The continued refinement of cost estimates and costing methodologies was of particular interest during this phase of the study. To satisfy these objectives a significant part of the study effort was devoted to the development and use of cost and cost-risk analysis computer models. These models are used to combine economic and technical data to provide information for programmatic decision making. Thus, the results of the second study phase differ from those of the first study phase in two basic ways:

1. They are nondeterministic; that is, results are given in the form of probability distributions rather than single number estimates.

2. It is not assumed that all technology goals are met, rather the chance of achieving various levels of technology developments were assessed and the corresponding system cost impacts evaluated.

The results of the cost-risk analysis were used to assess the economic viability of various SPS development program plans and to identify the critical technology areas. The economic viability, that is, the net present value, of the second SPS unit was assessed subject to the following assumptions:

1. The SPS unit availability factor is 0.95; that is, it is producing power 95 percent of the time. This includes power outages due to solar eclipses near the equinoxes.

2. The power output of the SPS unit decreases by 1 percent per year due to degradation of various components.

3. The lifetime of the SPS unit is 30 years.

4. The capital investment in the SPS unit is made in one lump-sum payment 2 years prior to the initial operation date of the SPS unit.

5. The real price of power at the rectenna busbar (1974 dollars) increases at the rate of 1 percent per year.

6. No charge is made for taxes and insurance.

7. Present value computations use a discount rate of 7.5 percent.

The probability distribution of net present value of the second unit is shown as a function of the price of power at the rectenna busbar on the first day of operation in Figure A-3. If the price of power at the busbar on the first day of operation is 30 mill/kWh (1974), Figure A-3 indicates that there is approximately a 21 percent chance (approximately one in five) that the second unit will be economically viable. Given these results, it is the low probability of a substantial payoff that is used to develop and justify SPS development program plans directed toward the pursuit of the SPS concept. Each step in such a program is viewed as a process of buying information necessary to make the decision to continue the development program.

The major conclusions derived by the study to date are as follows:

1. The SPS may be cost-effective with respect to terrestrial fossil fuel systems by 1995.

2. Potential cost-effectiveness of SPS improves beyond 1995 due to:

    a. Increasing scarcity of nonrenewable energy resources

    b. Increasing environmental impact of terrestrial alternatives

    c. Continued technology development (learning).

3. SPS has the potential for supplying approximately 10 percent of the U.S. energy needs by the year 2025 (equivalent to total present consumption).

4. It appears that an economically viable and risk controlled program plan to pursue the SPS concept can be developed.

5. The technology areas critical to the successful development of an SPS are:

    a. The ability of man to manufacture and assemble equipment in space

    b. Solar cell technology (mass, efficiency, lifetime, cost).

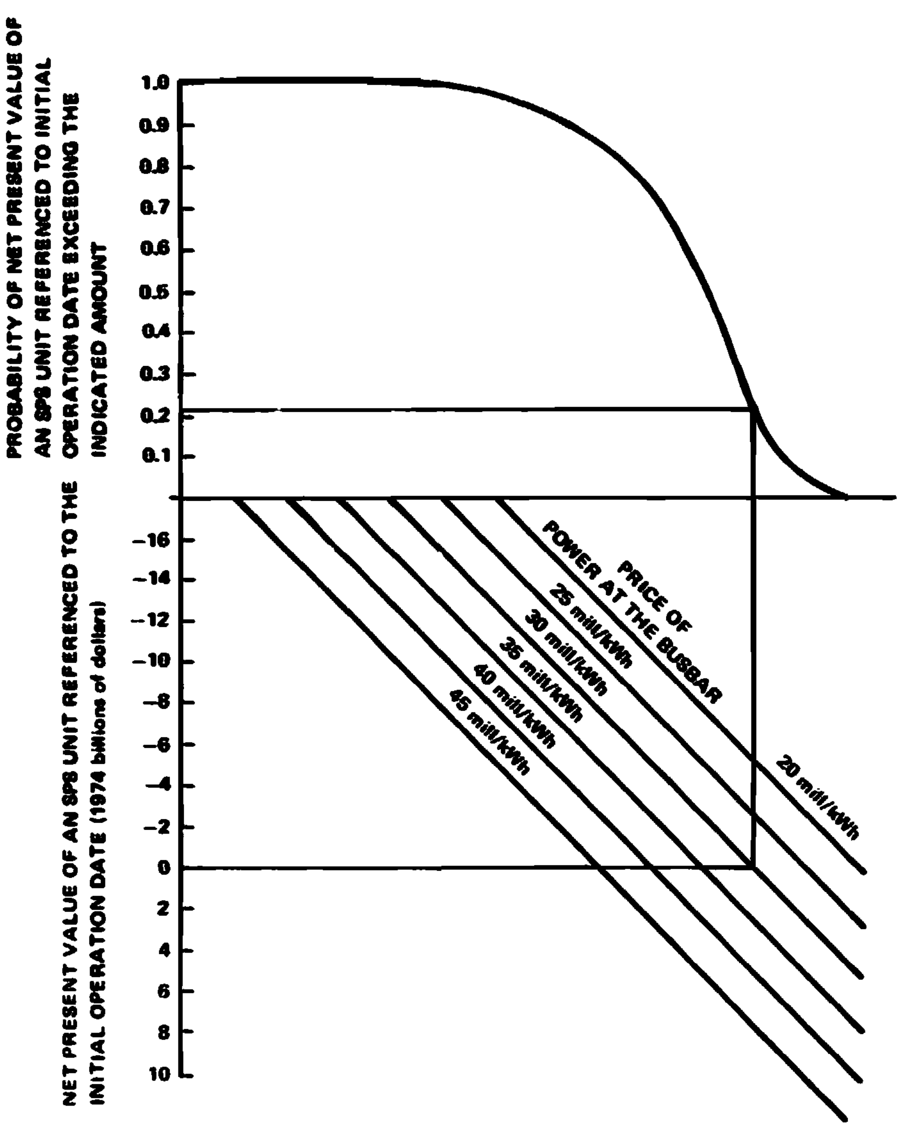

Figure A-3. Cumulative distribution function of net present value of an SPS unit at the initial operation date as a function of price of power at the rectenna busbar.

APPENDIX B

SYSTEMS DEFINITION – SPACE BASED POWER CONVERSION SYSTEMS[1]

BACKGROUND

This 16 month study was still 3 months from conclusion when this summary was prepared.

The study is investigating potential systems for the generation of electric power in space. Electric power would be beamed to Earth for use by consumers. Seven candidate power generation concepts were investigated (five have solar energy as the basic power source; two use nuclear reactors):

1. Solar Brayton Cycle
2. Solar Thermionic Di [illegible] diation Cooled
3. Solar Photovoltaic
4. Nuclear Brayton Cycle
5. Solar Thermionic Liquid Cooled
6. Solar Thermionic/Brayton Cycle Cascade
7. Nuclear Thermionic.

In addition a Power Transfer System was investigated. This was a system of mirrors in geosynchronous orbit to reflect sunlight to ground solar power plants.

At the 10 month point, a study extension placed emphasis on the first four of the previously listed systems.

DEMAND ANALYSIS

Analysis of past growth rates in electric power generation capability, population trends, etc., led to the baselining of the following SPS programs:

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[1] Preliminary Summary – NASA/MSFC Contract No. NAS8-31628.

End of concept definition phase	**1979**
Completed technology verification phase	**1987**
Initial operational capability	**1996**
Total ground output of 155 GW achieved (10 percent of national generating capacity[2])	**2006**
Total ground output of 605 GW achieved (25 percent of national generating capacity[2])	**2016**

THE POWER TRANSFER SYSTEM

Fundamental optics provides that the image cone from a mirror cannot have a smaller angular width than the cone to the (circular) source. The Sun as seen from the vicinity of the Earth has an angular width of 0.5°. Hence an image cast from a geosynchronous orbit mirror will have a minimum width of 0.5° as seen from the mirror. This is a width of 330 km at the Equator. Consequently the minimum area of illumination is approximately 86 000 km^2. If the image strength is to be equal to a "noon Sun" in the target area, it is necessary that the orbital mirrors have a total area in excess of the ground target area. After allowing for reflectivity losses and other inefficiencies, the smallest mirrors (for a "one Sun" ground image) had an area of 134 000 km^2 for a mid-U.S. location.

Orbital mirrors of this type may have significant environmental impact. No target area could be found within the U.S. which is not currently occupied by at least 50 000 persons. The nearly constant illumination of the target area would tend to produce high temperatures, perhaps 65°C, although storms may tend to form, partially obscuring the Sun and/or mirrors.

SUBSYSTEM ANALYSIS

Several subsystems have broad utilization among the seven power conversion systems studied. These subsystems were consequently investigated individually for a wide range of operating parameters, permitting them to be parametrically described for optimization studies of the total systems (which are composed of several subsystems).

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[2] Based on a 4.5 percent annual growth from 1976.

Radiators

Radiator panel concepts were evaluated to identify that tube/fin arrangement producing maximum power dissipation per unit mass while providing a baseline level of resistance against meteoroid penetration. The resultant panels were included in a computer model which provided the total radiator system mass (for panels, manifolds, pumps, and pumping power penalty). Manifold mass was found to be extremely critical, requiring a change from a linear radiator arrangement to a more compact "halo" which more closely clusters the radiator area around the heat source. Freezing of the liquid metal fluid of the radiator was found to occur during solar occultation. Radiators with heat pipe panels may be resistant to manifold freezing.

Solar Concentrators

The concept baselined for high concentration ratio (over 200) systems involves a large number (5000 or more) of individual steerable reflectors which form an approximately paraboloidal surface. A relatively light structure, which is somewhat compliant to stresses from thermal transients, gravity gradients, attitude control system operation, etc., is used to support these steerable facets. The facets are composed of metallized plastic film tensioned by a support frame. A computerized model of these solar concentrators was used to determine the sensitivity of the following variables:

1. Geometric concentration ratio (concentrator area ÷ aperture area)

2. Number of facets

3. Rim angle (angle between concentrator plane and aperture)

4. Facet pointing error

5. Sun angle relative to concentrator axis.

A range of concentrator types was also analyzed for use with solar cells, including both two-dimensional (strip) and three dimensional types.

Mass estimating relationships were developed for these concentrators.

An analysis of meteoroid effects indicated no problems for such concentrators; however, radiation effects could not be quantized and are therefore considered a potential problem.

Thermal Engines

Turbomachine systems were investigated by the Garrett Corporation, a subcontractor. Available and "emerging" materials were analyzed as candidates for the high temperature/high stress elements. Most promising were Astar 811C, a tantalum alloy, and silicon carbide, a ceramic. All portions were designed for a 30 year running life. Gas bearings are used throughout.

The investigation also included two important portions of a closed Brayton cycle system: the recuperator (a gas-to-gas heat exchanger) and the cooler (a gas-to-liquid metal heat exchanger which interfaces the Brayton gas loop to the liquid metal loop of the radiator).

Parametric descriptions of the above were developed for use in the Brayton cycle optimization process. A baseline engine size of 300 MW received primary emphasis; a 10 MW engine for pilot plant use will also be investigated.

Thermionic Converters

These passive devices directly convert high temperature thermal energy into electricity. Thermionic diodes are mounted in the wall of the solar cavity absorber so that their electron emitters are exposed to the concentrated solar energy. The electron collectors with their associated cooling fin are exposed to space for heat dissipation. The voltage level of a series string of diodes must be kept low to avoid insulation breakdown. Rotary converters are used to step up the output to the level required by the transmitter.

Initially tungsten was selected as the electrode material; however, all known reserves of this material would be required for the baseline SPS program. A diode of nearly equal performance was later baselined using a molybdenum emitter and a molybdenum-coated nickel collector. Subcontractor for thermionics analyses was the Thermo Electron Corporation.

Cavity Absorbers

The cavity absorber is a hollow sphere with an opening to admit the concentrated solar energy from the concentrator. The tubular heat absorber assemblies for the Brayton cycle system or thermionic converters are mounted on the interior walls. The cavity is insulated to reduce energy loss through the walls. The high temperature insulation is composed of multiple thin layers of refractory metals separated by zirconia or yttria pellets. Reflection and reradiation losses through the aperture are functions of cavity geometry.

Nuclear Reactors

Dr. J. Richard Williams of the Georgia Institute of Technology was the consultant on reactors for this study. Breeder reactors were baselined because uranium resources will not last throughout the baselined program duration without them. Candidate reactor types were evaluated on the basis of their availabilitv by the time required, complexity of the fuel reprocess system, and achievable temperature with the required 30 year life (baselined for all satellite systems).

Power Distribution

Photovoltaic systems all use series chains to produce 20 000 V or more as required by the transmitter. Because the thermionic systems require rotary converters to step up the output which is limited to approximately 25 V, the thermionics and Brayton cycle can use alternating current distribution at high voltage. Photovoltaic systems will thus use slip rings at the rotary joint to the transmitter, while the other systems can use a rotary transformer. Nuclear power systems do not require a rotary joint because they are not Sun-facing. All systems require switchgear at various points in the distribution network.

Solar Cells

Two solar cell types were selected for emphasis: conventional silicon cells and gallium arsenide (GaAs) heterojunction cells. Performance in the 1990's was estimated by the projection of past improvement trends, taking fundamental limitations into account where required. A photovoltaic optimizer model assessed the impact of solar concentration, cell cooling, and cover glass thickness on total satellite cost. Silicon and GaAs systems were found to achieve a cost benefit from solar concentration and cooling. Solar concentration reduces the total cell "buy" and allows thicker cover glass with lower mass penalty. Cooling fins behind the cells reduce their temperature and raise cell efficiency.

SYSTEMS

The previously described subsystems were integrated by optimization models to select design points for the total systems. Several initial findings were:

1. The nuclear thermionic system was not technically feasible (with the molten salt breeder reactor) because radiator pumping power requirements exceeded the power output.

2. The nuclear Brayton system was most massive because the low molten salt breeder reactor temperatures resulted in a low temperature (hence massive) radiator.

3. The thermionic/Brayton cycle was lightest, but most complex.

4. The power relay system (geosynchronous mirrors) had a high probability of severe environmental impact.

System masses, costs, and launch pollutants were estimated for all systems.

PRELIMINARY OVERALL STUDY RESULTS

At this point in the study, the following primary conclusions can be drawn regarding power satellites:

1. At least three approaches to power generation in space appear to be practical.

2. These systems have masses of approximately 10^8 kg for 10 GW ground output capability.

3. In a significant program (at least 300 GW total capability), power costs (busbar) can be approximately 40 mill/kWh (1976 dollars).

APPENDIX C

APPLICATION OF STATION-KEPT ARRAY CONCEPTS TO SATELLITE SOLAR POWER STATION DESIGN[1]

INTRODUCTION

During FY 75 NASA Study 2.5 (Contract NASW-2727), The Aerospace Corporation conceived a technique for building large devices in orbit without requiring the large structures inherent in single satellite designs. In this concept the required large structure is divided into a number of physically separate segments, each individually station-kept and oriented and so controlled that the combined operation of all the segments, taken as whole, proposes to be indistinguishable from that of an equivalent single monolithic structure.

A company-funded detailed analytical investigation of the concept led to the premise that it has potential application to the design of large satellite solar power stations and served as a base for the present effort.

STUDY OBJECTIVE

The objective of the study is to assess the feasibility, practicality, and implications of segmenting the large satellite solar power station structural assemblies. Solar photovoltaic and solar powered Brayton systems are examined and power levels of 5 GW and 10 GW are considered.

STUDY APPROACH

Five generic concepts were identified for assembling large satellite solar power stations:

1. A rigid, monolithic structure in which bending, compression, tension, and thermal loads are transmitted throughout the structure from one section to another.

2. A structural array made up of individual segments joined by hinged or universal joints which are capable of transmitting tension on compression loads but not bending loads.

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[1] Synopsis — NASA 'MSFC Contract No. NAS8-31842.

3. A structural array made up of individual segments joined by flexible wires which transmit tension loads but are not capable of transmitting compression or bending loads.

4. A structural array made up of free-flying individual segments which are station-kept to maintain the array formation. Communication linkages exist rather than structural linkages.

5. A structural array made up of individual segments which are attached to each other by a cable which is maintained in tension by gravity gradient forces.

Six solar photovoltaic and four solar powered Brayton conceptual designs were prepared. The following system comparison parameters were identified and comparisons made using, in each case, the monolithic design as a baseline:

1. Assembly/deployment technique
2. On-orbit power transfer technique
3. System efficiency chain
4. Station keeping requirements
5. Guidance, stabilization, and control requirements
6. Microwave requirements
7. Power distribution requirements
8. Technology requirements
9. Dimensions of largest major assembly
   a. Solar array
   b. Microwave transmitting antenna
   c. Reflecting/refracting antenna
10. Weight of largest major assembly

    a. Solar array

    b. Microwave transmitting antenna

    c. Reflecting/refracting antenna

11. Total system weight

12. System cost

    a. Nonrecurring

    b. Recurring

13. Launch, initialization, and operations requirements.

STUDY CONCLUSIONS

It was concluded that it is feasible to consider segmenting the major structural assemblies of large satellite solar power stations, although several problem areas require more detailed analysis.

In particular, the gravity gradient stabilized system appears to have special merit. This concept separates the very large solar flux collection assemblies (either solar arrays or solar collectors) into individual segments which are attached to each other in a linear fashion by means of the power distribution cable. It weighs approximately the same as the comparable monolithic concept but alleviates two of the significant problem areas associated with the other concepts: (1) excessively large structural assemblies and (2) excessively large station keeping and attitude control propellant expenditures.

FUTURE PLANS

The gravity gradient stabilized satellite solar power station appears to be feasible and to have a number of attractive features. It is proposed therefore to conduct a further detailed study in which a number of potentially viable gravity gradient stabilized satellite solar power station configurations will be defined. Analytical methods will be developed to perform a detailed investigation of the orbital motions and shading characteristics of recommended systems. Critical subsystem problem areas, such as power distribution and management, will be identified and design solutions conceived and analyzed. An integrated development plan will be defined, leading to the implementation of a full-scale operational system.

APPENDIX D

FEASIBILITY STUDY OF THE SATELLITE POWER SYSTEM CONCEPT[1]

The Space Division of Rockwell International is conducting a study of satellite power systems (SPS) for NASA's Marshall Space Flight Center under Contract No. NAS8-31161, titled "Feasibility Study of the Satellite Power System Concept." Within the general guidelines of a mid-nineties IOC date with subsequent buildup of capability to provide up to 600 GW of power over the next 30 years, the principal objectives of the study are as follows:

1. To establish technical feasibility of the concept for acquiring solar energy in space, converting it to electricity, and transmitting it to a receiving antenna on Earth for distribution over conventional power lines.

2. To estimate costs associated with development, production, and operational emplacement to define the range of economic visibility for SPS.

3. To delineate systems hardware and operational processes which will require advancements in technology to ensure SPS program success.

The study was begun on August 1, 1976, and is being conducted over a 5-1/2 month period. The first 4 months are devoted to the technical effort with presentation of final results scheduled for the end of November. The final month and a half will be used for preparation, review, and publication of the final report.

Of the two solar energy conversion techniques, e.g., solar thermal and solar photovoltaic, the Rockwell study will investigate only the latter. To explore the potential of different systems hardware, design concepts, and operational approaches, Rockwell is analyzing the "reference" 8 GW photovoltaic configuration shown in Figure D-1. This high-aspect-ratio concept was chosen to yield reduced operational propellant requirements for attitude control, to increase structural stiffness, and to remove potentially interfering structural members from the path of the radiated power to Earth.

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[1] Synopsis — NASA/MSFC Contract No. NAS8-32161.

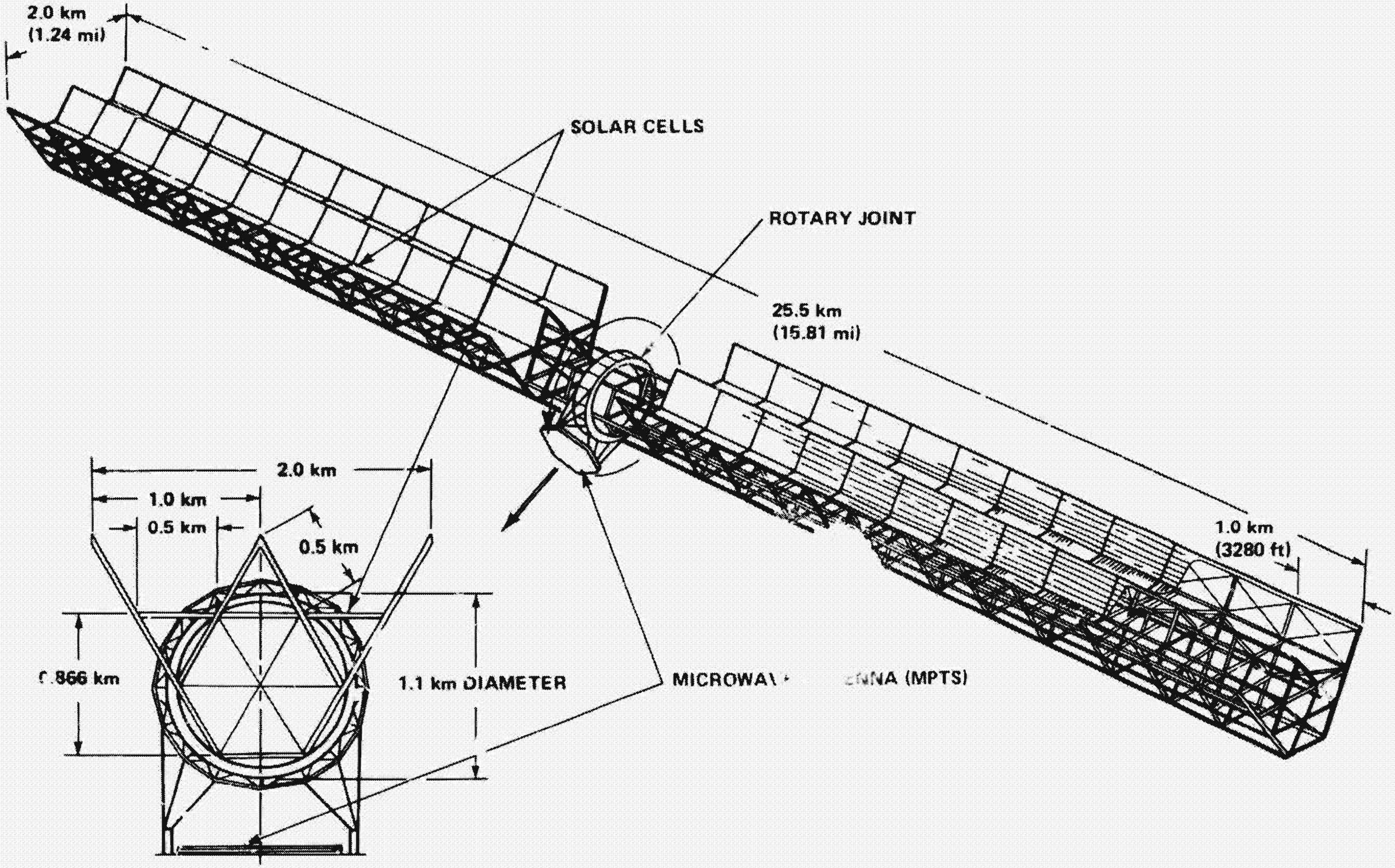

Figure D-1. Reference SPS configuration.

Analyses of the reference concept will be based on the use of gallium-aluminum-arsenide (GaAlAs) solar cells. In the time frame of interest, advances in GaAlAs cell technology and increased cell efficiencies may decrease blanket weights. The greater cell efficiency will permit a reduction in overall blanket area; however, the slightly greater weight of GaAlAs cells may not permit a significant cell blanket weight reduction. The reduced blanket area will require less supporting structure, resulting in reduced structural weight.

The reference microwave antenna structure being investigated is a hexagonal compression frame with a tension web. The dominant advantage of such a structure will be to minimize its on-orbit construction time. All construction operations for the SPS will take place at geosynchronous orbit, and the primary mode of transfer from low to geosynchronous orbit will be conventional chemical propulsion. Advantage will be taken of the previous NASA-contracted studies of future transportation systems and power transfer via microwave radiation.

The major s[illegible]asks are shown in Figure D-2, illustrating the degree of emphasis on analyses and definition of the reference satellite and on assembly operations — Tasks 1 and 2. The vehicles required for transport from Earth will be defined in Task 3, and capital cost estimates will be made in Task 4 together with the definition of significant technological advancements needed to successfully accomplish the program.

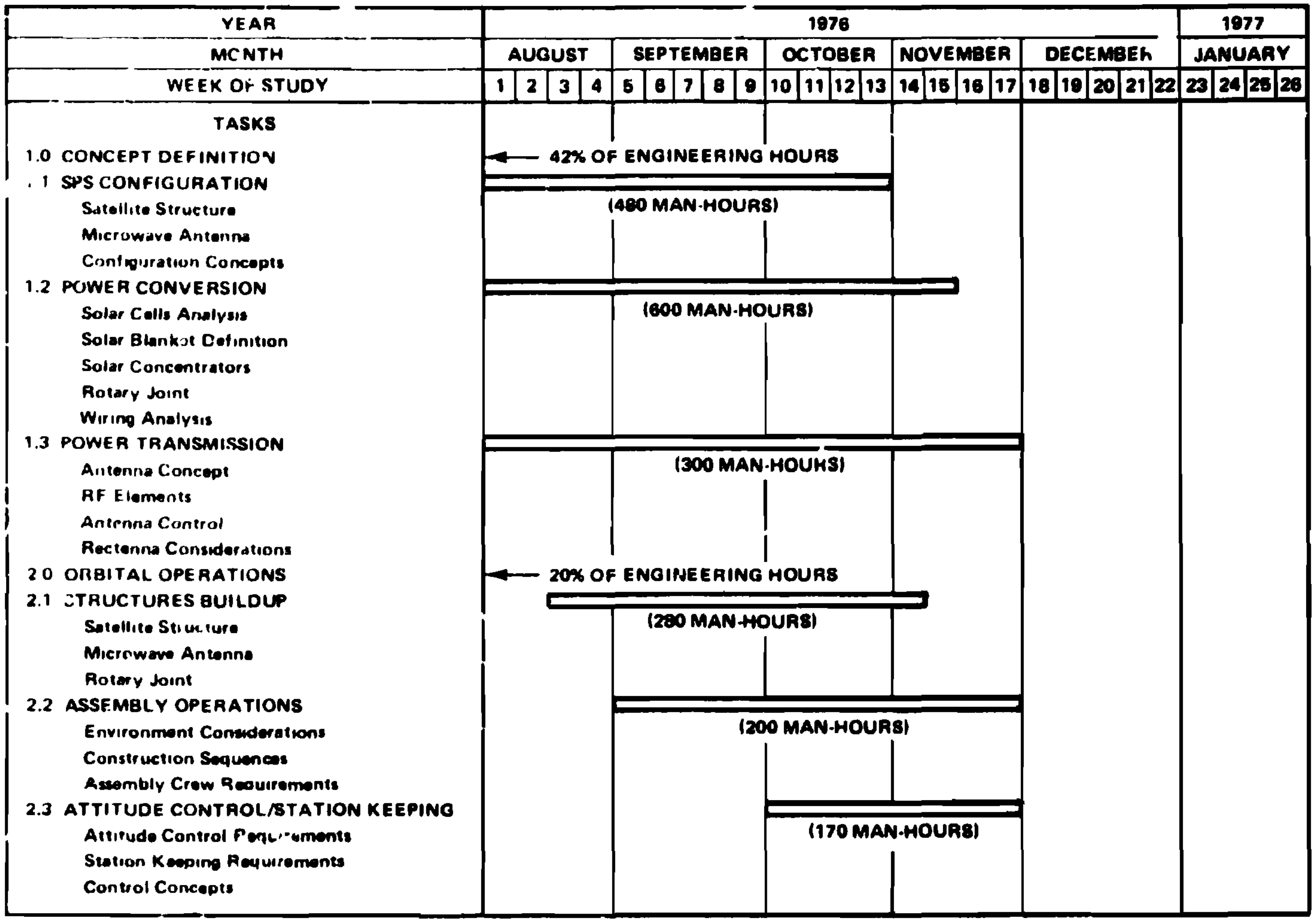

Figure D-2. Study schedule.

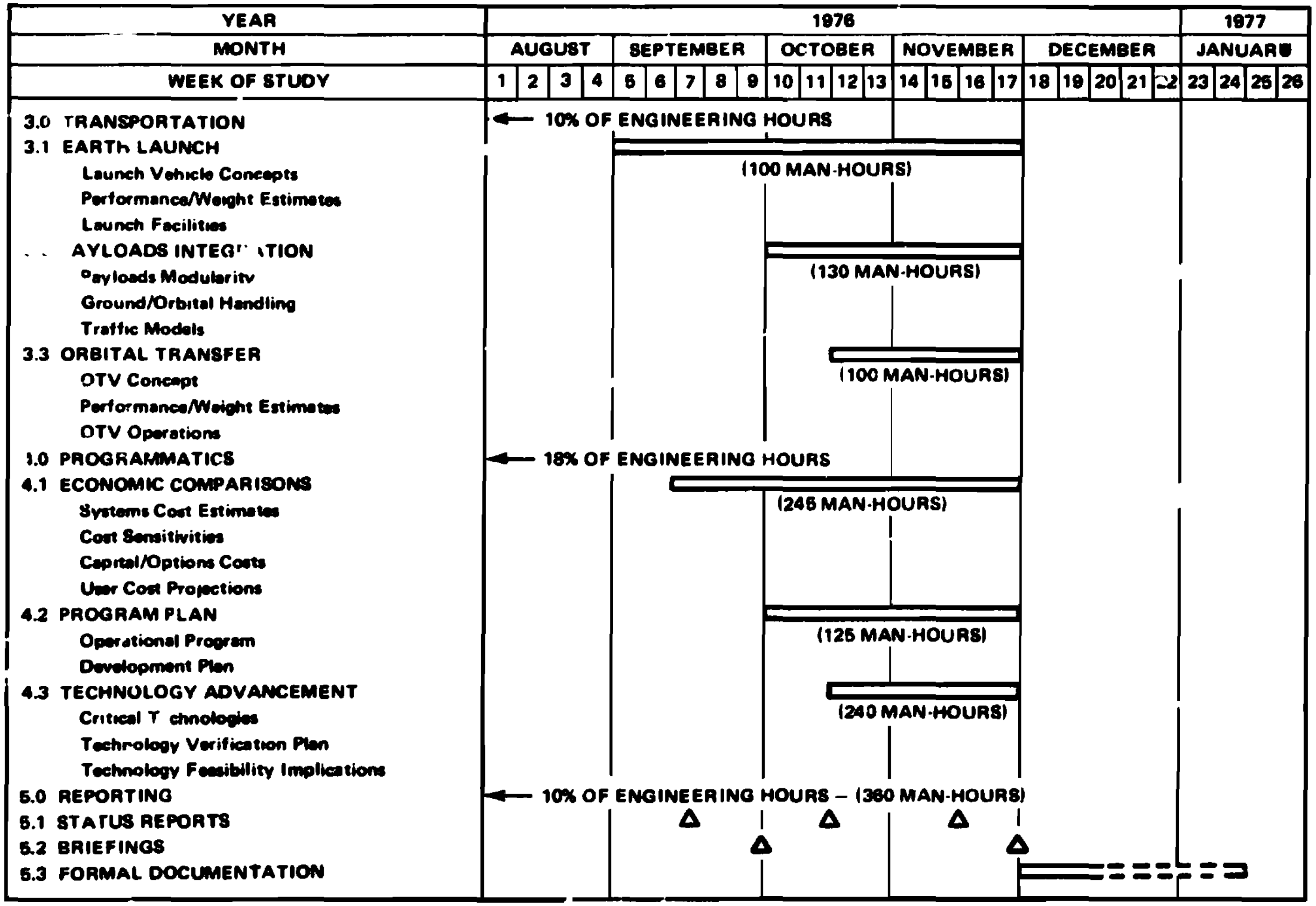

Figure D-2. (Concluded).

APPROVAL

SATELLITE POWER SYSTEMS
AN ENGINEERING AND ECONOMIC ANALYSIS SUMMARY

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

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