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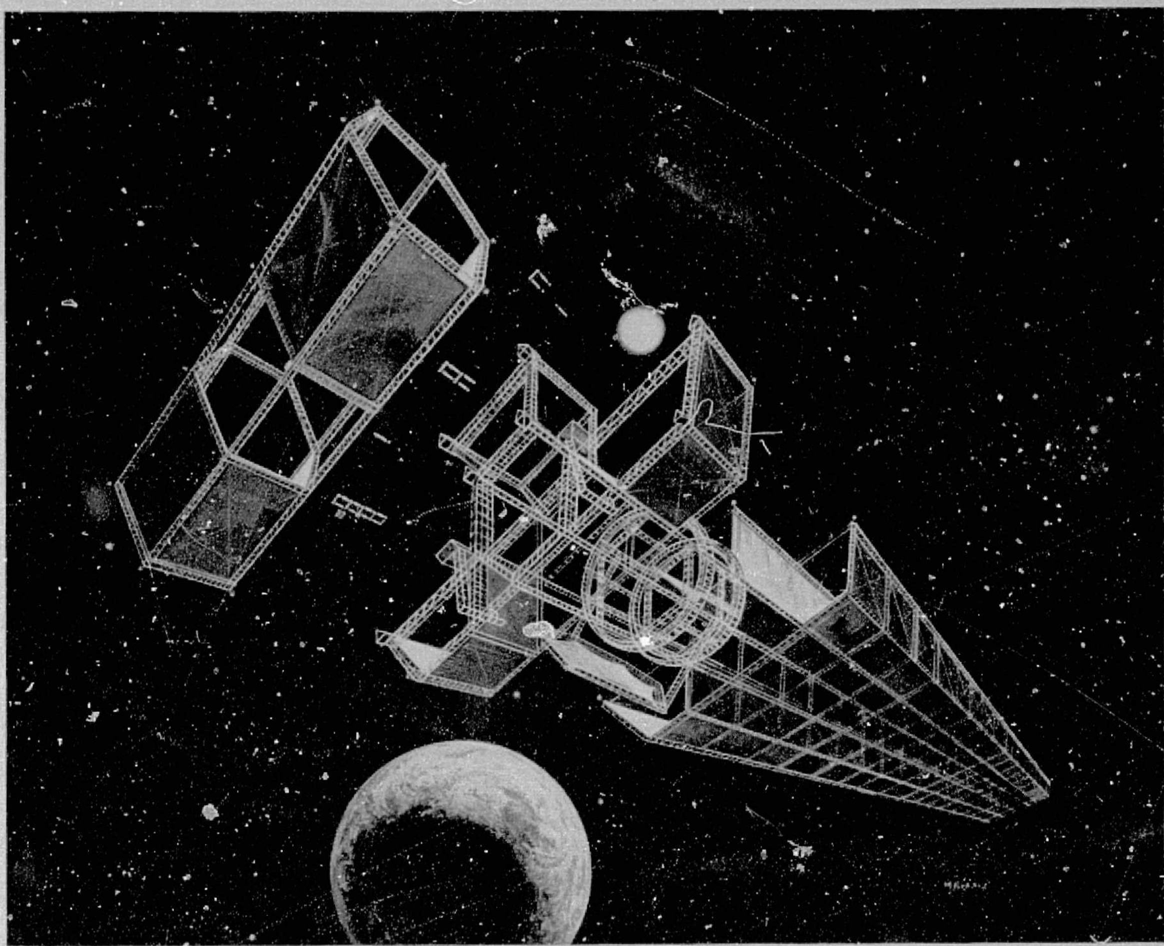

Satellite Power Systems (SPS) Concept Definition Study

FINAL REPORT

VOLUME I

EXECUTIVE SUMMARY

April 1978

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FOREWORD

This document summarizes the results of the *Satellite Power System Concept Definition Study* (Contract NAS8-32475), Exhibits A and B, and also incorporates results of NASA/MSFC in-house effort. Other volumes of this report that provide additional detail are:

- Volume II. SPS System Requirements
- Volume III. SPS Concept Evolution
- Volume IV. SPS Point Design Definition
- Volume V. Transportation and Operations Analysis
- Volume VI. SPS Technology Requirements and Verification
- Volume VII. SPS Program Plan and Economic Analysis

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INTRODUCTION

A projection of electrical energy demands over the next 30 to 50 years, coupled with reasonable assessments of known or developed energy sources, indicates that a critical incompatibility will occur about the turn of the century. It is rapidly becoming apparent that new, nondepletable energy sources and conversion/distribution systems must be defined and developed.

Recognizing this need, the Department of Energy (DOE) is currently evaluating candidate concepts. These include nuclear and solar sources. Fission nuclear sources require breeder reactors to assure a plentiful supply of fuel. Fusion nuclear sources would also supply virtually inexhaustible sources of energy, but require greatly advanced technology. Solar sources include direct solar-thermal or solar-photovoltaic conversion of solar energy to electrical power, or indirect sources such as wind, ocean-thermal, or the biomass.

The Satellite Power System (SPS) is a relatively recent approach that removes the energy collection system from the earth's surface to geosynchronous orbit. Several approaches have been considered for SPS, including solar-thermal or solar-photovoltaic conversion of solar energy to electrical power or use of a nuclear source. The SPS concept is illustrated in Figure 1 for a typical photovoltaic satellite approach. Solar energy is converted to

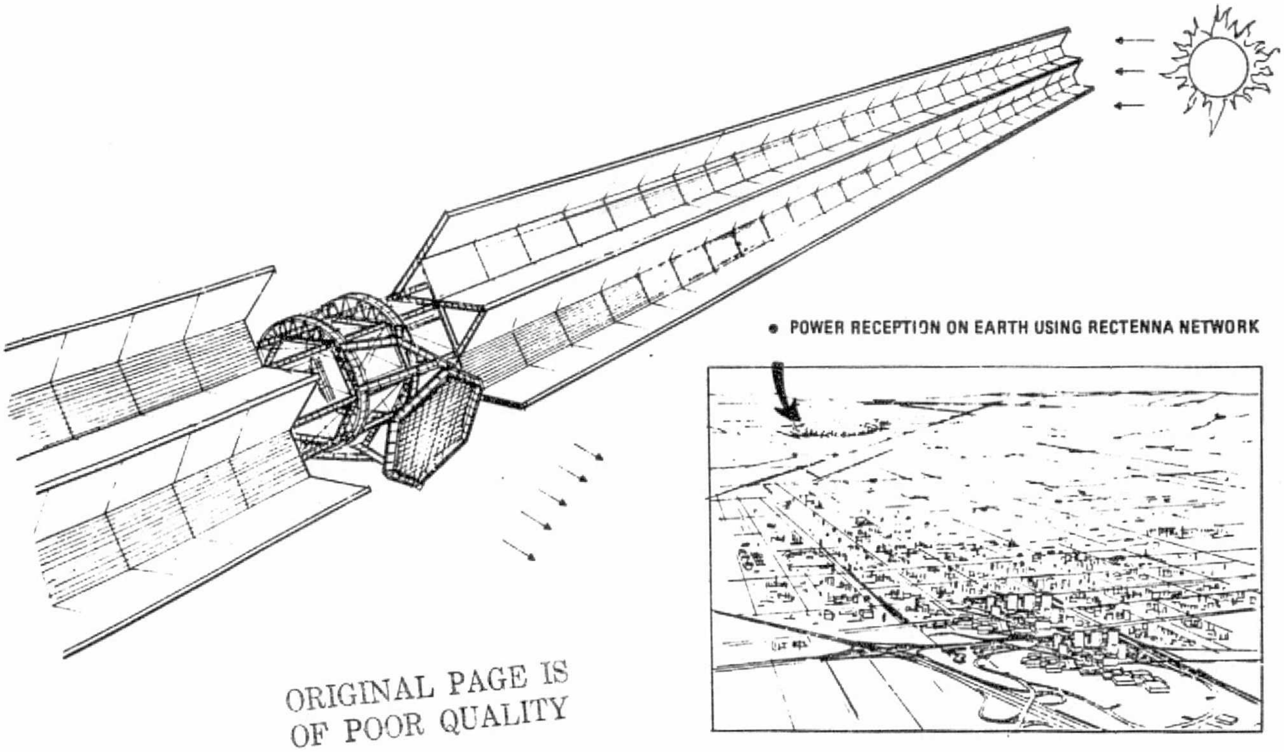

Figure 1. The Solar Satellite Power System Concept

electric energy using large solar arrays having reflectors that concentrate the energy on solar cells. The solar cells convert the solar energy to dc electrical energy which is conducted to a centrally located microwave antenna. The microwave antenna transforms the dc power to RF microwave energy and beams the energy to a receiving antenna (rectenna) on the ground. The rectenna converts the RF energy, at a very high efficiency, to dc electrical energy which is input to the utility system for distribution.

Typically, a single satellite system provides 5 GW of power to the utility interface on the ground. This amount of power is sufficient to meet the electrical needs of a large city such as Los Angeles or Chicago. Because a typical satellite has a solar collection area of about 75 km^2 and a mass of about 35-million kilograms, it is necessary to construct the satellite on orbit where the zero-gravity condition allows very low structural masses. The ground rectenna is nominally an elliptical array 10 km by 13 km. At the earth's surface, the microwave beam has a maximum intensity in the center of 23 mW/cm^2 (less than one-fourth the solar constant), and an intensity of less than 1 mW/cm^2 outside the rectenna fenceline (10 mW/cm^2 is the current U.S. exposure limit).

This SPS system study is a part of the total effort that will provide comparative concept evaluation data. The near-term purpose of this study, in cooperation with NASA-Marshall Space Flight Center effort, is to provide data that will allow selection of one or more preferred SPS concepts by the end of 1979. This process has already been initiated by a separate definition by NASA-MSFC and NASA-JSC of preliminary baseline concepts.[1]

This report volume summarizes the results of the SPS Concept Definition study. In the following sections, the overall study approach is described, significant results are summarized, and conclusions and recommendations resulting from the study are presented.

[1]Preliminary Baseline Concept Recommendations to DOE/NASA by the Marshall Space Flight Center, NASA-Headquarters, January 24, 1978.

STUDY APPROACH AND SCHEDULE

STUDY APPROACH
AND SCHEDULE

STUDY APPROACH AND SCHEDULE

Figures 2 and 3 present the approach and schedule for this study. The major output of the study was data to support NASA-MSFC recommendations to NASA-Headquarters and the Department of Energy of a preliminary baseline SPS concept.

The study was initiated by an analysis of the existing data base, comprised of NASA in-house effort and contracted studies. This initial effort resulted in an identification of key program issues and evaluation and synthesis of overall SPS system and subsystem options. A plan was prepared that specified the key issues, identified the data required to resolve the issues, and defined the steps and schedule for their resolution.

The system and subsystem concepts synthesized during this initial effort were subjected to analyses and trade studies to develop comparative data. These studies included an analysis of alternative satellite construction approaches to determine the preferred orbital locations for satellite construction.

The results of these analyses formed the basis for selection of point designs that were defined in greater detail during the last few months of the study. The point design data formed the basis for the preliminary baseline SPS concept recommendations.

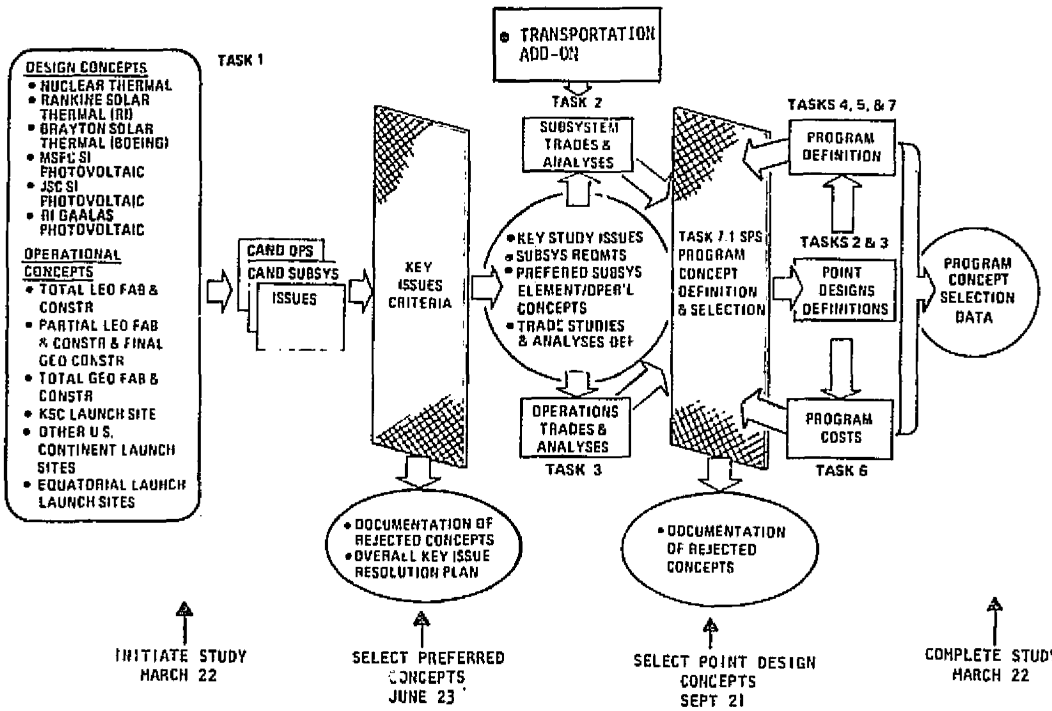

Figure 2. Study Logic

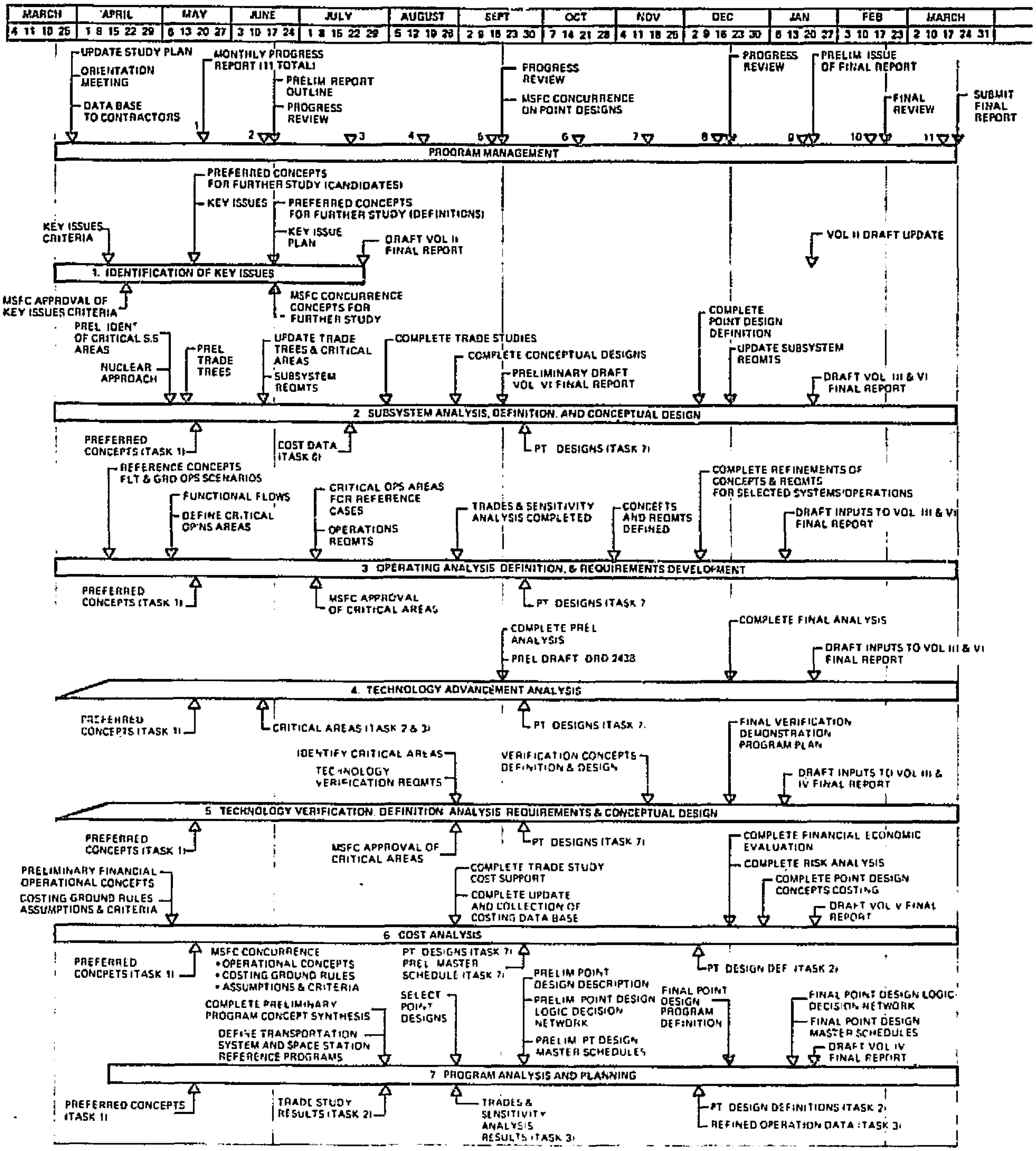

Figure 3. SPS Study Schedule

SIGNIFICANT RESULTS

SIGNIFICANT RESULTS

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SIGNIFICANT RESULTS

The most significant results of this contract are summarized in the following sections. These include SPS concepts evolution, point design definitions, end-to-end operations analysis, and programmatics.

SPS CONCEPTS EVOLUTION

Evolution of the total SPS Satellite Power System is described in this section. This includes a description of alternatives and trade study results for satellite concepts, ground receiving antenna (rectenna) concepts, satellite construction site concepts, and transportation concepts.

SATELLITE CONCEPTS

The data base existing at the beginning of the study was evaluated to determine which SPS approaches should be seriously considered as candidates for further analysis. These existing concepts are shown in Figure 4. Included were solar-photovoltaic concepts with silicon and gallium aluminum arsenide (GaAlAs) solar cells, a Brayton-cycle solar thermal concept, and a thermionic

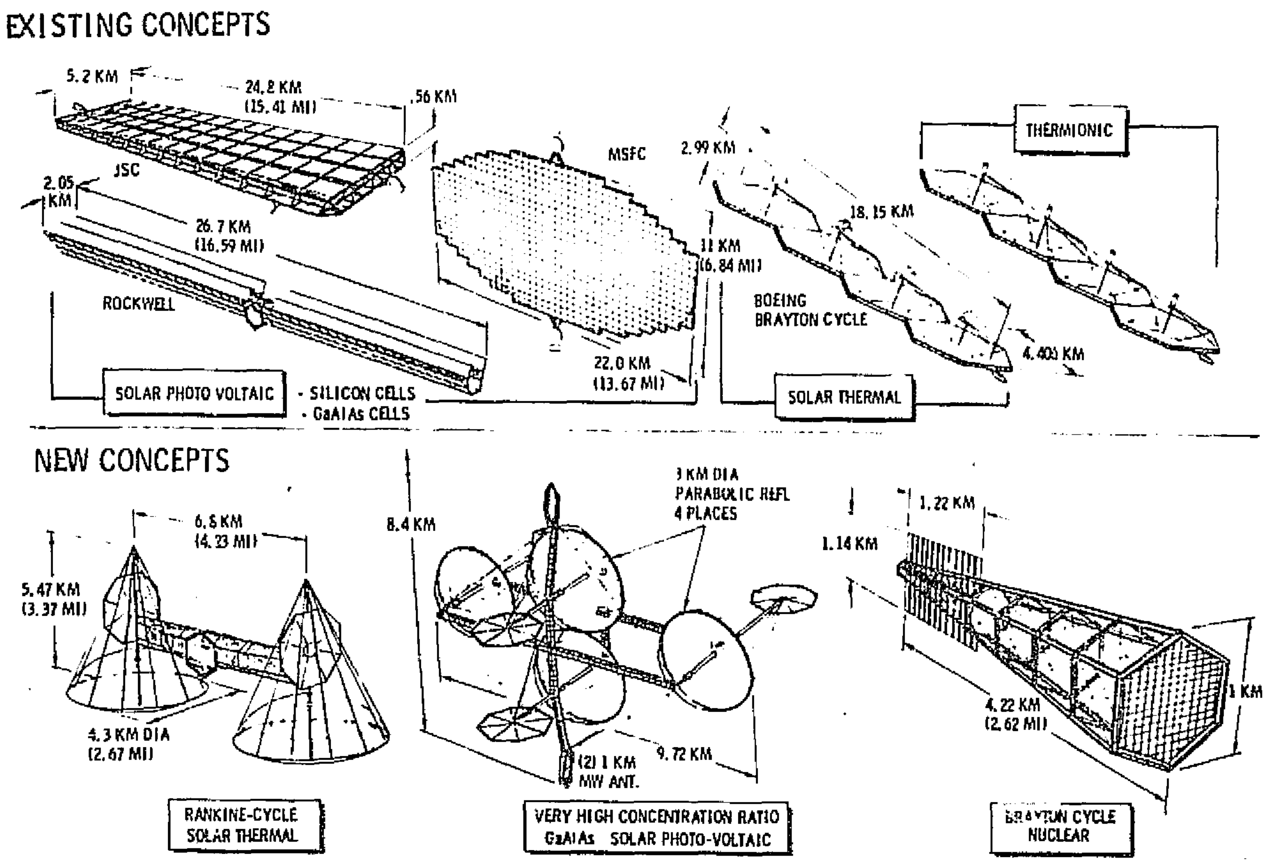


Figure 4. Initial Concepts Matrix

concept. In addition, new concepts (also shown in Figure 4) were introduced including a Rankine-cycle solar thermal concept, a very high concentration ratio (400:1) GaAlAs solar photovoltaic concept, and a nuclear concept using a Brayton cycle with a pebble-bed breeder reactor. As a result of these early evaluations, it was determined that the thermionic and the high concentration ratio GaAlAs concepts have much higher specific weights than the other concepts (e.g., 18 kg/kW for the thermionic concept and 13 kg/kW for the high concentration ratio GaAlAs concept, as compared to approximately 7 kg/kW for the low concentration ratio GaAlAs concepts). For this reason, the thermionic and high concentration ratio GaAlAs concepts were eliminated from further consideration.

The resulting matrix of concepts selected for additional analyses and trade studies is shown in Figure 5. Both silicon and GaAlAs solar blankets were considered for a concentration-ratio-of-1 solar photovoltaic concept. At higher concentration ratios, only GaAlAs blankets were considered because of the sharp drop-off in silicon cell performance at higher cell operating temperatures related to concentrator concepts.

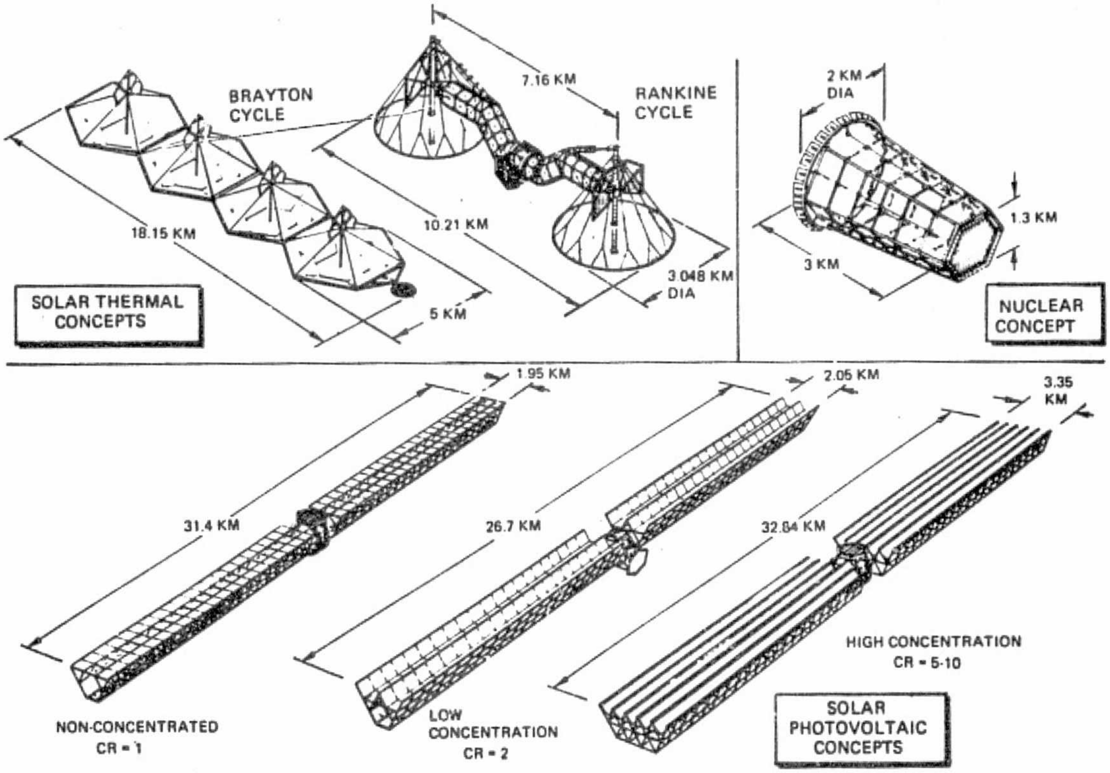


Figure 5. Concepts Selected for Trade Studies

Major subsystem alternatives and related trade study considerations are summarized in Table 1. The results of these studies are summarized in the following sections.

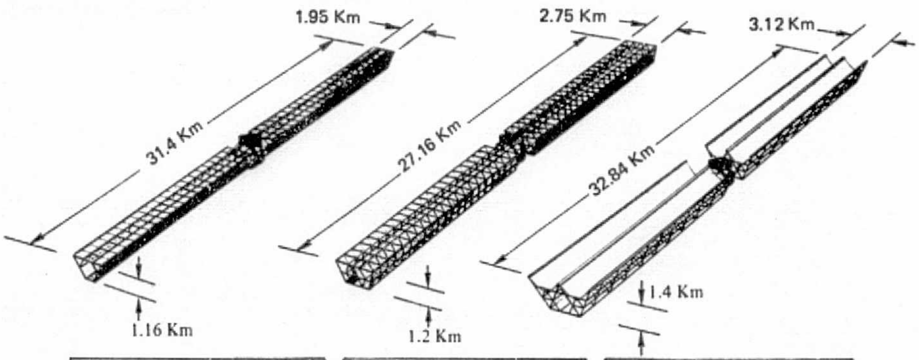
Solar Photovoltaic Satellites

Figure 6 summarizes the design and operational characteristics of the photovoltaic satellites resulting from the analyses and trade studies. All

Table 1. Major Subsystem Alternatives

SUBSYSTEM/ALTERNATIVES	CONSIDERATIONS
POWER CONVERSION	
SOLAR THERMAL	
CONCENTRATOR CONCEPTS: • MULTI-FACETED • INFLATABLE	WEIGHT AND FEASIBILITY
TURBINE INLET OPERATING TEMPERATURE LEVELS • 1389°C • 1040°C	WEIGHT AND TECHNOLOGY
HEAT REJECTION • NaK/HEAT PIPE • ORGANIC CONDENSING FLUID/HEAT PIPE • CONDENSING RADIATOR	WEIGHT AND FEASIBILITY
RANKINE BOTTOMING CYCLE	IMPROVED EFFICIENCY
SOLAR PHOTOVOLTAIC	
GaAs SOLAR CELL THICKNESSES • 1 TO 25 μ ACTIVE GaAs • 25-μ SINGLE CRYSTAL SUBSTRATE-- SAPPHIRE VS. GaAs SUBSTRATE	COST, AVAILABILITY AND EFFICIENCY
GaAs CONCENTRATION RATIO: 1 TO 10	COST, AVAILABILITY, COMPLEXITY, AND WEIGHT
MICROWAVE POWER/RECTENNA	
PREFERRED CONVERSION DEVICES • AMPLITRONS • KLYSTRONS • SOLID STATE	EFFICIENCY & WEIGHT (INCL. POWER DISTRIB.), TECHNOLOGY, & RELIABILITY/MAINTENANCE
IONOSPHERIC POWER DENSITY • PENCIL BEAM (GAUSSIAN) • SHAPED BEAM (BESSEL J)	D&F LAYER HEATING, RECTENNA SIZE, SIDE-LOBE EFFECTS, AND RFI
ANTENNA SIZE AND NUMBER PER SPS	RECTENNA SIZE AND NUMBER OF RECTENNAS
WAVEGUIDE	EFFICIENCY AND WEIGHT
POWER DISTRIBUTION	
PHOTOVOLTAIC LENGTH TO WIDTH • 2 TO 13	POWER DISTRIBUTION WEIGHT RELATED TO ATTITUDE CONTROL & STRUCTURE
REFLECTORS AS NEGATIVE BUSES	POWER DISTRIBUTION WEIGHT & FEASIBILITY
VOLTAGE LEVEL: 100 V TO 40,000 V	PWR DISTRIB WT; RELATED TO MW POWER TRADES
STRUCTURES	
PHOTOVOLTAIC LENGTH TO WIDTH • 2 TO 13	STRUCTURE, POWER DISTRIBUTION, AND ATTITUDE CONTROL WEIGHTS
PHOTOVOLTAIC LENGTH TO DEPTH	OPTIMIZATION OF STRUCT WT & NATURAL FREQ.
NATURAL FREQUENCY REQUIREMENTS • LEO VS. GEO CONSTRUCTION	POTENTIAL LARGE DIFFERENCE IN REQUIREMENTS
MATERIALS: METALLIC, ALKALI, & COMPOSITE	WEIGHT, STIFFNESS, FABRICATION, & ASSEMBLY
MICROWAVE ANTENNA STRUCTURE • COMPRESSION FRAME • BOX GRID	WT., FABRIC ASSY SIMPLICITY & FRAME CONTROL
ATTITUDE CONTROL/STATIONKEEPING	
PHOTOVOLTAIC CR-2 TO -5 • FIG. CONTROL VS. POINTING ACCURACY	MINIMUM WEIGHT AND COMPLEXITY
ATTITUDE CONTROL APPROACH • FREE DRIFT VS. ACTIVE PITCH CONTROL	MINIMUM WEIGHT AND COMPLEXITY
ATTITUDE CONTROL IMPLEMENTATION • INERTIA WHEELS • RCS	MINIMUM WEIGHT AND COMPLEXITY
STATIONKEEPING • REFERENCE PLANE • GEO	MINIMUM WEIGHT

of these concepts are sized for 5-GW power output at the utility interface on the ground. All of the satellites are assumed to be constructed in geosynchronous orbit.



CELL SELECTION	GaAlAs (Si AIT)	GaAlAs	GaAlAs
SELF-ANNEALING	NO	YES	YES
BAND PASS FILTER	NO	NO	YES
POINTING REQUIRE (ROLL)	±2°	± 0.10	± 0.025
FLIGHT PLANE ORIENT	EQUATORIAL	EQUATORIAL	EQUATORIAL
FLIGHT ATTITUDE	Y-POP, X-IOP	Y-POP, X-IOP	Y-POP, X-IOP
ATTITUDE CONTROL TYPE	ELECTRIC THRUSTERS	ELECTRIC THRUSTERS	ELECTRIC THRUSTERS
LENGTH/WIDTH	16.1	9.9	10.5
LENGTH/DEPTH	27.0	22.6	23.5
SYST SP WT (KG/KW)	7.35	6.65	8.15

Figure 6. Solar Photovoltaic System Characteristics (Geosynchronous Orbit Construction)

Energy Conversion. At a concentration ratio of 1, either GaAlAs or silicon solar cells may be used. At higher concentration ratios, the operating temperature of the solar cells increase, and silicon cells do not appear to be advantageous at concentration ratios of 2 and 5 because their efficiency is greatly reduced at elevated temperatures.

GaAlAs cells have several advantages when compared to silicon cells. These advantages are summarized in Figure 7. As shown, the GaAlAs cells have a higher efficiency than silicon cells at much lower cell thicknesses (e.g., 22 percent at 1 μm for GaAlAs as compared to 20 percent at about 30 μm for silicon). This results in a much lower blanket weight. In addition, based on recent test data, it has been determined that GaAlAs solar cells are resistant to natural space radiation damage and are virtually self-annealing (tend to return to their original cell efficiency) at cell temperatures of about 125°C. Silicon cells are prone to greater damage from electrons, and annealing of the cells to approximate their original efficiency requires annealing temperatures of greater than 500°C. This indicates an important difference between these two cells. At concentration ratios of 2 and above, GaAlAs cells will normally operate at temperatures in excess of 125°C, thus providing potential for continuous self-annealing of radiation damage. Furthermore, it is desirable to have higher concentration ratios on GaAlAs cells to reduce costs because their efficiency is not degraded seriously and the cost of 1/2-mil aluminized kapton reflectors is considerably less per unit area than solar cells.

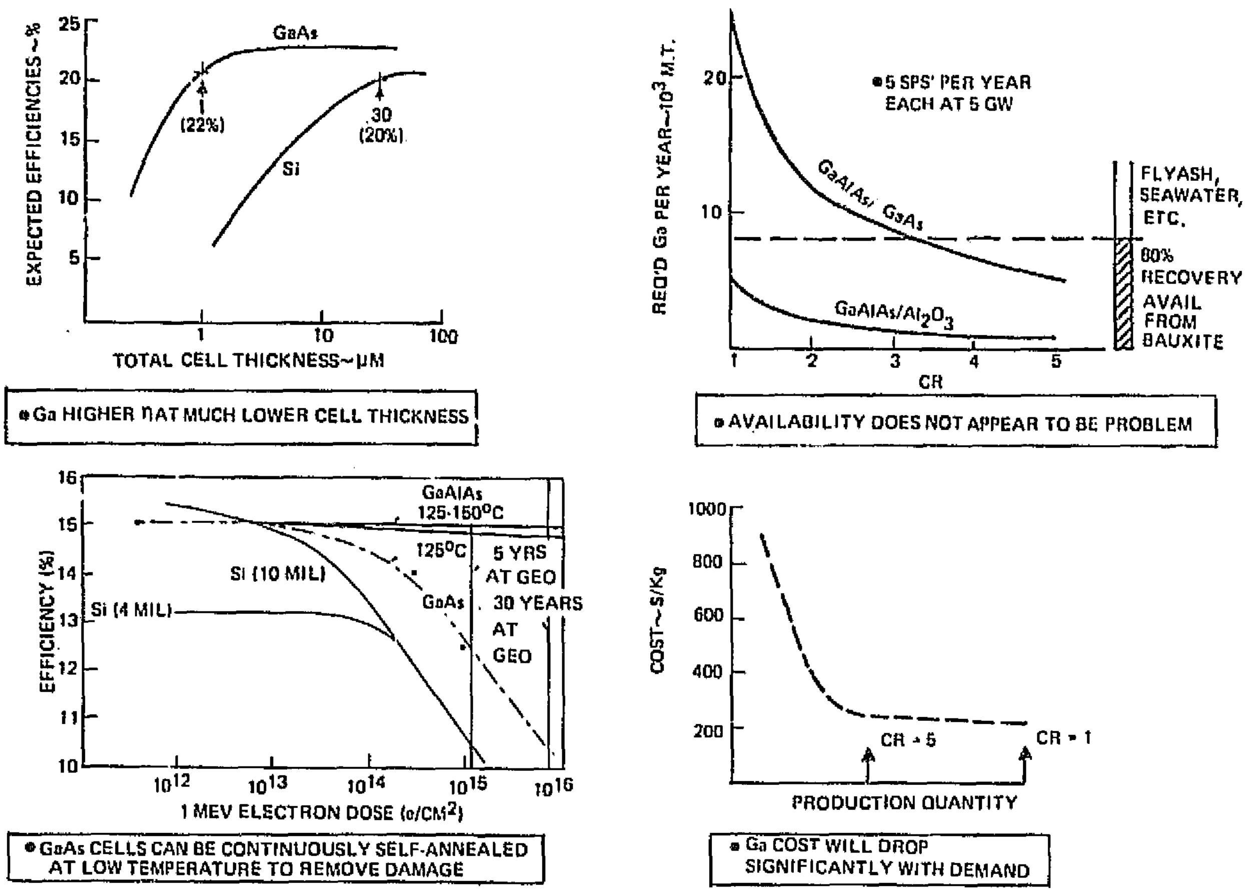


Figure 7. Advantages of GaAlAs Solar Cells

Two areas of concern for GaAlAs cells are the availability of gallium in sufficient quantities for SPS and the cost of gallium. Current GaAlAs cells utilize GaAs as the substrate material. As shown in Figure 7, at a CR = 3, it is expected that sufficient gallium can be recovered from available bauxite, used for aluminum production, to produce five satellites per year. Although data are not yet available on substitute substrate material, it is expected that a synthetic sapphire (Al_2O_3) substrate material will still provide good cell efficiency while reducing the required gallium by a factor of 5. As shown (Figure 7), sufficient gallium should be available at any concentration ratio for this cell concept. Gallium, currently produced in small quantities, costs about $900 per kilogram. It is expected that this price will decrease significantly when the large quantities related to SPS are produced.

Bandpass filters may be used on the reflectors to reduce incident radiation to the solar cells in regions of the solar spectrum where they are ineffective and, therefore, to reduce cell operating temperature. As a result of the trade studies, it was concluded that GaAlAs cell temperatures should be maintained at about 125°C to assure self-annealing from radiation damage. At

CR = 2, the cells will reach an equilibrium temperature of approximately 125°C without bandpass filters. For this reason, bandpass filters are not recommended at CR = 2. At CR = 5, it is desirable to have bandpass filters to reduce cell temperatures and increase efficiency. Even with bandpass filters, the temperature will reach approximately 180°C.

Selection of Flight Conditions. Selection of flight conditions at geosynchronous altitude encompassed the selection of inclination and vehicle attitude.

Orbit planes considered for the SPS, shown in Figure 8, include the ecliptic, equatorial, and a 7.3-degree inclined orbit which eliminates the north-south stationkeeping ΔV requirement. Collector pointing, always normal to the sun, can be achieved in the ecliptic plane with a "wings level" (Y-POP) attitude which minimizes the gravity gradient induced RCS propellant consumption. This orbit results in a daily earth eclipse period. The equatorial plane produces photovoltaic collector losses up to 8 percent for the wings-level attitude. Trade study results indicate that the equatorial orbit is preferred, primarily due to the dominant cost impacts of the increased rectenna sizes for the ecliptic and 7.3-degree inclined orbits, and the daily eclipse in the ecliptic orbit.

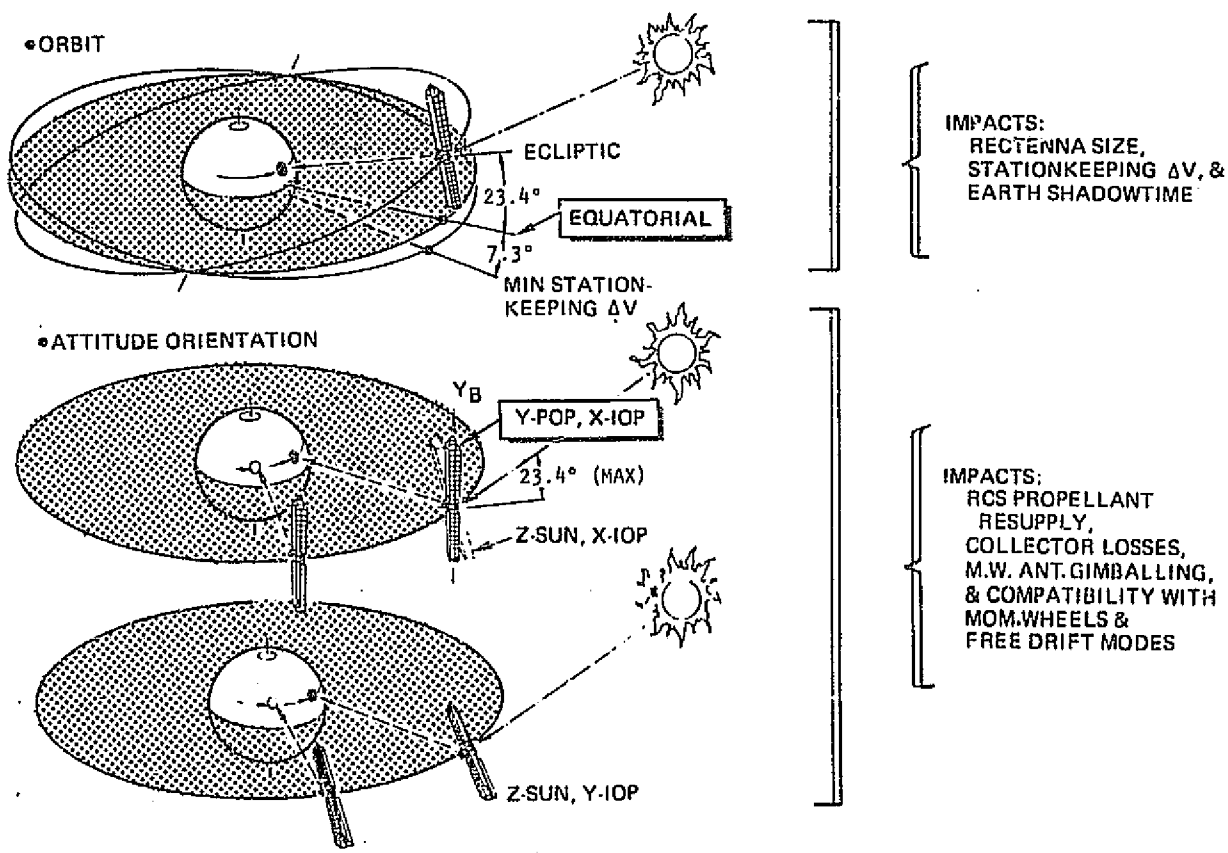


Figure 8. Selection of Flight Conditions

Three attitude orientations (illustrated in Figure 8) have been evaluated in the ACSS trade studies. The wings-level attitude (Y perpendicular to orbit plane, X in orbit plane) is preferred over the Z-sun, X-IOP attitude due to the large savings in RCS propellant resupply. The X-sun, X-IOP attitude places the long axis of the vehicle in the orbit plane. This orientation requires two microwave antennas in order to simultaneously prevent interference of the microwave beam with the spacecraft structure and obtain acceptable solar pressure torque balancing. Two antennas result in double-sizing of the spacecraft, leading to additional complexity and power distribution mass penalties. On this basis the Y-POP, X-IOP is the preferred orientation.

Solar Thermal Concepts

Two basically different cycle approaches were considered for the solar thermal system—the Brayton cycle and the Rankine cycle. The primary difference between them is that the Brayton cycle occurs entirely in the gas phase, and the Rankine cycle occurs in gas and liquid phases. The Brayton cycle utilizes helium as the working fluid. Both cesium and potassium were considered as primary working fluids for the Rankine cycle, and a steam bottoming cycle was also studied to assess overall performance advantages.

Figure 9 summarizes areas of major trade studies for the concentrator, power conversion subsystem, and thermal control system (radiators). The results are discussed below.

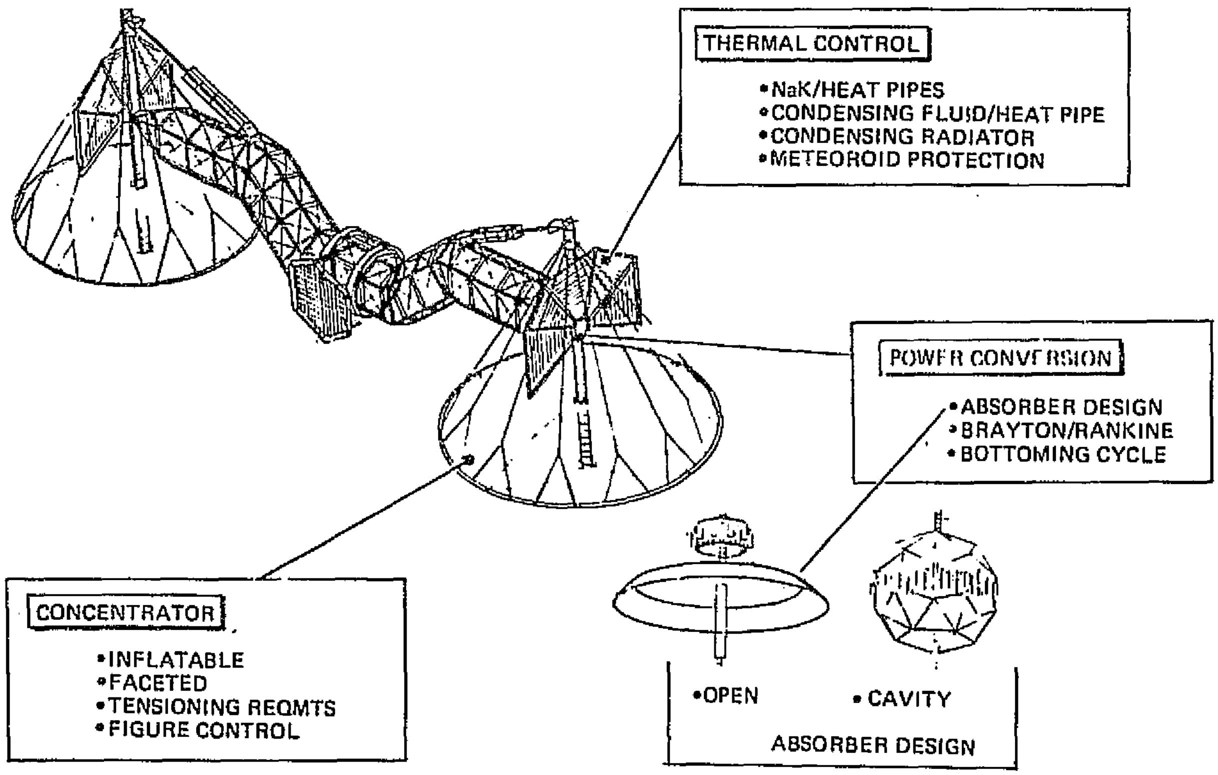
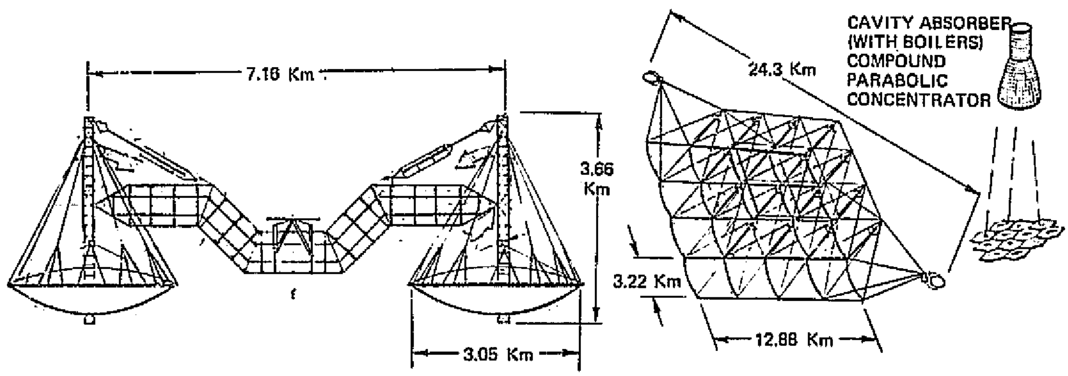


Figure 9. Solar Thermal Major Trade Areas

Concentrator. Two approaches have been considered for the concentrator: a multi-faceted approach that uses many flat surfaces to concentrate sunlight and an inflatable approach that uses large pressure-stabilized parabolic reflectors to concentrate sunlight. Concepts using these two approaches are illustrated in Figure 10. The inflatable approach was devised by Rockwell, and the multi-faceted approach was introduced by Boeing. In both cases, the need to maintain the concept in a wings-level attitude, because of gravity-gradient impacts on attitude control, results in a need to steer the reflectors to a normal incidence to the sun by articulation of the reflectors. Alternatively, the satellites could be operated in an ecliptic-plane 24-hour orbit to avoid articulation of the reflectors. However, the previously cited disadvantages of this orbit (enlarged rectenna and daily solar eclipse) favor articulation of the reflectors. As shown in Figure 10, the two large pressurized reflectors on the Rockwell concept are rotated along with their absorbers and radiators by an electric actuator. The rotational rate is very slow (93.6 degrees per year).



	BRAYTON	RANKINE	RANKINE	RANKINE
WORKING FLUID	HELIUM/XENON	CESIUM	CESIUM/STEAM	POTASSIUM
AREA (REFLECTOR) KM^2	24	30.32	23	30.32
η OA %	15	12	16	12
CYCLE η%	45	36	47	36
TURBINE INLET TEMP $^\circ$C	1379	1038	1038	1038
GENERATOR SIZE MW	30	30	30	30
SAT. ORIENT	Y–POP, X–IOP	Y–POP, X–IOP	Y–POP, X–IOP	Y–POP, X–IOP
RADIATOR AREA KM^2	2.35	1.35	2.2	1.35
SYSTEM SP. WT. KG/KW	8.65	8.35	6.9	11.0

Figure 10. Solar Thermal System Characteristics

Each of the facets on the multi-faceted concept must be individually moved to track the sun. The major concern with the pressurized concept is the reflector tension needed to maintain good reflective characteristics. Tests conducted during the study on small samples indicated that 300 psi is adequate. The gas required to pressurize the reflectors is less than one percent of the satellite mass over a 30-year operational period, including the losses due to meteoroid punctures.

Thermal Cycles. Figure 10 compares the characteristics of the thermal cycles. One of the major differentiators between the Brayton and Rankine cycles is the turbine inlet temperature. In order to be competitive in terms of system specific weight (kg/kW), it is necessary to run at a high turbine inlet temperature with a Brayton cycle (1379°C) when compared to the Rankine cycle (1038°C). The result of this higher temperature is a more advanced technology requirement for the Brayton cycle (use of ceramics in the turbomachinery). Although the overall efficiency is lower for the cesium Rankine cycle without steam bottoming (12 percent) than for the Brayton cycle (15 percent), the system specific weight is lower due to lower-weight turbomachinery and radiators. As shown, the cesium Rankine cycle with steam bottoming has a significantly improved overall efficiency (16 percent) and the lowest specific weight (6.9 kg/kW).

Both the cesium-Rankine and the helium-Brayton concepts utilize a condensing steam radiator with a heat exchanger between the water and the working fluid. A condensing steam radiator was also used with the cesium Rankine having a steam bottoming cycle.

Nuclear Concept

Table 2 lists the types of reactors considered for this study and potentially suitable energy conversion schemes. The criteria used for rejection are listed at the bottom of the table. Due to the 1991 availability requirement, a selection was required from reactors listed as "intermediate term." Three fission reactor systems are considered as long-range possibilities. These are the various high-temperature plasma core reactors known as the transparent partition (or light bulb) plasma core reactor, the open-type plasma core reactor, and the multi-phase reactor. However, the plasma core reactors require substantial development before they can be put into operation. The multi-phase reactor has a relatively stagnant core of gaseous uranium and the heat is removed by conduction through the walls of the reactor.

Table 2. Nuclear Reactor Options

	REACTOR TYPE	COSED BRAYTON CYCLE	POTASSIUM RANKINE	LIQUID-METAL MHD	PLASMA MHD	THERMIONIC
NEAR TERM						
FP	LIQUID-METAL FAST BREEDER		X	X		
FP	GAS-COOLED FAST REACTOR	X	X	X		
R	MOLTEN SALT BREEDER REACTOR		X	X		X
INTERMEDIATE TERM						
NB	CERAMIC THERMIONIC REACTOR					X
NB	NERVA-TYPE REACTOR	X			X	
C	ROTATING FLUIDIZED BED	X			X	
T	COLLOID CORE REACTOR	X			X	
C	UF_6 FUELED REACTOR	X			X	
✓	CERAMIC PEBBLE-BED REACTOR	X			X	X
LONG TERM						
T	"LIGHT BULB" PLASMA CORE				X	
T	"COAXIAL FLOW" PLASMA CORE				X	
T	MULTI-PHASE REACTOR		X	X		X
T	FUSION REACTOR				X	

NOTES: NB = NON BREEDER; T = LOW TECH STATUS; FP = COMPLEX FUEL PROCESSING; R = PAST STUDY REJECTION; ✓ = SELECTED; C = ADDED COMPLEXITY

The nuclear power system selected was based on meeting the overall system requirements with a nuclear reactor concept that is reasonably achievable within the schedule and cost of SPS. A pebble-bed fast breeder reactor cooled by helium gas was selected. This reactor uses plutonium fuel and U-238 breeding material. The fuel elements, reprocessed in a space recycle process plant, are 1/4-inch spheres consisting of a mixed oxide of uranium and plutonium (UO_2/PUO_2) and are carbon-coated to prevent the escape of fission products. A Brayton power conversion unit uses He from the reactor as the working fluid for production of electric power. Waste heat is rejected using pumped NaK through a series of manifolds and heat pipe radiator elements.

The system will provide 5 GWe on the ground, thereby requiring 7.86 GWe at the power conversion system terminals. The total power system consists of twenty-six 0.3-GWe power modules. Each module is comprised of a reactor, Brayton PCU, fuel processing plant, and space radiator heat rejection system.

As shown in Figure 11, the spent fuel recycled out of the reactor is reprocessed by either an Airox or a Modified Purex process, depending on the amount of fission product poisons which must be removed from the fuel prior to refabrication. The Airox process uses a series of oxidation-reduction heating cycles which cause the fuel to swell and crack, thereby releasing only the gaseous fission products. The final reduction reaction leaves the fuel in a powdered form which still contains the remaining fission product solids and inerts. The mass proceeds to Refabrication where it is mixed with resupply UO_2, formed into carbon-coated pellets, and recycled back to the reactor.

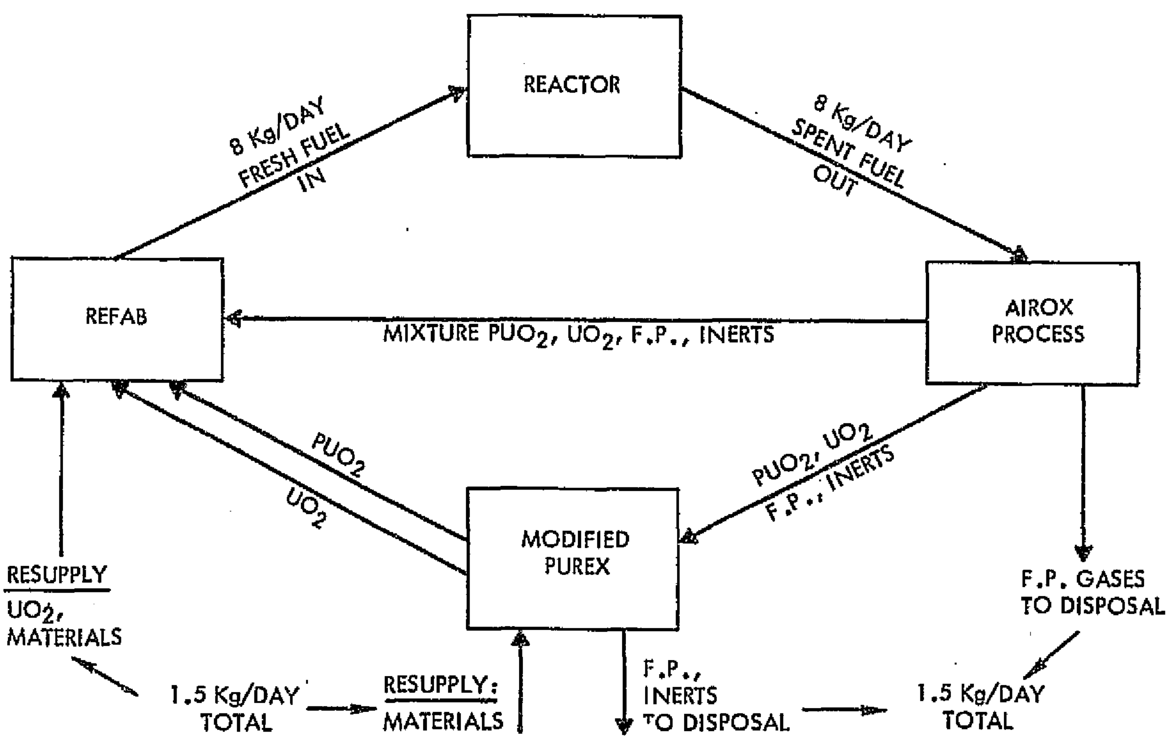


Figure 11. Fuel Management Schematic

Depending upon the buildup of fission product poisons and inerts in the reactor core (which would lower the reactivity and power density), the fuel mixture leaving the Airox process may also be treated in the Modified Purex process. The latter consists of a series of counterflow solvent extraction columns. The fuel mixture is first dissolved in aqueous HNO_3 and the fission products, inerts, and higher oxides of Pu and U are separated out successively with a counterflow organic steam of tri-butyle phosphate. The resultant separated PUO_2 and UO_2 are then sent to Refabrication.

The Refabrication process mixes the processed fuel from either pretreatment process with resupply UO_2, and forms pellets which are sintered and then carbon coated. The resulting pellets are then returned to the reactor.

The overall nuclear SPS concept and the approach to maintenance is shown in Figure 12. Twenty-six power modules are located in a ring on the left side of the structure which separates the power modules by 3 km from the microwave antenna to provide an adequately low radiation environment.

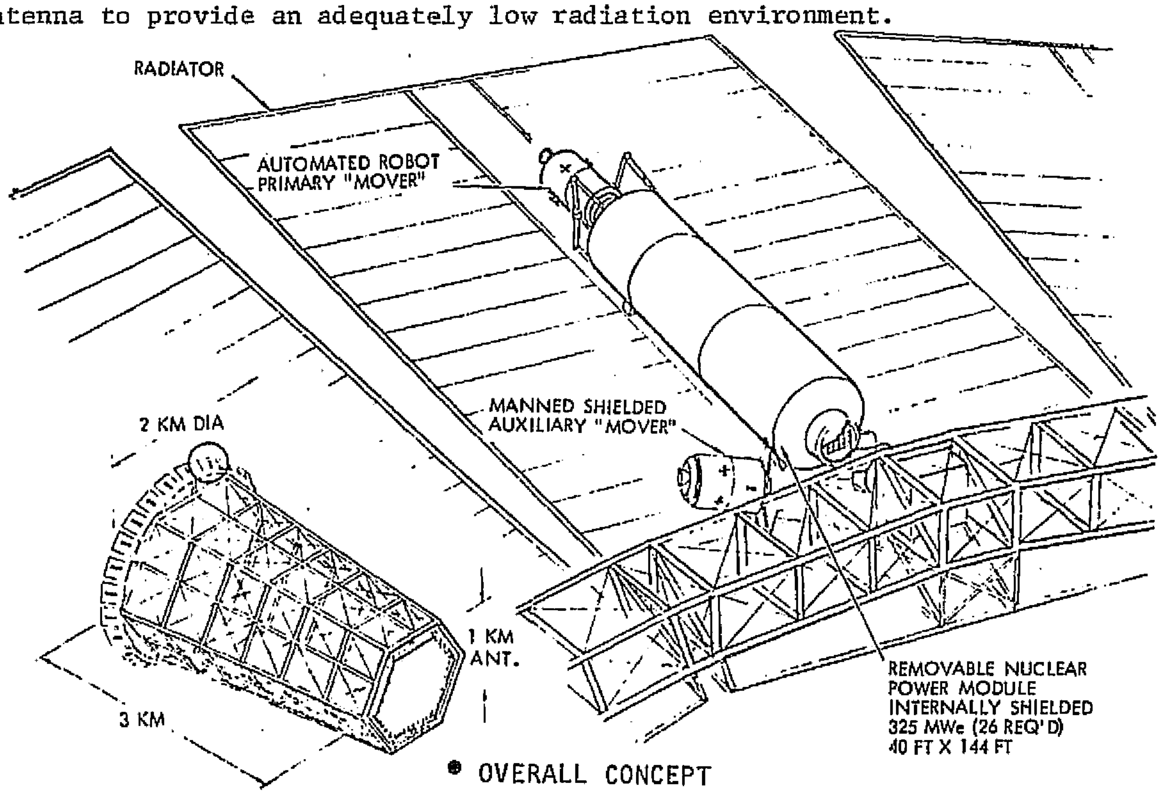


Figure 12. Nuclear SPS Concept

When being maintained, the radiator remains in place with the SPS, coolant lines being disconnected from the power module by remotely operated actuators. Self-propelled "movers" remove the power module from the SPS. One mover shown as manned requires substantial radiation shielding due to fission fragment radiation from the shut-down reactor and fission radiation from adjacent operating reactors. The shut-down power module is removed to an orbiting hot laboratory for maintenance operations.

Microwave Antenna

The microwave antenna design is virtually independent of overall satellite design; i.e., it is basically the same for photovoltaic, solar-thermal, and nuclear concepts. Its basic function is to transform electrical energy from the power source into microwave radio frequency energy and to efficiently propagate the microwave beam to a receiving antenna (rectenna) located on the ground.

Devices that were considered for dc-to-RF conversion included amplitrons, klystrons, and transistors. Of these, only the amplitrons have been rejected on the basis of their low efficiency and high driver complexity. The klystron, operating at a frequency of 2.45 GHz, has an efficiency of 85 percent and a power level of 50 kW. This device is the current baseline. Solid-state transistors are also being considered, but additional effort is required to define their impact on the SPS design. It may be necessary to operate these devices at 0.915 GHz and with a larger-diameter antenna in order to simultaneously satisfy thermal operating constraints of the transistors and the maximum microwave density (23 mW/cm^2) dictated by atmospheric interactions.

The microwave transmission system is the key element of the SPS. Several constraints must be placed on the design of this system in order to satisfy potential environmental concerns. Major concerns include: excessive heating of the D and F layers of the atmosphere, biological effects of microwave radiation outside of the controlled rectenna area, interference with other radio-frequency receivers and sources, and land requirements for the rectenna. Atmospheric heating effects cause ionization of the D and F layers which, in turn, causes interference with certain radio-frequency transmission bands. Since insufficient test data are available to set a definite upper limit, a preliminary conservative limit of 23 mW/cm^2 has been specified. Biological considerations have resulted in a current limit of 1 mW/cm^2 outside of the controlled rectenna area. The problem of interference with other radio-frequency receivers and sources has not yet been adequately defined. These investigations, along with other environmental concerns, are a portion of planned DOE effort.

Several design approaches may be devised to work around these problem areas including modifications of the antenna taper, microwave antenna diameter, transmitted power, and basic beam shape. Two different basic beam shapes have been studied and are illustrated in Figure 13--a Gaussian shape and a Bessel-J shape. Both patterns are designed to deliver 5 GW of power at the utility interface on the ground. The significant feature of the Bessel-J distribution is much lower minimum power density for the same delivered power. However, as shown, the rectenna for the Bessel-J beam is about 7.5 km radius (to 1 mW/cm^2) as compared to about 5 km radius for the Gaussian beam.

In order to arrive at the best combination of conditions, it was necessary to conduct a combined antenna/rectenna cost trade study. The results of such a study are contained in Figure 14 for klystrons and a 2.45-GHz frequency. This figure shows that two regimes exist, with largely different antenna diameters for the Gaussian and Bessel-J beams. Lines of constant maximum beam density (mW/cm^2) are shown by the solid lines going from the upper right to the lower left. Dashed lines intersecting the solid lines are lines of constant maximum heat flux which must be rejected at the center of the beam. A heat flux of 5.3 kW/m^2 is assumed as the baseline. Points of intersection of the

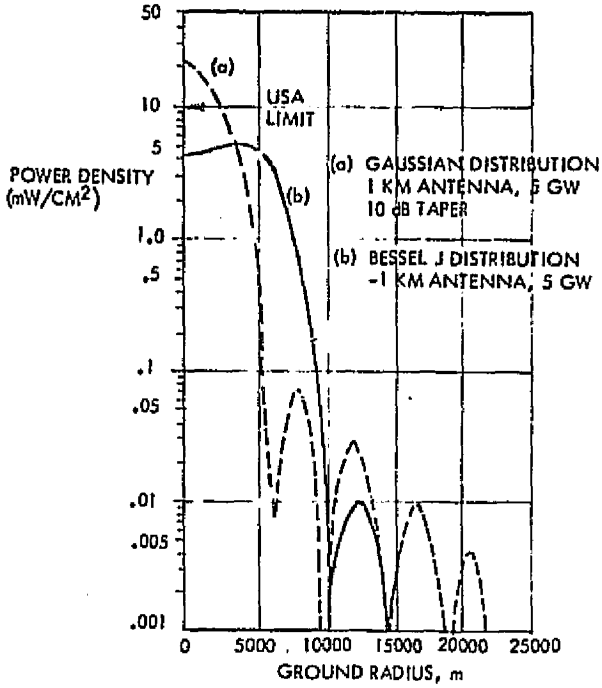


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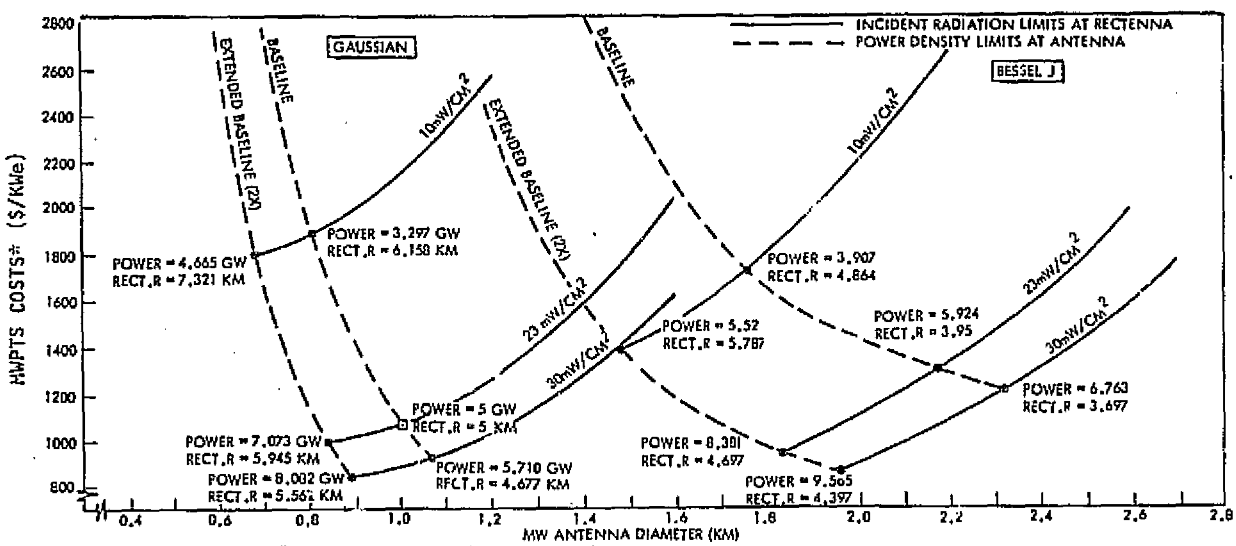
Figure 13. Microwave Radiation Problem Areas and Approaches



* COSTS INCLUDE RECTENNA, ANTENNA & ANTENNA TRANSPORTATION COSTS

Figure 14. MWPTS Cost Trades Comparison

dashed and solid lines result in minimum achievable cost. For the current maximum beam density of 23 mW/cm^2 and the baseline heat flux, the resulting rectenna diameter is 1.0 km, the rectenna radius is 5.0 km, and the power at the utility interface is 5 GW. At the same beam density and baseline heat flux, the Bessel-J has a higher total cost per kW, an antenna diameter of 2.2 km, a rectenna radius of 3.95 km, and 5.9 GW of power at the utility interface. The Bessel-J beam has the lowest cost at lower maximum beam densities such as 10 mW/cm^2.

RECTENNA CONCEPTS

The rectenna system is comprised of row-on-row of arrays which are oriented normal to the incoming microwave beam. Major attention to date has been given to arrays comprised of individual dipole antennas, each with their own diode rectifier. The major problems with this approach are the low rectification efficiency of the diodes at the low power densities that exist near the edge of the rectenna (where the major areas exist), and the large number of diodes that are required.

Other approaches exist that overcome the collection inefficiency and reduce the number of diodes required. One of these is the stripline pattern of bow-tie dipoles, illustrated in Figure 15. The stripline track collects the dipole signals at a common outlet where it is converted to dc by a diode rectifier. The bow-tie panels offer ease of manufacturing, excellent weather resistance, and low diode count. The diode count for this approach is only 2 percent of that required by the individual dipole approach. Because this approach results in a phased array, angular accuracy requirements of the beam are increased. This does not appear to pose any significant constraints on the satellite.

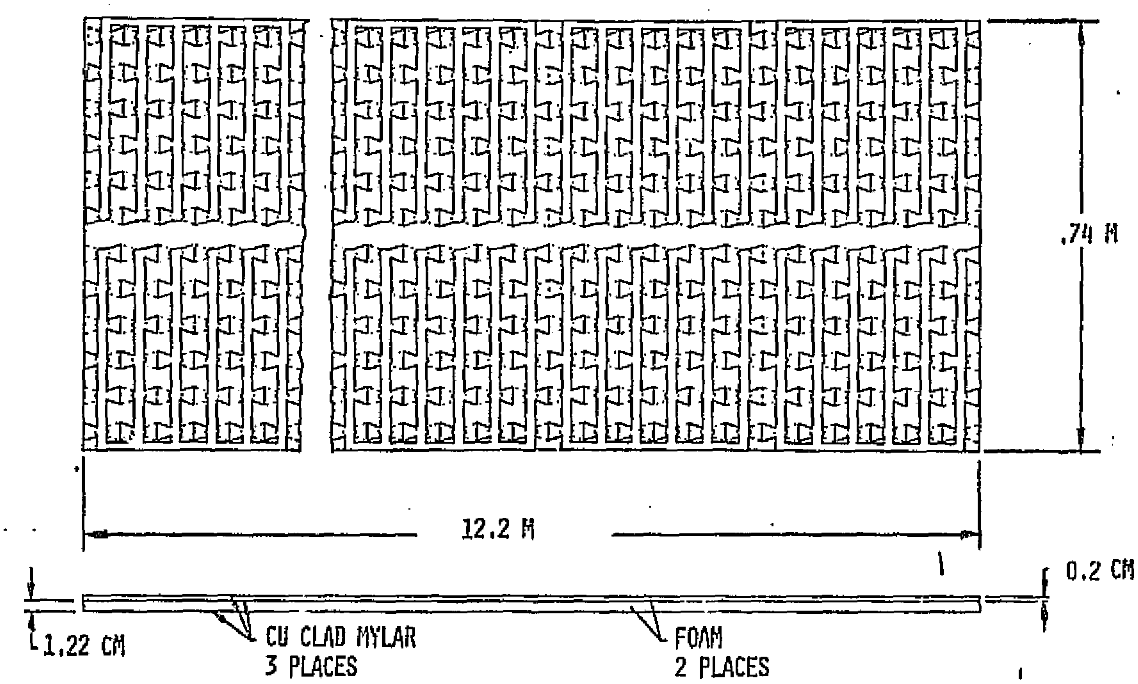


Figure 15. Stripline Rectenna Module (High-Density Area)

SATELLITE CONSTRUCTION SITE CONCEPTS

Two basic approaches for satellite construction have been considered. One of these concepts assumes that the entire satellite is constructed in geosynchronous orbit and that the necessary construction material is transported from low earth orbit to geosynchronous orbit using orbit transfer vehicles (OTV's). The other concept assumes that the satellite is either partially or totally constructed in low earth orbit (LEO) and then propelled to geosynchronous earth orbit (GEO) using power from the satellite's solar arrays to power ion-electric thrusters. If the satellite is only partially constructed in LEO, the remainder of material required for completion of construction in GEO is placed on board the satellite. The operations and timelines associated with these options are illustrated in Figures 16 and 17.

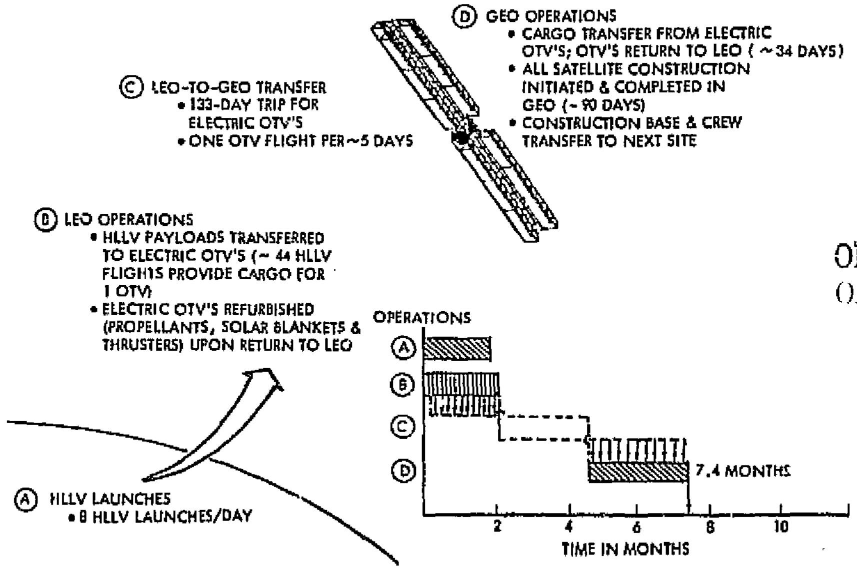


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Figure 16. GEO Satellite Construction Scenario

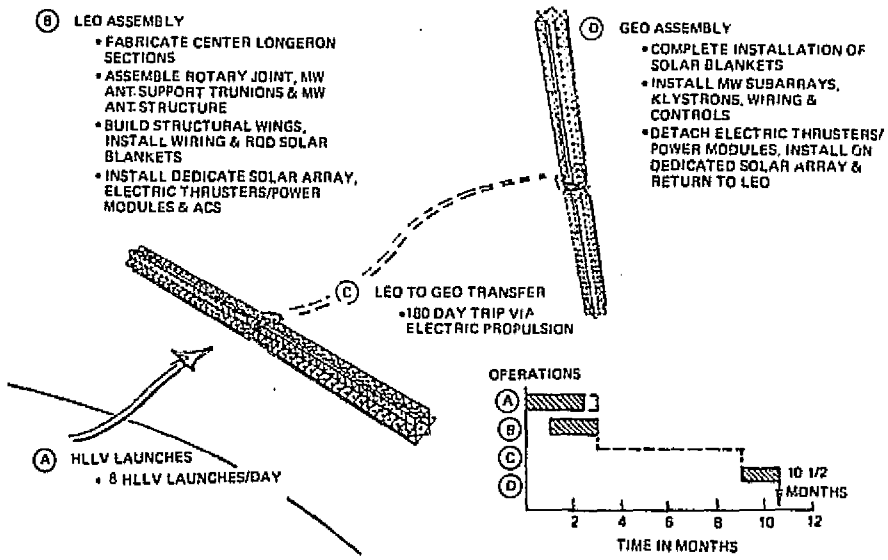

Figure 17. LEO-GEO Satellite Construction Scenario

Impacts on Overall Concept

Major differences exist in the environment in LEO and GEO. Two of the paramount differences are the gravity-gradient torques and the natural frequencies at the two locations. Figure 18 illustrates the impacts of these on the satellite configuration. The original GaAlAs solar cell photovoltaic concept having a concentration ratio of 2 had two troughs. Studies of construction location indicated that a satellite partially constructed in LEO has to accommodate much larger gravity-gradient torques during the time in LEO, and during much of the transfer during self-propulsion from LEO to GEO. This results in attitude control penalties for concepts which are poorly balanced about the Y-axis (axis perpendicular to the orbit plane during operations). The concept shown for combined LEO/GEO construction has three troughs arranged (end view) like an equilateral triangle, which gives the desired balance about the Y-axis. Because of the spread-out (about the Y-axis) nature of this configuration, it appears that construction would be difficult. Because of the rotational period being lower in LEO, a stiffer configuration is required, leading to a deeper structure.

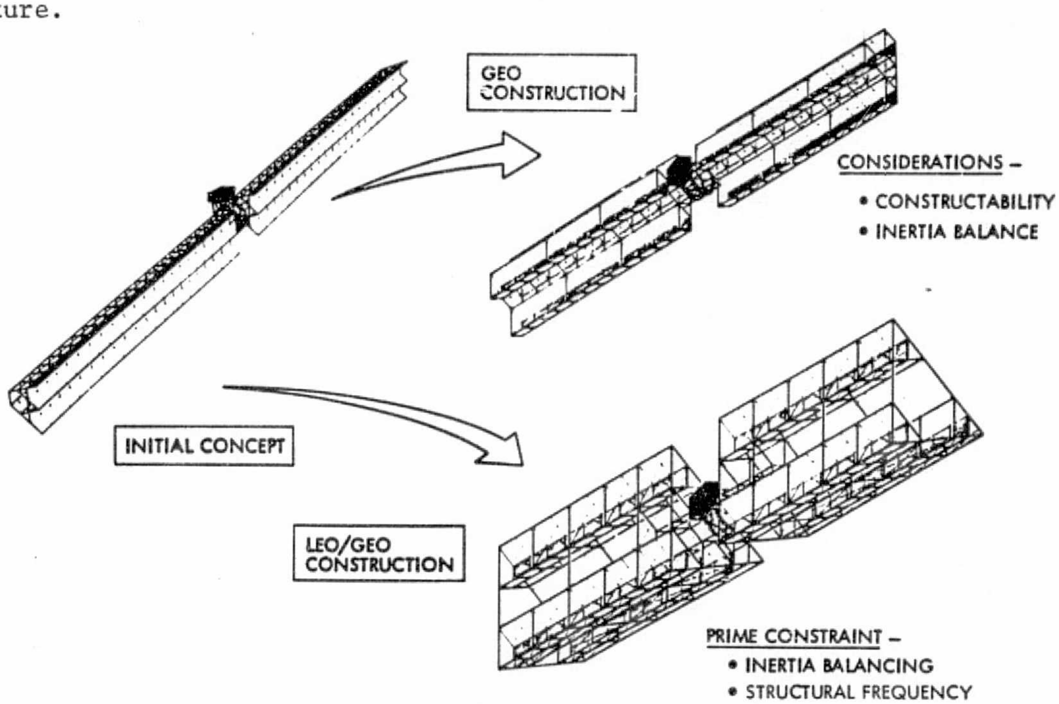

Figure 18. Photovoltaic Point Design Evolution

When construction is accomplished totally in GEO, large gravity gradients are no longer as significant a problem. As a result, the concept shown for GEO construction represents a compromise between Y-axis balancing and constructability. Although the concept still has three troughs, they are not arranged in an equilateral triangle and the resulting concept is more compact about the Y-axis. Although not perfectly balanced, the balance is adequate to reduce operational attitude control requirements due to gravity-gradient torque to a low level.

Cost Impacts

Another major area that was considered in comparing the construction site approaches is cost. Table 3 summarizes the results of the cost analysis using "delta" costs rather than total costs.

Table 3. Cost Differences for GEO and LEO/GEO Construction (GaAlAs Photovoltaic Satellite)

	CHEMICAL OTV	ELECTRIC OTV	
	GEO CONSTRUCTION	LEO/GEO CONSTR	GEO CONSTRUCTION
NO. OF HLLV LAUNCHES	1092	434	453
EARTH LAUNCH COSTS	$\$2597\times10^6$	$\$988\times10^6$	$\$2029\times10^6$
ELEC. PROP. MODULE REPLACEMENT COSTS	-	$\$128\times10^6$	$\$156\times10^6$
INTEREST COSTS (7.5%)	-	$\$272\times10^6$	$\$255\times10^6$
TOTALS	$\$2597\times10^6$	$\$1388\times10^6$	$\$1410\times10^6$

Two OTV approaches are shown for all-GEO construction: an LO_2/LH_2 chemical OTV and an electric OTV. For partial LEO and partial GEO construction, electric propulsion is also used, but the partially constructed SPS provides the power to the electric thrusters and transports the mass to be used for partial GEO construction to GEO (self-propulsion). Although long-duration travel is involved through the Van Allen Belt for both electric propulsion concepts (the electric OTV and self-propelled SPS), radiation damage to the solar cells was assumed to be self-annealed (GaAlAs solar cells). Interest costs are included because of the long trip time from LEO to GEO for electric propulsion transfer.

A comparison of the totals shows that GEO construction using a chemical OTV for cargo transfer from LEO to GEO costs about $1.2 billion more than either electric OTV option. Costs of the self-powered (combined LEO/GEO construction) and the GEO construction options with an electric OTV are about the same.

Other Considerations

In addition to the above considerations, trade studies also considered other factors of construction site related to the subsystems, transportation systems, and operations; these are summarized in Table 4. The accumulation of these factors tend to favor geosynchronous construction.

Table 4. Additional Trade Study Results

AREA OF INVESTIGATION	LEO PREFERRED	GEO PREFERRED	REASONS
SPS SUBSYSTEMS			
MICROWAVE		X	OPEN TUBES, NO EFFICIENCY LOSS DUE TO MULTIPACTION
THERMAL		X	LESS FREQUENT TEMP CHANGES
STRUCTURAL		X	LESS STIFFNESS
STABILITY AND CONTROL		X	LESS CONTROL TORQUE AND OCCULTATION PROBLEMS
POWER CONVERSION		X	NO CELL DEGRADATION AND TOTAL ARRAY DEPLOYMENT (WITHOUT SELF-ANNEALING)
POWER DISTRIBUTION		X	NEEDS ONLY SINGLE SYSTEM AT 40 KV
CREW SYSTEMS	X		NO MAJOR DISCRIMINATORS
FLIGHT AND ORBITAL MECHANICS		X	ORBITAL TRANSFER OF SPS
TRANSPORTATION SYSTEMS	X		RELIEVES OTV SELECTION
OPERATIONS	X		FEWER HLLV FLIGHTS

TRANSPORTATION CONCEPTS

Transportation requirements and concepts for SPS vary as a function of program phase. Figure 54 , presented later in the Programmatics section, illustrates the time-phased SPS transportation requirements. During the verification planning period (1981-1987), the baseline Shuttle is initially used to conduct sortie missions. Later in the verification program, an extended-duration orbiter configuration is necessary to conduct on-orbit experiments related to on-orbit construction. Following the verification phase, from 1988 to 1998, the mass to orbit will increase significantly to support construction of a large-scale (possibly 1-GW) prototype demonstration. In order to reduce launch costs, a Shuttle-derived heavy-lift launch vehicle (HLLV) will be needed. Additionally, during this period, a Shuttle-derived manned orbit transfer vehicle (MOTV) will be required to carry personnel to geosynchronous orbit. When commercialization of SPS is initiated in about 1998, the masses to orbit will again increase by large amounts and it will be necessary to have a transportation system specially suited for SPS, including manned and cargo orbit transfer vehicles and a heavy-lift launch vehicle for earth to low orbit transportation with launch costs of approximately $10 per pound (1977 dollars). This study concentrated on the transportation requirements during the post-1998 period. The results of these analyses are presented in the following paragraphs.

Operational Transportation Requirements

Some of the basic transportation system requirements include orbital attitudes and inclinations, mass flow to orbit, environmental concerns, and cost.

Geosynchronous orbit is the eventual destination of SPS construction equipment and supplies, unless the SPS is built entirely in LEO. Under any circumstances, it will be necessary to transport the crew from earth to LEO and from LEO to GEO. During the period of SPS operations, cargo for maintenance and crew will have to be transported from earth to LEO and from LEO to GEO. Two low earth orbits were considered--a 500-km orbital altitude at approximately 28.5 degrees inclination, and an equatorial LEO of 550 km. The desired LEO is a function of earth launch vehicle concept selection.

The minimum mass flow to orbit for a mature program ranges from 200×10^6 to 350×10^6 kg/year for the electric OTV concept, and as high as 10^9 kg/year for a chemical OTV. Dependent upon HLLV payload capability, this results in HLLV flight rates ranging from 500 to 4000 per year.

Because of the large quantities of fuels consumed by the HLLV, environmental consideration must be given to fuel consumption and storage hazards, atmospheric contamination in localized areas, and the maximum allowable acoustic levels in the area of the launch site. In addition, since a significant mass will be required for packaging of payload elements, the ultimate disposition of this mass in orbit must be considered.

Since the operational cost of the transportation system represents a major portion of SPS costs, transportation system selection is dominated by \$/kg to orbit. Costs of the order of \$22/kg to LEO are required to meet the economic goals of SPS. This implies rapid turnaround time, low-cost recovery and refurbishment, and virtually complete reusability of system components.

<u>Transportation System Options</u>

The transportation system operational regimes include earth to LEO, LEO to GEO, and on-orbit (for short distance and duration flight). These systems must be capable of transporting both crew and cargo.

Table 5 lists the options that have received serious consideration for SPS in each of these regimes. The earth-to-LEO concepts are arranged from top to bottom in the order of increasing technology and decreasing operational complexity. The first two concepts take off vertically (the ballistic also

Table 5. Transportation System Options

CARGO	PERSONNEL/PRIORITY CARGO
EARTH - LEO	
• TWO-STAGE BALLISTIC • TWO-STAGE WINGED - VTO • HTO - TWO STAGE • HTO - SSTO	• SHUTTLE DERIVATIVE • TWO-STAGE WINGED - VTO • HTO - TWO STAGE • HTO - SSTO
LEO - GEO	
• CHEMICAL • NUCLEAR - GCR • ELECTRIC - SELF PROPELLED - DEDICATED	• CHEMICAL • NUCLEAR - GCR
ON ORBIT	
• CHEMICAL	• CHEMICAL

land vertically), and the last two take off and land horizontally. The two-stage ballistic launch vehicle is intended to carry cargo to LEO, and it would require a Shuttle derivative to carry personnel and priority cargo to LEO. The other three HLLV options can carry cargo and personnel to LEO.

Three basic approaches have been considered for transportation from LEO to GEO: chemical (LH_2/LO_2), nuclear (gas core reactor), and electric propulsion systems. The chemical and nuclear systems both have high thrust and, therefore, short flight times from LEO to GEO. Consequently, they can transport both cargo and personnel. The electric systems include both self-propelled (in the event the SPS is partially or totally constructed in LEO) and dedicated (in the event the SPS is built in GEO) concepts. For either electric propulsion concept, it is necessary to have either a chemical or nuclear vehicle to transport crew. The chemical concept appears to be the best candidate in conjunction with electric propulsion because of the relatively low mass-flow of crew and priority cargo compared to construction cargo. The only option analyzed for on-orbit propulsion in this study was an LO_2/LH_2 chemical system.

Figure 19 illustrates some of the concepts described above. The HTO-HLLV depicted is a single-stage concept with a payload capability of 91,000 kg to equatorial LEO. The two-stage ballistic VTO-HLLV has a payload capability of approximately 400,000 kg to LEO at 28.5 degrees inclination. The chemical and nuclear GCR are sized to deliver a 91,000-kg payload from LEO to GEO. The dedicated electric OTV is sized to deliver approximately 4×10^6 kg from LEO to GEO. The upper portion of the figure compares its size with the SPS satellite.

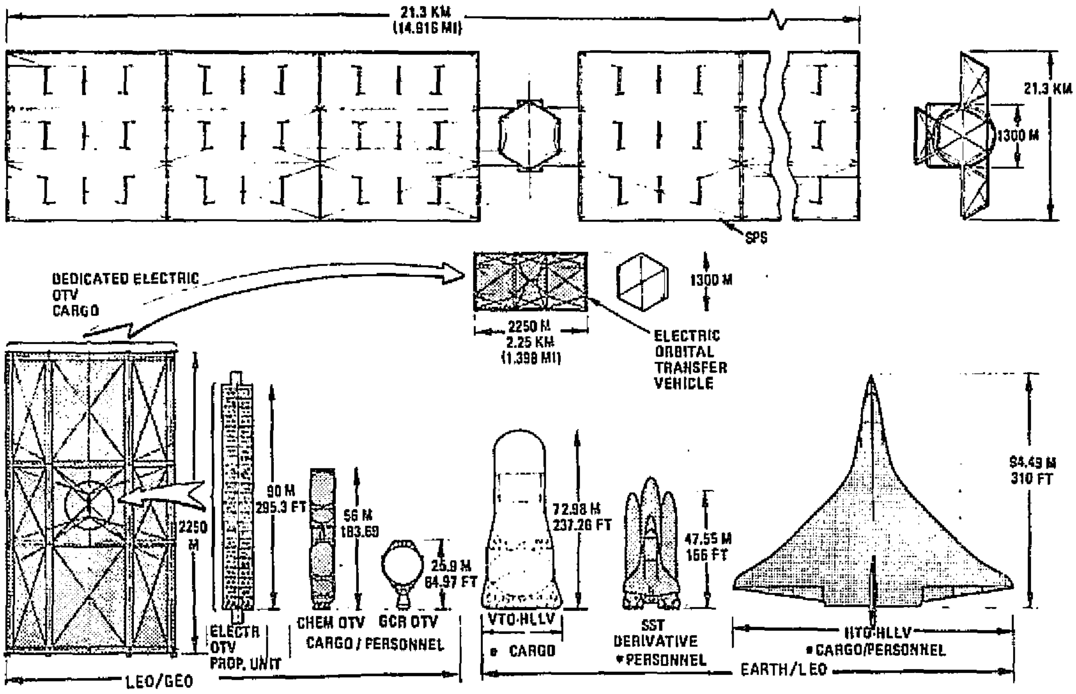

Figure 19. Comparison of Transportation System Options

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HLLV Operations Comparison

Some of the key areas of comparison between the two-stage ballistic HLLV and the HTO-SSTO configurations are presented in Table 6. Because of the larger payload capability of the VTO-HLLV, four launches per day are required as opposed to 16 HTO-SSTO flights. In order to meet the launch rate requirement, ten launch pads are required for the VTO-HLLV; whereas, a single runway may be employed for the HTO vehicle. Because of the stacking requirement for the two-stage ballistic HLLV, two high bay vertical assembly buildings are required as opposed to two aircraft maintenance-type buildings for the HTO vehicle. In addition, considerable heavy handling equipment is required for moving and stacking of the VTO-HLLV. The difference in the number of processing stations is due to two-stage versus single-stage processing. The turnaround time for the HTO vehicle is approximately one-third that for the VTO-HLLV, primarily because of the recovery method and no requirement for mating and stacking. The risk of recovery damage from water impact is considerably higher than that for aircraft landing. The launch site area requirements are driven by the number of launch pads required and the separation required to minimize the acoustic hazard for personnel and equipment. The maximum allowable level without ear protection is 130 dB, and 120 dB is the maximum allowable on a repetitive basis.

Table 6. HLLV Operations Comparison

	TWO-STAGE BALLISTIC	HORIZONTAL TAKEOFF
• LAUNCH RATE/DAY	4	16
• LAUNCH WINDOWS/DAY	2/ORBIT (3 HR)	12/ORBIT (CONT.)
• LAUNCH PADS	TWO HIGH-BAY BUILDINGS	2 A/C MAINT. TYPE
• HANDLING	CRANES, TRANSPORTERS, TUGS, MOBILE LAUNCH PLATFORMS & CRAWLERS, & RECOVERY SHIPS	TWO VEHICLE
• PROCESSING STATIONS	16	10
• ENGINES/VEHICLE	24	11-14
• FLEET SIZE (W/O ATTRITION)	22 FIRST STAGES 23 SECOND STAGES	30
• TURNAROUND TIME (DAYS)	5.5	1.8
• LAUNCH PAD REFURB.	EXTENSIVE	NIL
• RISK OF RECOVERY DAMAGE	HIGH	NIL
• ACOUSTIC LEVELS	130 dB @ 5.6 km 120 dB @ 13.0 km	<120 dB @ 1 km
• LAUNCH SITE AREA REQUIRED	850 km^2	<20 km^2

POINT DESIGN DEFINITION

As a result of the trade studies described in the previous section on SPS evolution, two satellite point designs were selected for further definition—a photovoltaic satellite with GaAlAs solar blankets, and a solar thermal concept with a cesium Rankine/steam bottoming cycle. Major emphasis in point design definition was placed on the photovoltaic concept.

The following sections summarize the rationale for their selection and describe the characteristics of the point designs.

POINT DESIGN SELECTION RATIONALE

In the eventual selection of an SPS concept, several criteria related to cost, technology requirements, resources availability, environmental impacts, operational considerations, and complexity will be applied. A preliminary list of criteria were developed and are presented in Table 7. At this point in the SPS program, not all of these criteria can be evaluated, and the following comparisons will be made on the basis of technology issues, critical material availability, and cost.

Table 7. Potential Concepts Selection Criteria

COST DDT&E INITIAL CAPITAL INVESTMENT* REPLACEMENT CAPITAL INVESTMENT OPERATIONS AND MAINTENANCE* COST SENSITIVITY • TO SATELLITE BUILDUP RATE • TO TRANSPORTATION COSTS • TO CRITICAL SUBSYSTEM COSTS*	ENVIRONMENTAL IMPACTS SPACE IMPACT (SATELLITE SIZE, RFI, AND NUCLEAR RADIATION) TRANSPORTATION IMPACT (NO. OF LAUNCHES) NUCLEAR- (DISPOSAL, PROCESSING, ETC.)*
TECHNOLOGY REQUIREMENTS CRITICAL DEVELOPMENTS (NUMBER & TYPE)* VERIFICATION TESTING COMPLEXITY (ANALYTICAL, GROUND, SPACE, AND RELATIVE COST/COMPLEXITY) SCHEDULE IMPACTS (STATE-OF-THE-ART CUTOFFS) GROWTH POTENTIAL ($\Delta\eta$, ΔW) MODULARITY/SCALEABILITY	OPERATIONAL CONSIDERATIONS RELIABILITY (FMEA DATA) SUPPORT REQUIREMENTS (SUCH AS SPECIAL EQUIPMENT FOR NUCLEAR) SAFETY CONSIDERATIONS (HAZARDS--TO THE SATELLITE AND CREW) PERFORMANCE DEGRADATION (SOLAR DELLS, REFLECTORS, RADIATORS, & MACHINERY) MAINTAINABILITY (DOWN TIME) ECLIPSE FACTORS (SATELLITE OVERSIZE AND GROUND REQUIREMENTS)
MATERIAL AVAILABILITY GALLIUM AND CESIUM*	COMPLEXITY DESIGN COMPLEXITY MANUFACTURING AND ACCEPTANCE CONSTRUCTION AND ASSEMBLY SUBSYSTEM INTERFACES
*PRIMARY CONCEPT DIFFERENTIATORS	

Table 8 lists the most significant technology issues that differentiate the concepts. It should be noted that microwave transmission subsystem issues do not appear here because they are common to all concepts. Both silicon and

Table 8. System Concept-Related Technology Issues

SYSTEM CONCEPT	CRITICAL TECHNOLOGY ISSUES
	SOLAR PHOTOVOLTAIC
SILICON SOLAR CELLS	THIN FILM SILICON--HIGH TEMPERATURE ANNEALING, WEIGHTS, EFFICIENCY
GaAs SOLAR CELLS	SELF-ANNEALING TEMP CONTROL, AL_2O_3 SUBSTRATE, MATERIAL COSTS
COMMON REQUIREMENTS	LOW-COST SOLAR BLANKET PRODUCTION
	SOLAR THERMAL
BRAYTON CYCLE	HIGH-TEMP HEAT EXCHANGER MATERIALS--LIQUID METAL COOLING--RIGID CONCENTRATOR MATERIALS--CAVITY ABSORBER
RANKING CYCLE	INFLATABLE CONCENTRATOR MATERIALS/SURFACE DEGRADATION CONDENSING TUBE/FIN RADIATOR
COMMON REQUIREMENTS	FABRICATION/ASSEMBLY TOLERANCES--SOLAR POINTING REQUIREMENTS OPTICAL PROPERTIES--AC GENERATION/HIGH-VOLTAGE TRANSMISSION
	NUCLEAR THERMAL
HIGH-TEMPERATURE BREEDER REACTOR TECHNOLOGY (FUEL PROCESSING PLANT) RADIATION SHIELDING--MAINTENANCE	

GaAlAs solar cells have critical development problems. Thin-film silicon cell technology is necessary to provide adequate efficiency at low weight to keep blanket weight and related transportation costs in a competitive range. Additionally, since silicon cells must be annealed at 500°C to alleviate radiation damage (compared to 125°C for GaAlAs cells), a method needs to be developed for on-orbit use that will not damage the substrate and other subsystems. A low-cost method of production of either the silicon or GaAlAs cells needs to be demonstrated. Additionally, a substrate material replacement for GaAs in the GaAlAs cells, such as synthetic sapphire (Al_2O_3), must demonstrate good cell efficiency. The major technology difference between the Brayton and Rankine thermal cycles is the material requirements, because of high turbine inlet temperatures. The Brayton cycle requires ceramic technology; whereas, the Rankine cycle requires a more easily obtained refractory metal technology. The nuclear system has the engine technology problems of the Brayton cycle and the additional technology problems of a high-temperature breeder reactor (automatic fuel processing).

Material availability is an issue for the GaAlAs solar thermal concept (Ga availability) and for the cesium Rankine solar thermal concepts (cesium availability). Figure 20 summarizes data related to availability of Ga and cesium. The Ga data assume the use of a synthetic sapphire substrate for the GaAlAs cells. These data indicate that sufficient materials should be available for 120 5-GW satellites.

A comparison of satellite specific weights, shown in Figure 21, indicates that (1) the silicon concept has the greatest weight, (2) the CR = 2 GaAlAs concept has the lowest weight, (3) the cesium Rankine solar thermal concept is the lowest-weight solar thermal concept, (4) the potassium solar thermal concept

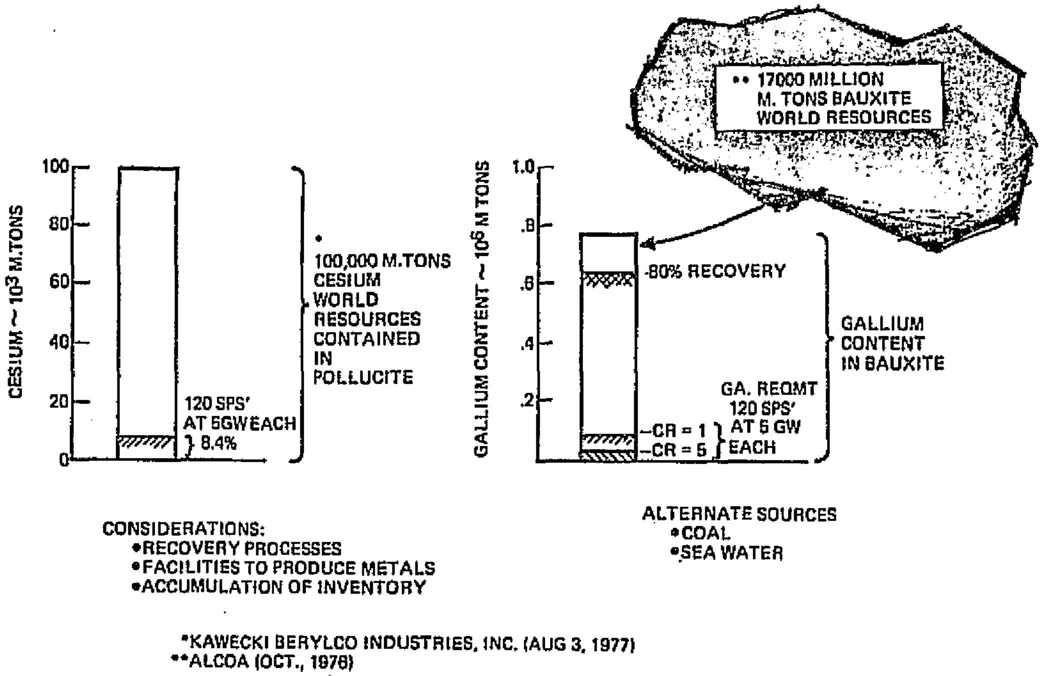

Figure 20. Material Availability Evaluation

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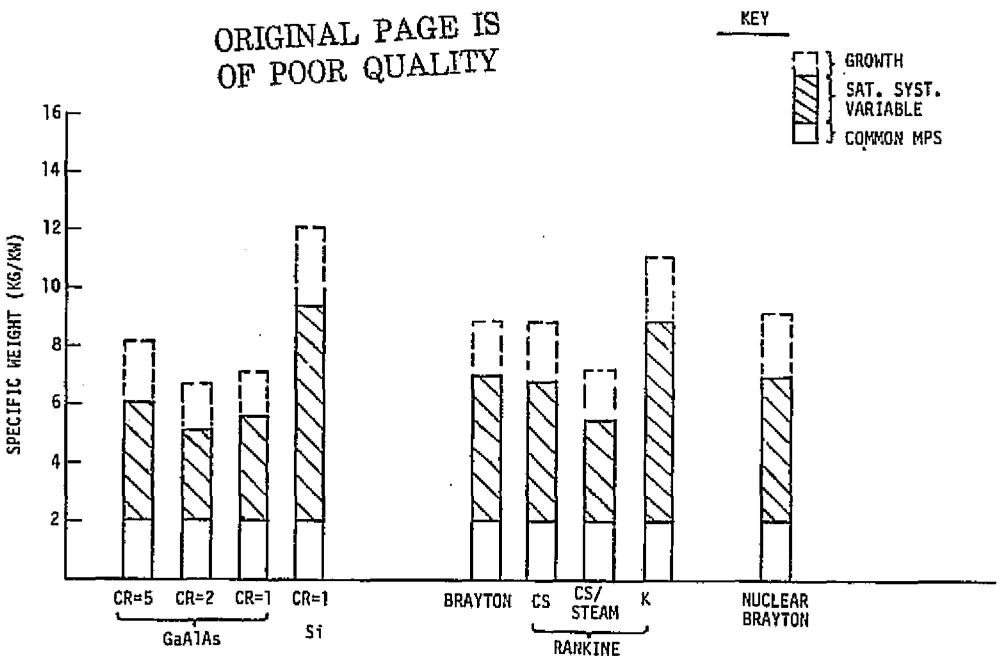

Figure 21. Satellite Weight Comparisons

is the highest weight of all solar thermal concepts, and (5) the nuclear Brayton weight is generally in the same range as the solar thermal systems. System weight is directly related to transportation costs.

Figure 22 compares the total relative investment cost for the concepts. The CR = 2 and CR = 5 GaAlAs concepts have about the same cost and are the lowest-cost systems analyzed. The cesium Rankine solar thermal concept with steam bottoming is the lowest-cost solar thermal system. As shown, the nuclear Brayton concept is considerably higher in cost than the other concepts.

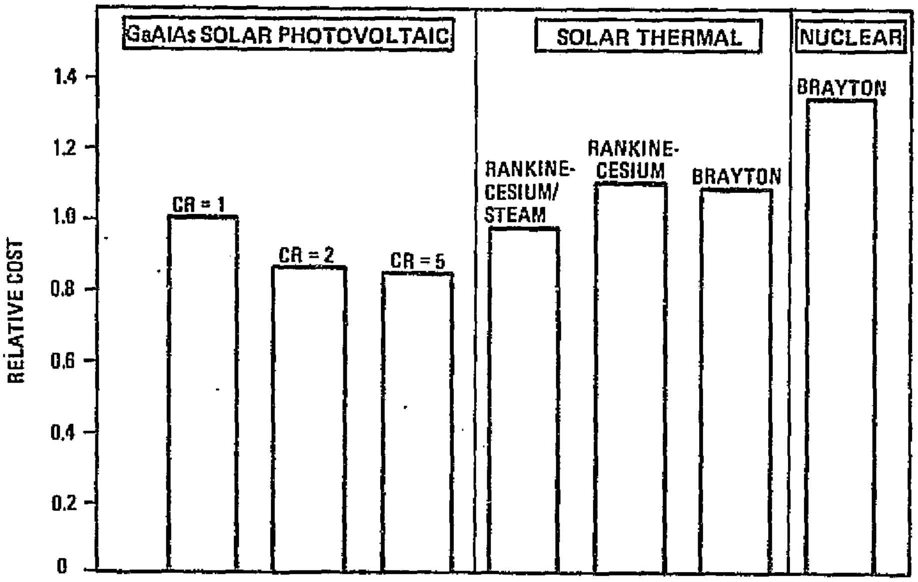

Figure 22. Total Investment Cost Comparisons

The nuclear concept has considerations which are unique because of safety and international considerations. There does not appear to be anything inherent in the design of a breeder reactor to indicate that failure rates and modes are reduced by operation in space. On the contrary, failure mode corrections in space would be more difficult than on earth. Potential radiation poisoning at geosynchronous altitude could present a long-range problem to the future use of this very important earth orbit. It may be concluded that space-based versus earth-based nuclear systems is a national policy issue as opposed to a technical issue. For this reason, it was recommended that no further studies of nuclear concepts be made in this contract.

The concepts recommended as candidate SPS satellite options, based on these trade studies, are shown in Figure 23. The GaAlAs concept is recommended as the primary photovoltaic concept with silicon as an alternative. The cesium Rankine solar thermal satellite is also recommended as a candidate option. Primary emphasis in the point design studies was placed on the GaAlAs photovoltaic concept. Further definition of the cesium Rankine solar thermal concept with steam bottoming also was accomplished.

In addition to the selection of the satellite concepts, specific rectenna, satellite construction site, and transportation system approaches were selected

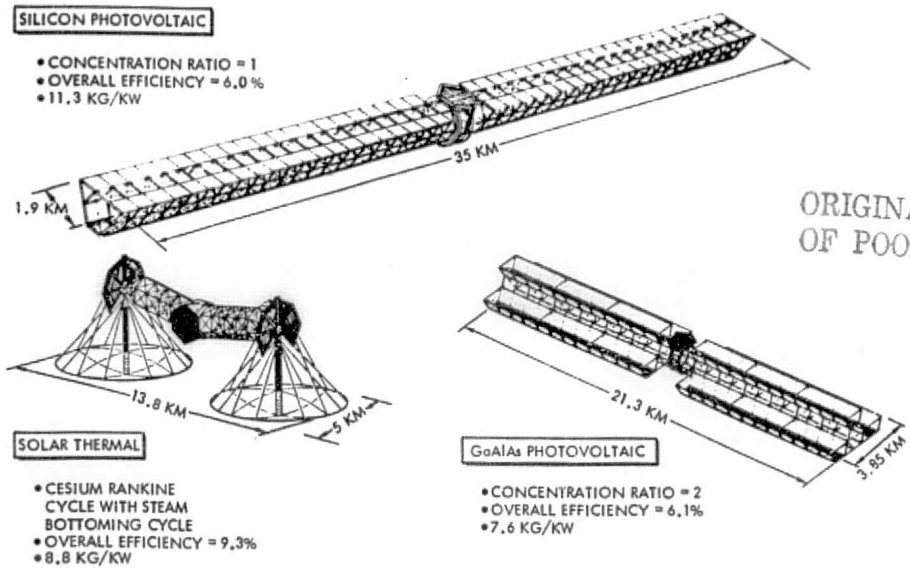

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Figure 23. Present Candidate Options (All 5 GW at Utility Interface)

as point design options. The stripline rectenna described during the tradeoff discussion was selected for additional definition. Total SPS construction in geosynchronous orbit was recommended, with a dedicated electric OTV for cargo transport and a two-stage chemical OTV for personnel transport. All of the HLLV concepts currently under consideration for earth-to-LEO transportation appear to be viable. However, since little effort has been spent on the horizontal takeoff concept, this approach is recommended for additional definition.

The following sections summarize the point design characteristics.

PHOTOVOLTAIC SATELLITE POINT DESIGN

The photovoltaic concept shown in Figure 24 was designed for construction at GEO. This three-trough configuration delivers 5 GW to the utility interface on the ground. It has a single, centrally located microwave antenna.

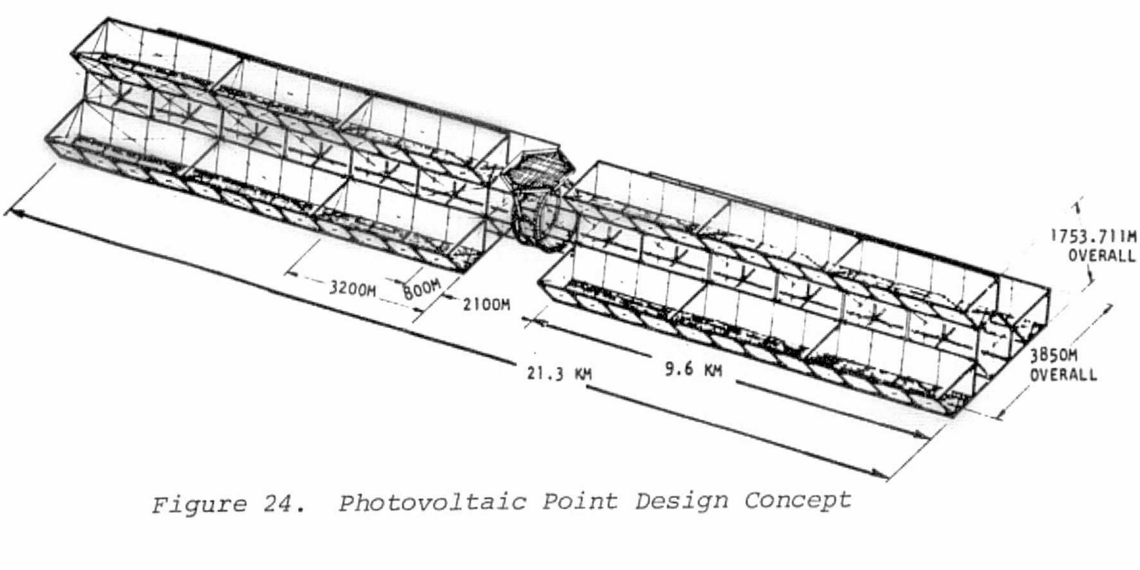


Figure 24. Photovoltaic Point Design Concept

GaAlAs solar cells are used to produce the power using concentrators that give a 2:1 concentration ratio. A more detailed description of the system and subsystems is given in Table 9.

Table 9. Photovoltaic Point Design Characteristics

OVERALL DESCRIPTION

- 5-GW POWER TO UTILITY INTERFACE
- GEOSYNCHRONOUS CONSTRUCTION LOCATION
- SINGLE MICROWAVE ANTENNA
- GEOSYNCHRONOUS EQUATORIAL OPERATIONAL ORBIT

SUBSYSTEMS

POWER CONVERSION
- GaAlAs SOLAR CELLS
- CONCENTRATION RATIO = 2

ATTITUDE CONTROL/STATIONKEEPING
- Y-POP, X-IOP
- ARGON ION THRUSTERS

POWER DISTRIBUTION
- 45.5 KV DC
- STRUCTURE/WIRING NOT INTEGRATED

MICROWAVE ANTENNA
- GAUSSIAN BEAM
- 2.45-GHz FREQUENCY
- ELECTRIC PHASE CONTROL
- RCR WAVEGUIDE PANELS
- TENSION-WEB, COMPRESSION FRAME STRUCTURE

STRUCTURE
- ALUMINUM (GRAPHITE/THERMAL PLASTIC ALTERNATE AS NEEDED)
- BEAM MACHINE CONSTRUCTION

INFORMATION MANAGEMENT
- DISTRIBUTED

Figure 25 shows the end-to-end efficiency chain for the recommended baseline concept which has been sized to provide 5 GW of electric power to the utility busbar. With an overall efficiency of 6.08 percent, it is necessary to size the solar arrays to intercept 82.2 GW of solar energy. The quoted efficiency is the minimum efficiency, including the worst-case seasonal variation (91%), the end-of-life (30-year) concentrator reflectivity (86%), and the end-of-life (30-year) solar cell efficiency (15.2%).

A summary of the satellite mass properties is presented in Table 10. The two major segments, the collector array and the antenna section, are nearly equal in mass. The major contributor to the collector array mass is the power source, which includes the solar blanket and the reflectors. The solar blanket is the predominant mass. Antenna section mass properties are driven by the microwave power segment which includes the RF radiators and the klystrons. Total structure and mechanism mass is approximately 20 percent of the satellite dry weight. Total satellite weight, including a 30-percent growth factor, is 36.5-million kilograms. Propellant resupply for attitude control and stationkeeping is a very small annual mass compared to the satellite mass.

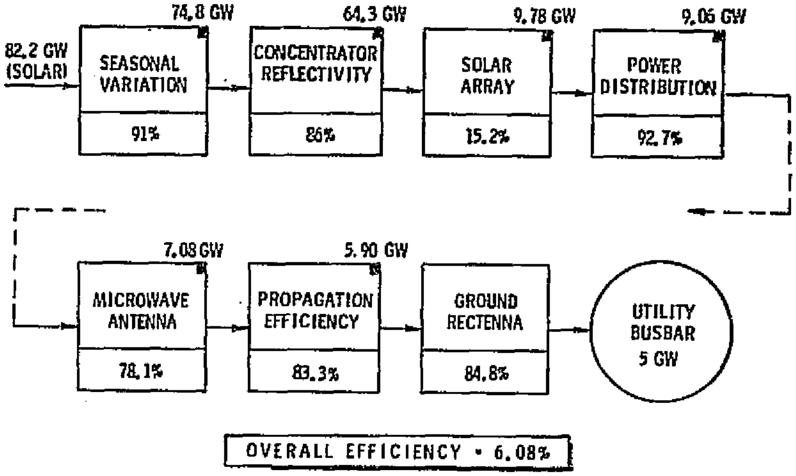


Figure 25. Photovoltaic Point Design End-of-Life Efficiency Chain

Table 10. Photovoltaic Point Design Mass Statement

SUBSYSTEM	WEIGHT (MILLION KG)
COLLECTOR ARRAY	
STRUCTURES AND MECHANISMS	3.777
POWER SOURCE	8.830
POWER DISTRIBUTION & CONTROL	1.166
ATTITUDE CONTROL	0.095
INFORMATION MANAGEMENT & CONTROL	0.049
TOTAL ARRAY (DRY)	(13.917)
ANTENNA SECTION	
STRUCTURE AND MECHANISMS	1.685
THERMAL CONTROL	1.408
MICROWAVE POWER	7.012
POWER DISTRIBUTION & CONTROL	3.438
INFORMATION MANAGEMENT & CONTROL	0.630
TOTAL ANTENNA SECTION (DRY)	(14.167)
TOTAL SPS DRY WEIGHT	28.084
GROWTH (30%)	8.425
TOTAL SPS DRY WEIGHT WITH GROWTH	36.509
PROPELLANT PER YEAR	0.040

The following paragraphs summarize the characteristics of the major satellite subsystems including energy conversion, power distribution, attitude control and stationkeeping, and antenna.

Energy Conversion

Figure 26 shows the configuration of the SPS point design solar array wing structure. The concept is a three-trough, two-tier system. The structure is made up of tri-beam girders whose longitudinal members and transverse struts are fabricated on orbit by a beam machine. Shear stabilization of the tri-beam girders and the wing itself is achieved by the use of X-tension cables. Current structure material is structural aluminum. Excessive stresses and/or deflections could drive the material selection to the regime of composites. The dimensions indicated have been verified to be adequate when the vehicle is subjected to a worst-case forces and torques environment in geosynchronous orbit in that they result in an acceptable margin of safety for a basic material thickness of 0.254 mm (0.010 in.), which is considered minimum gauge.

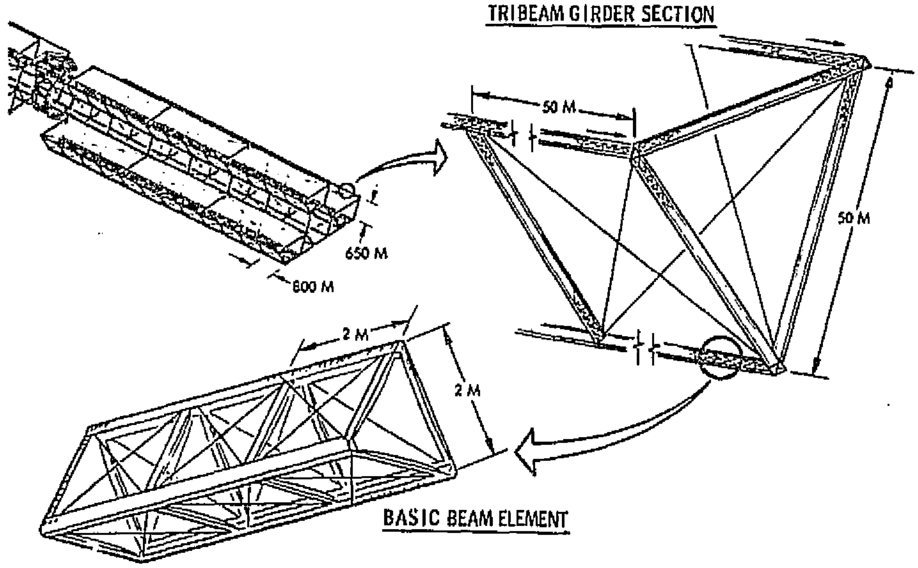


Figure 26. Photovoltaic Wing Structure

Figure 27 shows the solar array blanket description and array characteristics. The point design utilizes a GaAlAs solar cell efficiency of 20-percent AMO, 28°C, and the sizing of the array is based on 125°C operating temperature (17.6% cell efficiency). The total output of the array is 9.92 GW with a voltage output of 45.5 kW for each array panel. The solar blanket weight is 7.65×10^6 kg, and the total array weight (including the concentrator) is 8.83×10^6 kg. This weight is based on a specific weight for the blanket of 0.25 kg/m^2 and 61.2×10^6 m^2 cell area. A cross-section of the solar cell is also shown (Figure 27). The 20-μm synthetic sapphire (Al_2O_3) substrate, used in an inverse orientation, also acts as the cell cover. The reflectors are composed of 12.5-μm aluminized kapton.

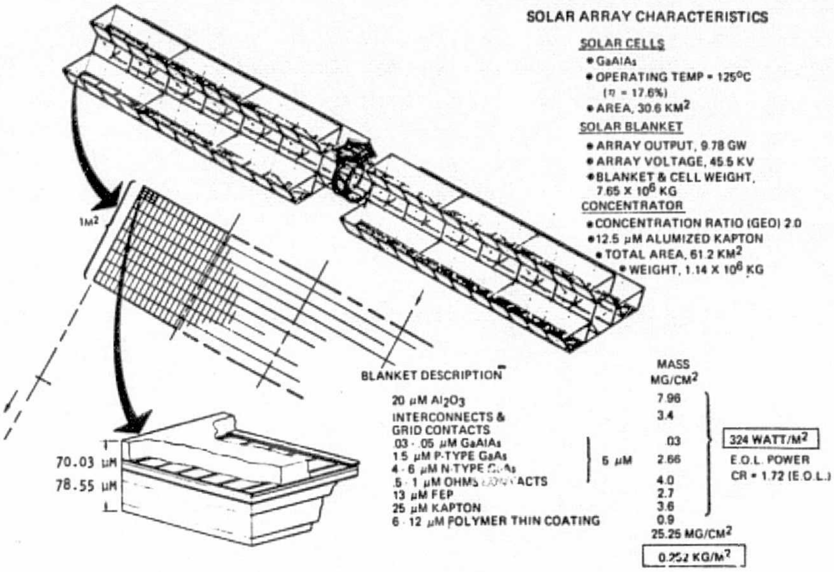


Figure 27. Photovoltaic Energy Conversion System

Power Distribution

A flow diagram of the overall power distribution subsystem is presented in Figure 28. Power obtained at the subarray is transferred to a summing bus through a switch gear (SG) and manually operated circuit-breaker. Power is then transferred from the nonrotating member to the rotating member of the rotary joint through slip rings and brushes. On the rotating member, power is conducted through switch gears to dc/dc converters which output the six primary voltages required by the klystrons. Each voltage is conducted to a summing bus through a switch gear. Subsequently, each voltage is conducted from the summing buses to the 135,864 klystrons.

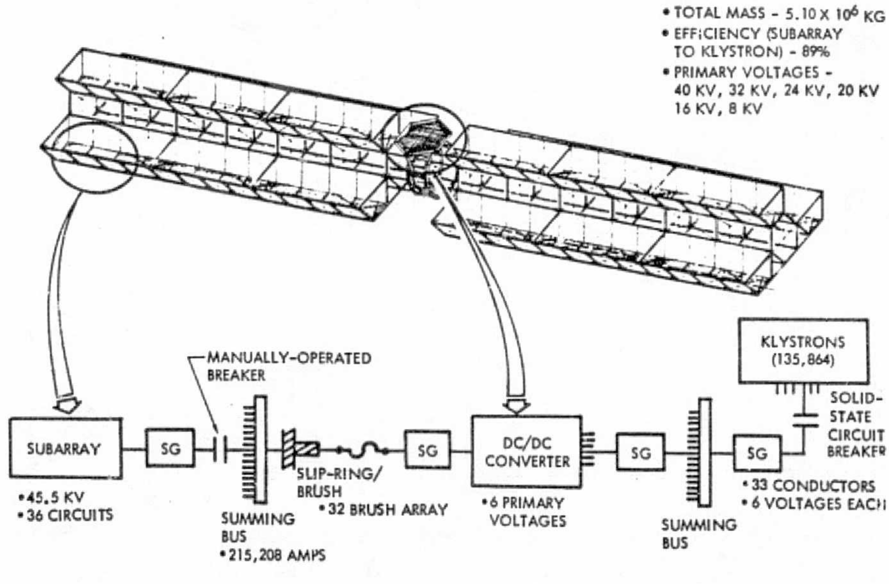
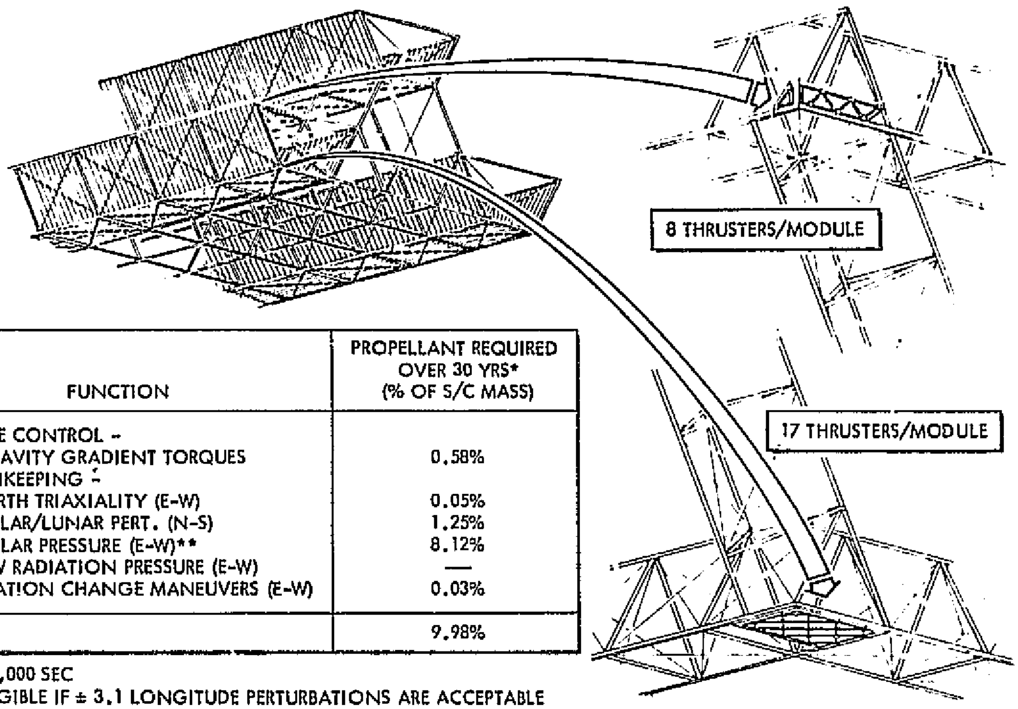


Figure 28. Photovoltaic Power Distribution Subsystem

Attitude Control and Stationkeeping

Trade studies indicated the desirability of a simple ACSS employing high-performance electric thrusters, and use of the Y-POP, X-IOP orientation and inertia balancing to minimize attitude control propellants. Figure 29 shows such a system employing eight RCS quads, two on each corner of the spacecraft. The total RCS propellant requirements (see table) are low, due primarily to the high specific impulse (13,000 s) which is believed to be feasible with the argon ion bombardment thrusters.



FUNCTION	PROPELLANT REQUIRED OVER 30 YRS* (% OF S/C MASS)
ATTITUDE CONTROL –	
• GRAVITY GRADIENT TORQUES	0.58%
STATIONKEEPING –	
• EARTH TRIAXIALITY (E-W)	0.05%
• SOLAR/LUNAR PERT. (N-S)	1.25%
• SOLAR PRESSURE (E-W)**	8.12%
• MW RADIATION PRESSURE (E-W)	—
• STATION CHANGE MANEUVERS (E-W)	0.03%
TOTAL	9.98%

*I_{sp} = 13,000 SEC
** NEGLIGIBLE IF ± 3.1 LONGITUDE PERTURBATIONS ARE ACCEPTABLE

Figure 29. Photovoltaic Attitude Control and Stationkeeping Subsystem

The dominant stationkeeping propellant requirement is the complete correction of the solar pressure perturbation. This requirement can be eliminated if a ±3.1-degree longitude stationkeeping accuracy is acceptable. The present stationkeeping accuracy goal is 0.5 degree. An effort is currently underway to define a solar pressure correction policy which meets the 0.5-degree accuracy requirement, but will reduce the propellant requirement.

The ACSS attitude reference determination system features charge-coupled device (CCD) star and sun sensors as well as electrostatic or laser gyros and dedicated microprocessors. Five attitude reference determination units are at various locations on the spacecraft to sense thermal and dynamic body bending and to desensitize the system to these disturbances. The control algorithms will feature statistical estimators for determining principal axis orientation, body-bending state observers or estimators, and a quasi-linear RCS thrust command policy to provide precise control and minimize structural bending excitation. The ACSS hardware mass is very small relative to the 30-year propellant requirement.

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Satellite Antenna

The basic satellite antenna configuration is shown in Figure 30. Three main components comprise the structure—a tension web made from composite wires or tapes, a catenary cable that transfers the web tension to the vertices of the third component which is a hexagonal compression frame. Original analyses of this structure assumed an allowable midspan web deflection of only 0.75 cm. As a result, the structure weight was comparable with other concepts such as the rigid matrix system. Recent work in the microwave subsystem area indicates that midspan deflections of approximately one meter are acceptable with the resulting misalignment being compensated by electronic beam steering.

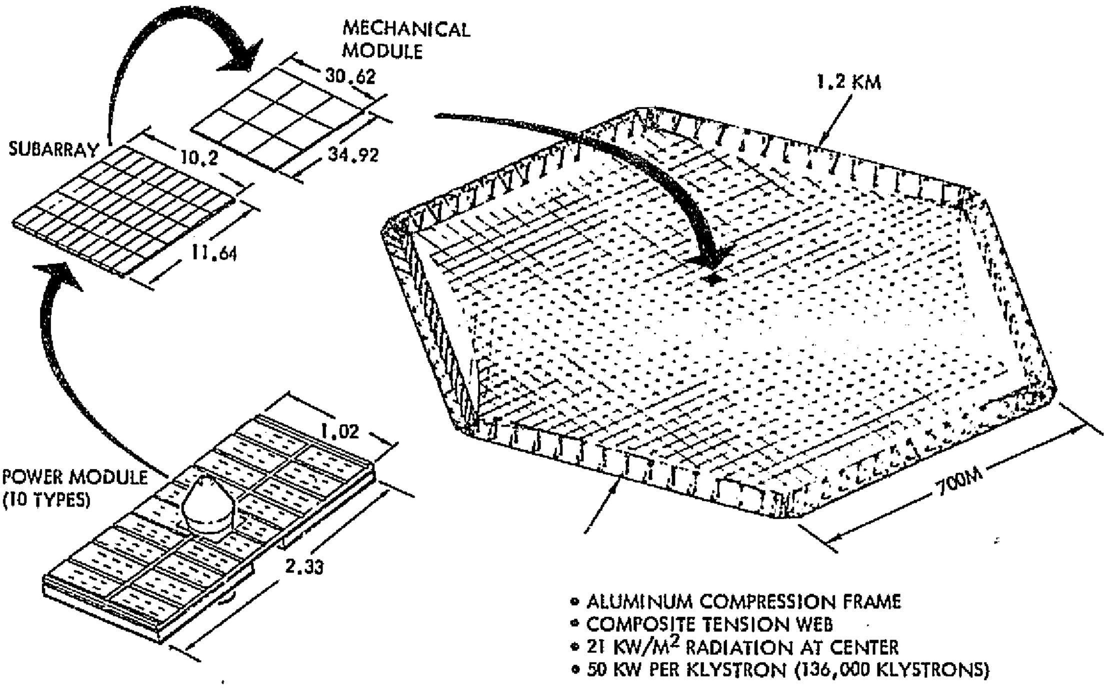

Figure 30. Satellite Antenna

The smallest antenna building block is the power module, which varies in size from the one illustrated (which is used at the center portion of the antenna) to 3.40 by 5.82 meters at the periphery of the antenna. Ten different power module sizes are used to comprise the antenna. Each power module has a klystron located in its center. The power modules are arranged into subarrays measuring 10.2 by 11.64 meters. Each subarray has its own phase control electronics. Nine subarrays are connected to form a mechanical module 30.62 by 34.92 meters. The mechanical modules are attached to the tension webs.

SOLAR THERMAL CONCEPT

The solar thermal concept received considerably less emphasis during the last portion of the study when compared to the photovoltaic concept point design. The overall satellite concept resulting from these studies is shown in Figure 31. Two separate collector modules are used to facilitate gravity-gradient balancing (with a single antenna) and allow the rotary joint to be located at the satellite center of gravity. Each collector module is hinged

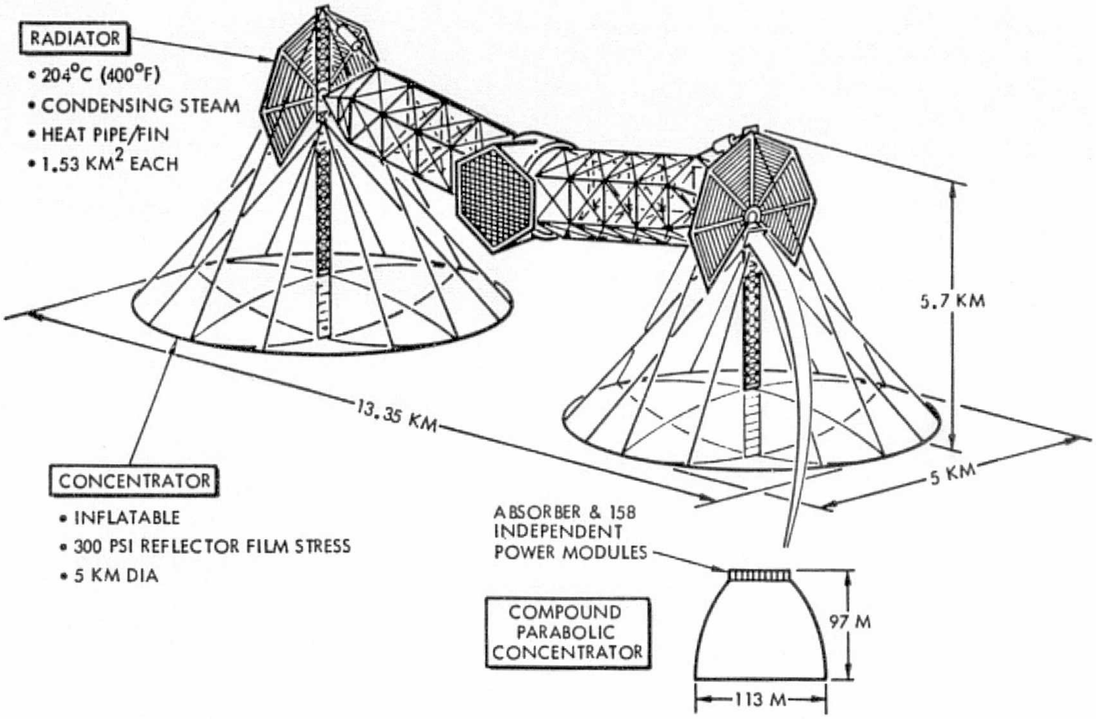

Figure 31. Solar Thermal Design Concept

to the structure to permit seasonal tracking of the sun without affecting gravity-gradient balance. Each absorber has 158 independent power modules utilizing a cesium/steam Rankine cycle. Waste heat is rejected to a condensing steam radiator (heat pipes and fins). The concentrator design utilizes an inflatable parabolic aluminized kapton reflector enclosed on the top by a mylar dome. A 300-psi reflector film stress is sufficient to provide a good reflective surface. A compound parabolic concentrator is employed near the focal point to allow greater error in the large reflectors. Solar energy is concentrated on the absorber which contains the independent power modules.

A relatively small (34 MW) module, shown in Figure 32, was selected to (1) minimize generator and turbine specific weight, (2) reduce development cost, (3) provide economics of mass production, (4) improve SPS redundancy, and (5) reduce boiler manifolding. The loss in turbine and generator efficiency (compared to a 300-MW size) is only 2 percent, while a 65-percent reduction in weight is achieved. Further reductions in size would not benefit SPS cost due to the added instrumentation, control, and monitoring requirements.

Each power module is completely independent and self-sufficient, except for steam and water connections to a common radiator. This allows all leak checks and performance tests of the unit to be accomplished on the ground. Interconnecting cesium lines permit sharing of flow between neighboring boiler panels and turbines in the event of meteorite punctures or turbine/generator failures. Only minor maintenance (modular component replacement) would be performed on station. Up to 10 percent of the units can be shut down (major failures) without affecting SPS output. A complete power module can be replaced by breaking plumbing connections and withdrawing it through the under side of the absorber.

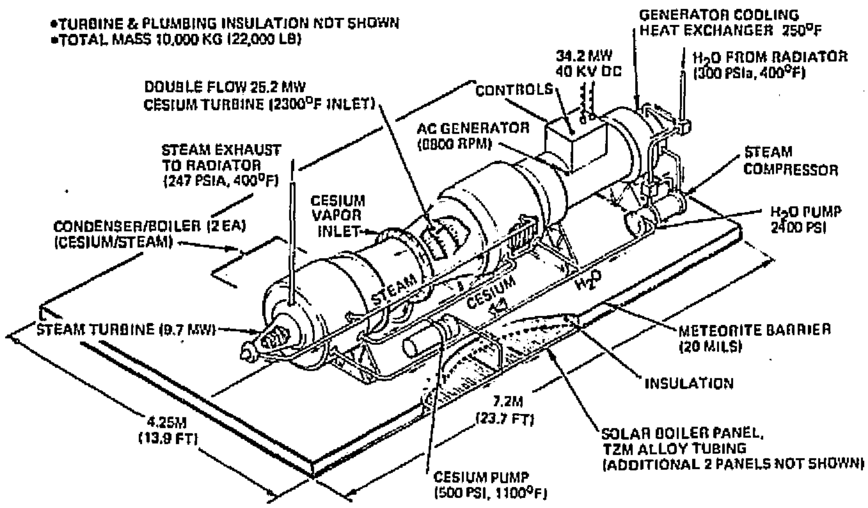

Figure 32. Independent Power Module

A summary of the major design characteristics for the solar thermal SPS is presented in Table 11. The solar thermal system is sized to deliver 5 GW to the utility company on the ground using a single microwave antenna. Each of the subsystems required for a solar thermal SPS is listed, and the major design parameters are shown.

Table 11. Solar Thermal Satellite Design Characteristics

OVERALL DESCRIPTION
- 5-GW POWER TO UTILITY INTERFACE
- GEOSYNCHRONOUS CONSTRUCTION LOCATION
- SINGLE MICROWAVE ANTENNA
- GEOSYNCHRONOUS EQUATORIAL OPERATIONAL ORBIT

SUBSYSTEMS
- POWER CONVERSION
  - CESIUM/STEAM RANKINE CYCLE
- ATTITUDE CONTROL/STATIONKEEPING
  - Y-PCP, X-IOP
  - ARGON THRUSTERS
  - ARTICULATED, SUN-POINTING REFLECTORS
- POWER DISTRIBUTION
  - 40 KV DC
  - STRUCTURE/WIRING NOT INTEGRATED
- MICROWAVE ANTENNA
  - GAUSSIAN BEAM
  - 2.45-GHz FREQUENCY
  - RCR WAVEGUIDE PANELS
  - TENSION-WEB, COMPRESSION-FRAME STRUCTURE
- STRUCTURE
  - ALUMINUM (GRAPHITE/THERMAL PLASTIC COMPOSITE ALTERNATIVE AS NEEDED)
  - BEAM MACHINE CONSTRUCTION
- INFORMATION MANAGEMENT
  - DISTRIBUTED

End-of-life efficiency for the solar thermal concept is presented in Figure 33. All efficiencies include a 30-year degradation allowance, except for the canopy which is replaced after 15 years. Although substantially more than 5 GW could be delivered at the beginning of life, the solar concentrator would be defocused slightly and the turbines run at lower temperatures (at 5 GW) to extend hardware life.

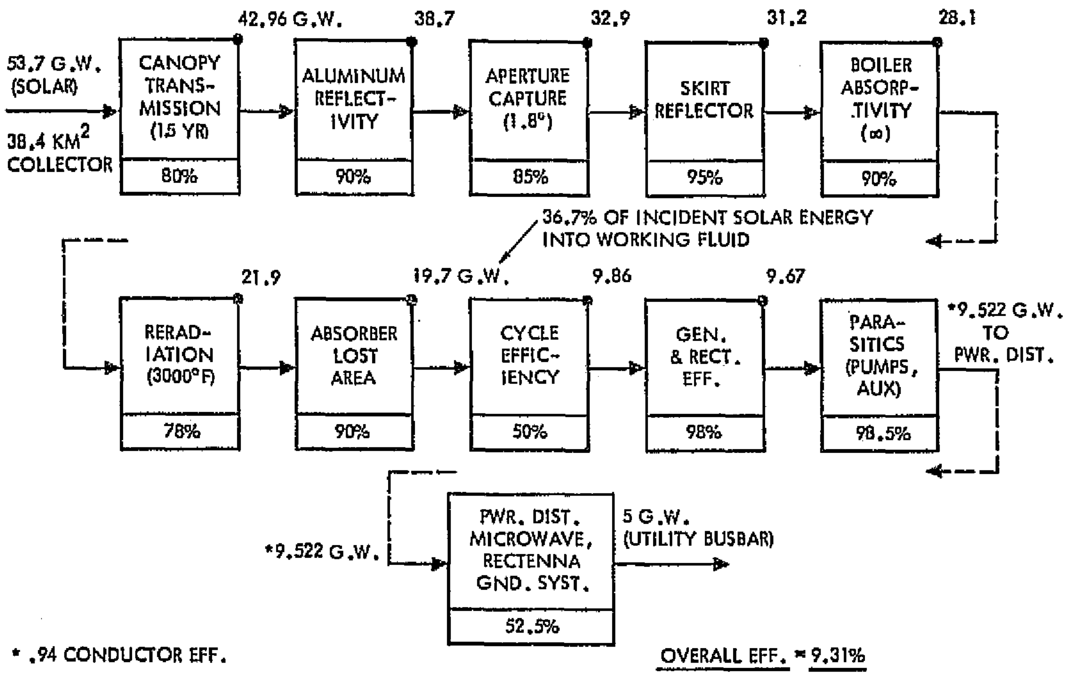

Figure 33. Solar Thermal SPS End-Of-Life Efficiency

A summary of the mass properties for the solar thermal concept is presented in Table 12. The radiator is the major weight contributer of the system. The 43.8-million-kilogram total mass is about 20 percent greater than the photovoltaic point design mass.

Table 12. Solar Thermal Design Mass Properties Summary

SUBSYSTEM	WEIGHT (KG)
STRUCTURE	2.32×10^6 KG
CONCENTRATOR	1.20
ABSORBER	1.60
TURBINES	1.40
GENERATORS	1.10
POWER DISTRIBUTION	1.92
POWER CONDITIONING	2.10
RADIATOR	8.53
ATTITUDE CONTROL	0.10
AVIONICS	0.05
TOTAL (NON-ROTATING)	20.32
ANTENNA SECTION	13.39
TOTAL SPS	33.71
GROWTH	10.11
TOTAL SPS WITH GROWTH	43.82×10^6 KG

RECTENNA CONCEPT

Each rectenna is designed to accept power from a single satellite and provide 5 GW of power to the utility interface. As shown in Figure 34, a typical rectenna site located at 34°N latitude covers an elliptical area 13 km in the north-south direction by 10 km in the east-west direction. This area contains 814 rows of rectenna panels tilted 40 degrees from the horizontal, providing an active intercept area of 78.54 km^2.

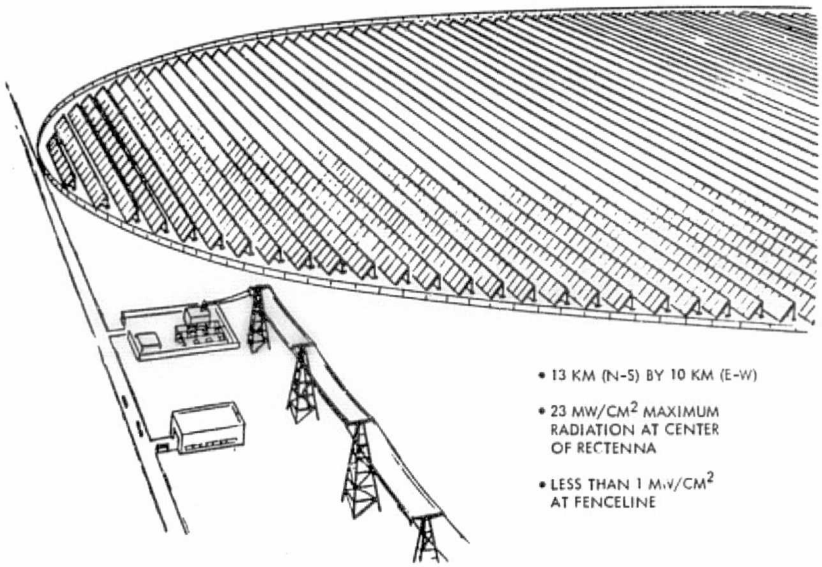

Figure 34. Rectenna Site

Based on the trade studies previously discussed, the phased array comprised of stripline patterns of bow-tie dipoles, shown in Figure 35, was selected. This selection was based primarily on the increased efficiency and decreased diode

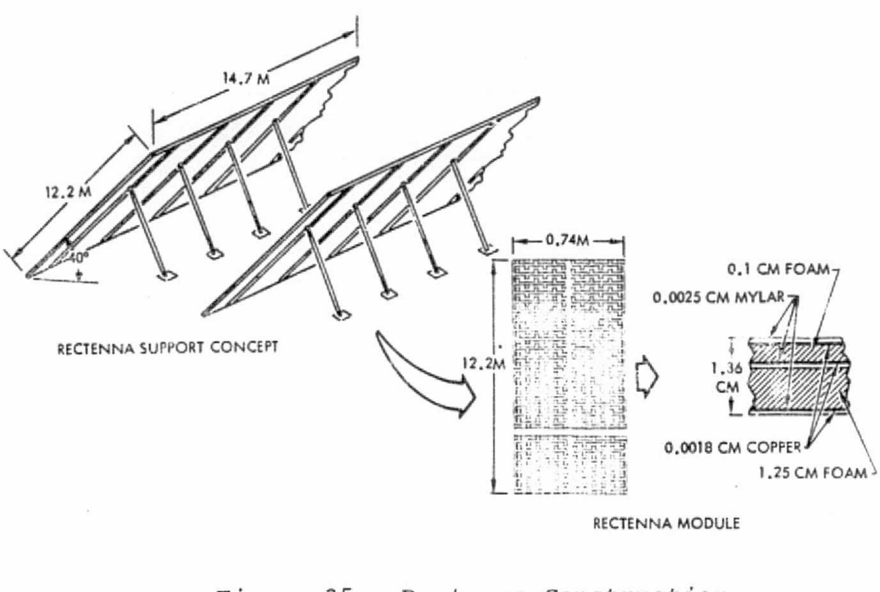

Figure 35. Rectenna Construction

count obtained using this approach. The support concept (Figure 35) will be discussed in the rectenna construction portion of the end-to-end analysis section.

TRANSPORTATION SYSTEM

As previously mentioned, the transportation system selected for the point design included (1) a horizontal takeoff HLLV (one- or two-stage) for earth-to-LEO operation, (2) a dedicated electric ion thruster OTV for cargo transport, (3) a two-stage LO_2/LH_2 OTV for personnel and priority cargo transport, and (4) a single-stage LO_2/LH_2 vehicle for orbit transfer (short distance and small ΔV) of personnel and cargo. Figure 36 illustrates these elements in a low earth orbit scenario.

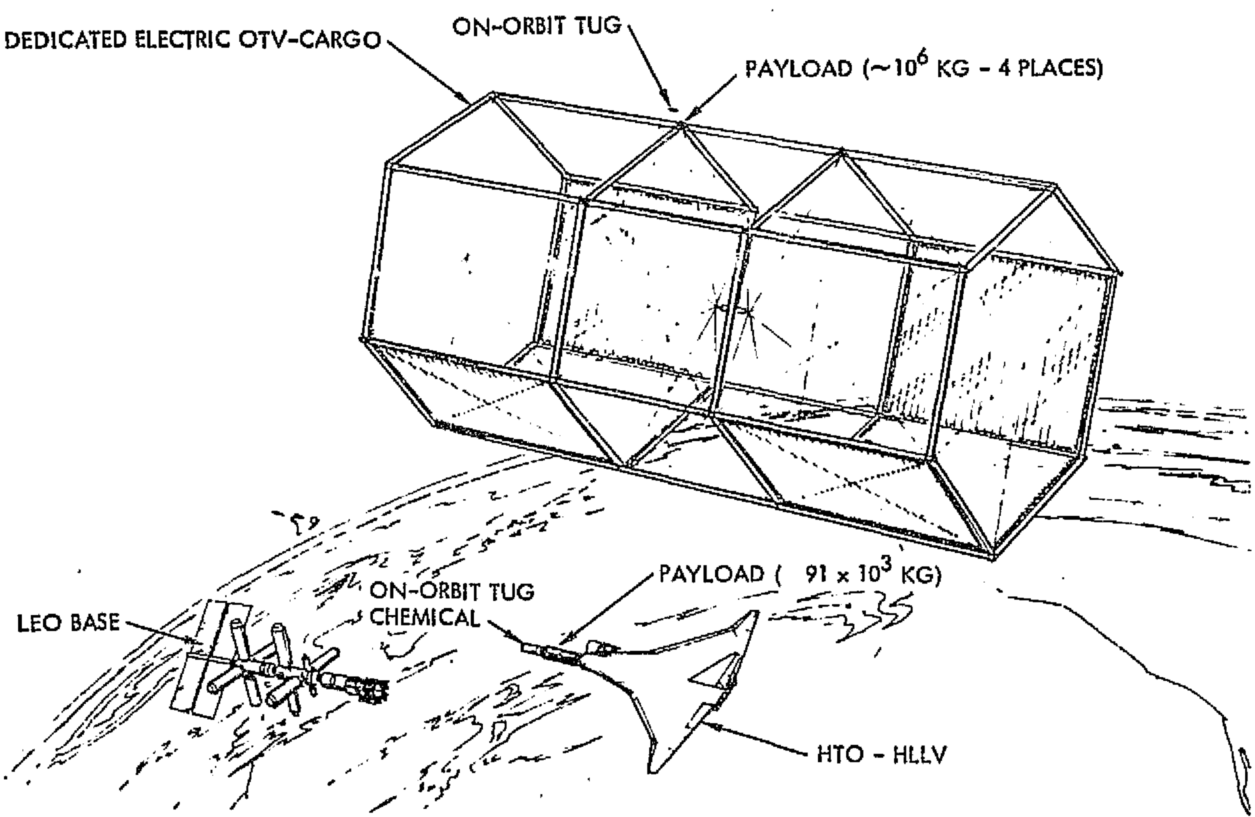

Figure 36. Selected Transportation System Concept

The horizontal takeoff HLLV shown in this illustration is a single-stage concept that carries 91,000 kg into a 550-km geosynchronous orbit. This concept takes off from an airport, using airbreathing engines similar to a supersonic transport, as shown in Figure 37. After climbing to 20,000 feet, Figure 38, the airplane cruises to the equator at Mach = 0.85 and turns into the equatorial plane. Following a climb to 45,000 feet, a dive to 37,000 feet is accomplished to increase velocity to Mach = 1.2. Still using the airbreathing engines, acceleration to a velocity of 6200 ft/s is accomplished. Between

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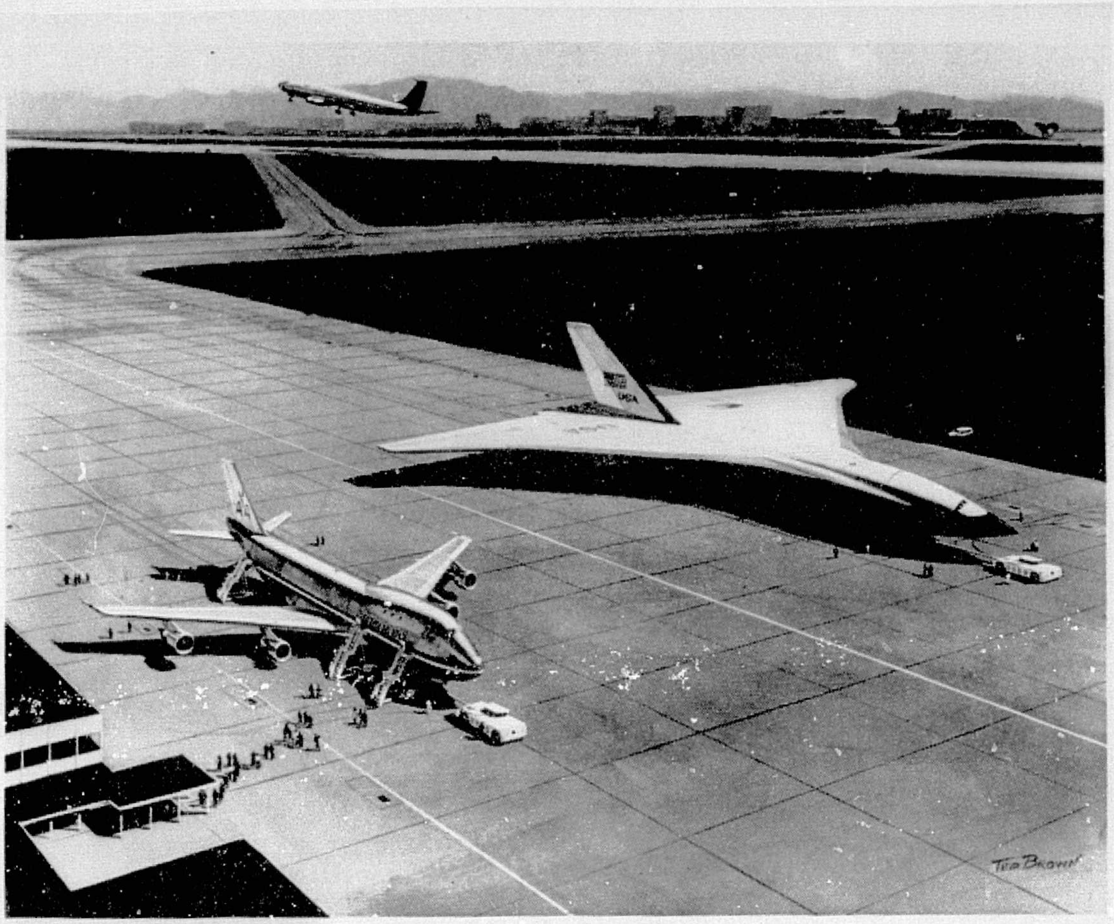


Figure 37. Single-Stage-to-Orbit Concept at Airport

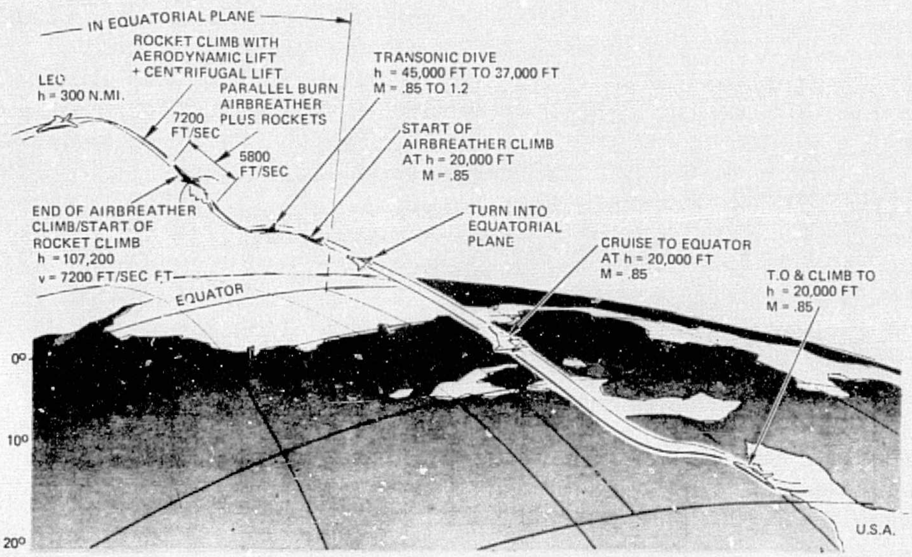


Figure 38. Air-Augmented HTO-SSTO Concept

6200 ft/s and 7200 ft/s, both airbreathing and rocket engines are used for acceleration to an altitude of 107,200 feet. Final injection into low equatorial orbit is accomplished with rocket engines. Some of the design features of this concept are shown in Figure 39. The wings contain propellant tanks to effectively utilize the available volume. A two-stage horizontal takeoff concept is also considered as a viable candidate for SPS; however, this concept has not yet been studied for this application.

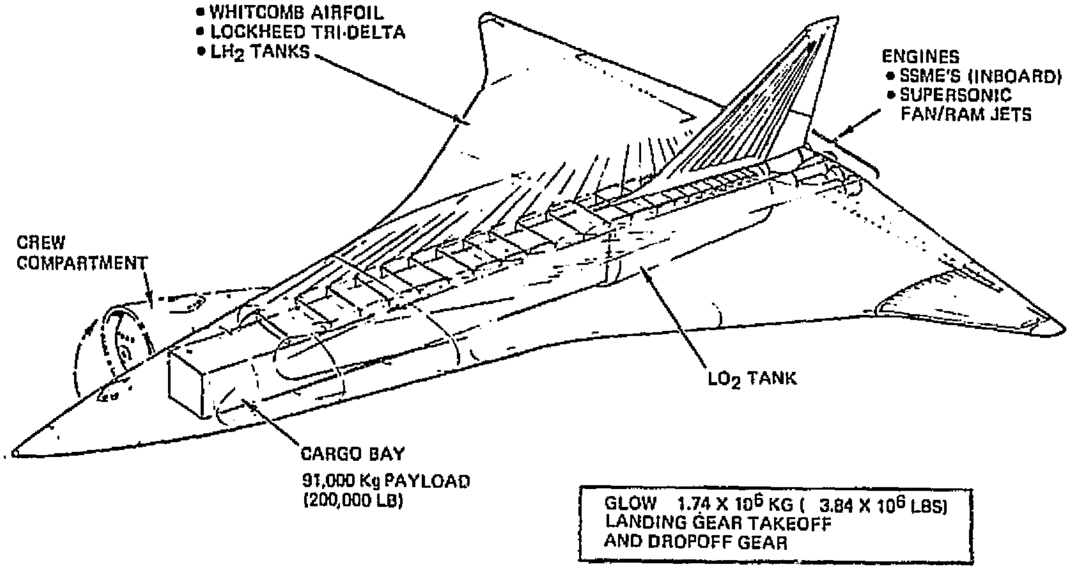
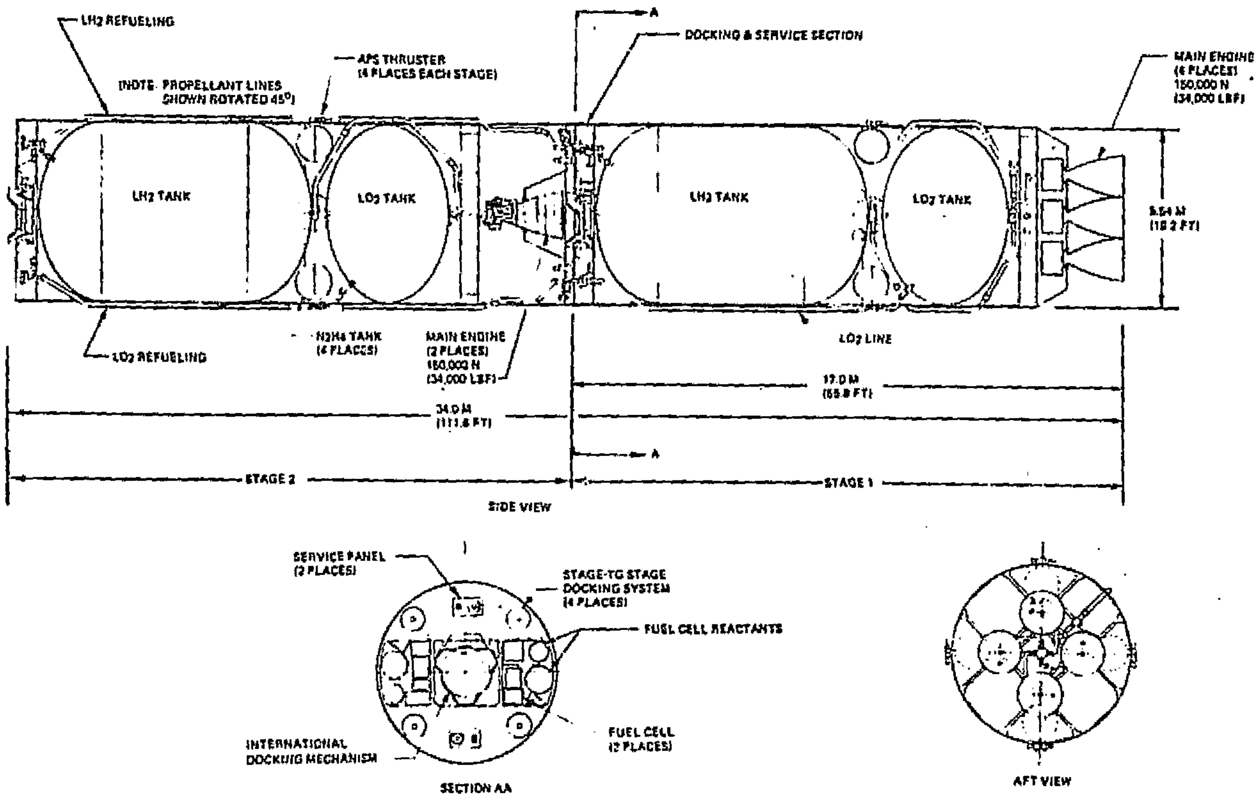


Figure 39. Air-Augmented HTO-SSTO Design Features

The electric argon ion thruster OTV was selected for cargo transfer from LEO to GEO to reduce SPS transportation costs by reducing required propellant mass to orbit. As illustrated (Figure 36), the structure and dimensions of the concept are the same as the center structure of the SPS. The same power approach is used, and it produces 270 MW. The configuration has a concentration ratio of 2, and uses GaAlAs solar cells. The methods for blanket and reflector installation are also similar to SPS. Because of the self-annealing characteristics of the GaAlAs cells at their normal operating temperature, it is expected that the OTV can pass through the earth radiation belt without significant solar cell damage. With an outbound payload of 4×10^6 kg, trip time from LEO to GEO is 133 days. A return trip time of 34 days is accomplished without payload. One OTV flight every five days is required to satisfy mass flow requirements.

A two-stage LO_2/LH_2 OTV, shown in Figure 40, was selected for transfer of personnel and priority cargo. Each of the stages is sized at 91,000 kg gross weight to be compatible with launch in the HTO-SSTO HLLV concept. A payload of approximately 91,000 kg can be transferred from LEO to GEO with this concept.



*REF
BAC FSTSAS CONTRACT NAS9 14323, PHASE I EXTENSION
FINAL BRIEFING, 19 NOVEMBER 1976
EACH STAGE SCALED TO 91,000 KG (200,000 LBS)

Figure 40. Common Stage LO_2/LH_2 OTV Concept

END-TO-END ANALYSIS

The end-to-end analysis of the GaAlAs solar photovoltaic (point design) satellite and rectenna concept was directed toward quantitative definition of key operations associated with major system elements and functions throughout the initial 30 years of the SPS program. The logic flow of this analysis, shown in Figure 41, begins with the SPS point design definition. Within each box, the major design concepts, functional analyses, and systems, facilities, mass flows, and operational requirements considered in the analysis are listed.

The analysis first requires definition of the construction, operational, and maintenance concepts for both the satellite and the rectenna. Time-phased mass flows on earth and in space are derived from these data. Definition of the space transportation system provides the constraints for development of the cargo packaging concept which, together with the satellite production rate and construction concept, forms the basis for development and/or definition of other system elements and operations. These include development of the space traffic model; definition of operations and facilities at LEO; and definition of the launch complex operations, mass flows, and facilities.

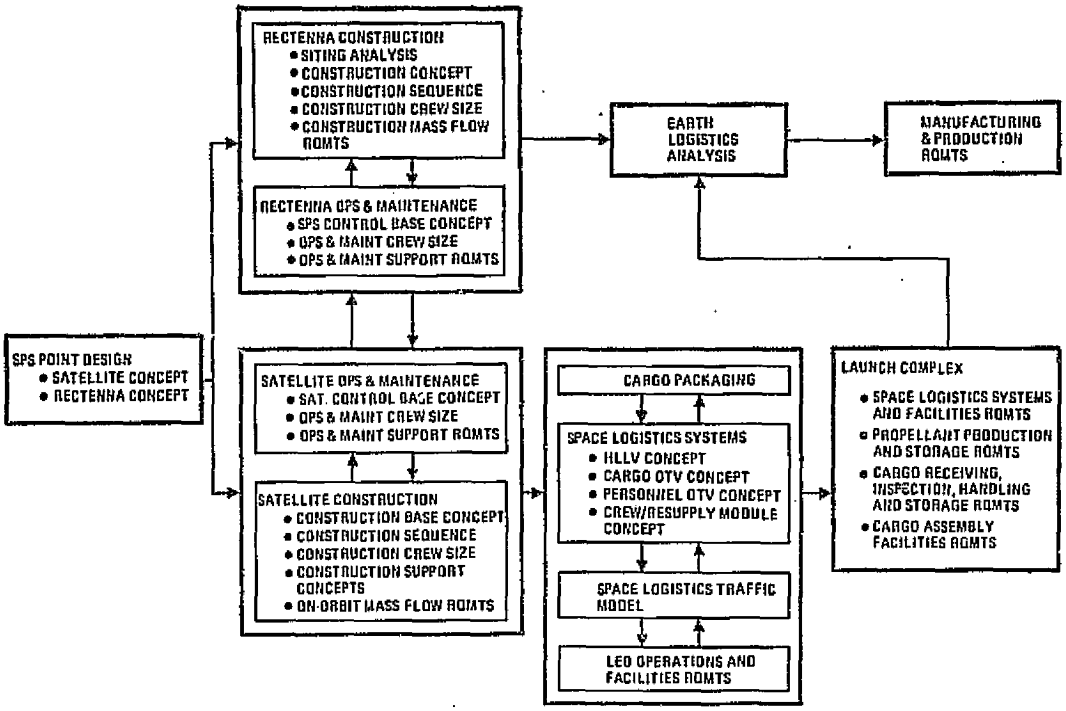


Figure 41. Major Tasks of the End-to-End Analysis

Rectenna site construction, operations, mass flows, and facilities are established in a similar manner. Earth manufacturing requirements, location of manufacturers, and earth logistics concepts are then defined to provide the best support of the satellite and rectenna construction and maintenance requirements.

The following sections summarize the most significant results of this analysis including satellite construction, orbital base concept, mass flow to orbit, propellant production, rectenna construction, and earth logistics.

SATELLITE CONSTRUCTION

The entire satellite is constructed in geosynchronous orbit. Construction equipment and materials are transported from LEO to GEO on a solar-electric propulsion vehicle previously described. Men are transported between LEO and GEO on a two-stage LH_2/LO_2 rocket-propelled vehicle.

The sequence and schedule for satellite construction are shown in Figure 42. A single integrated construction facility builds the structure and installs the solar blankets, reflectors, power distribution system, and other subsystem elements located in the wings. Construction starts with one wing tip and progresses toward the center section where the rotating joint for the microwave antenna will be located. Construction then continues outbound, building wing number 2, and terminating at that wing tip.

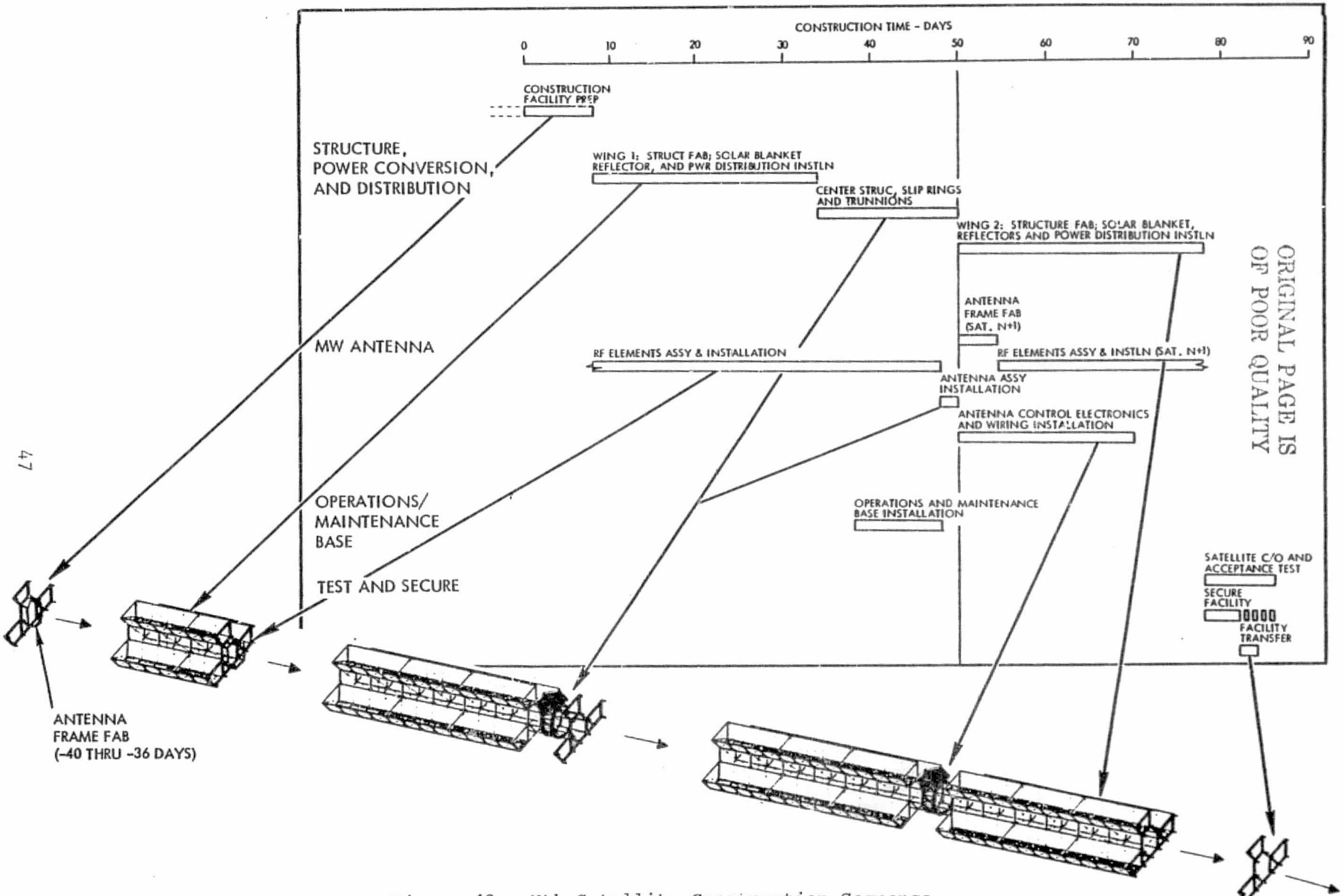


Figure 42. Nth Satellite Construction Sequence

The first eight days are designated for preparation of the construction facility. Prior to the eighth day, sufficient materials have been delivered to satisfy the first several days of consturction—primary structural material (beam machine cassettes) for half of the satellite; solar blanket and reflector rolls, electrical conductors, and switch gear for the first two bays; and microwave antenna components. Since the rear side of the facility is always exposed to space with no interference from the main construction activities, it is used as the jig for building the microwave antenna frame and as the location for assembly and installation of the 30×30-m microwave subsystem mechanical modules. Fabrication of the microwave antenna for this Nth satellite was started on the 50th day of construction of the previous (N-1) satellite and is continued up through the 48th day of construction of this satellite. At that time, it is ready for installation on the slip ring mounted trunions.

Each satellite wing consists of 12 bays which are 800 m long. These are constructed at the rate of one every two days using three 8-hour shifts per day. The structure and installation of the power conversion system of Wing 1 is completed on the 34th day. While Wing 1 construction is taking place, the microwave antenna crews are proceeding with the assembly, test, and installation of the antenna elements into the antenna frame. The antenna assembly continues during the construction of the center station.

Subsequent to completion of Wing 1, the construction facility constructs the longerons and frames in the center section, installs the slip rings, constructs the tension supports, installs the trunions, and installs power wiring in the center. Although 16 days are scheduled for this activity, the timeline requires only 12 days with two additional days scheduled for transfer of the antenna to the trunion mounts. Two days are allowed for contingencies.

Immediately upon completion of the center section primary structure, the facilities for the operation and maintenance base are installed and the first operational maintenance crew arrives to support installation of the antenna control electronics and satellite checkout, which takes place from Day 50 through Day 69.

By the 51st day, all satellite hardware has been delivered. On-site logistics activities are therefore greatly reduced, freeing construction support personnel for subsystems hookup and checkout during the Wing 2 construction period.

Use of the construction facility is completed on Day 78, and flyaway transfer to the construction site of the next satellite occurs on Day 84. Final satellite checkout and acceptance testing is completed on Day 86.

Figure 43 shows a perspective view of the tribeam construction facility. The facility is configured to restrain the free end of each cross-frame member as it is fabricated. After completion of each 800 m of longitudinal members, the construction facility is stopped, the cross-frame complexes are translated to their offset positions, and the cross-frame members are completed and joined to the longitudinals. A more detailed description of this facility and the approach for total satellite construction is contained in Volume V.

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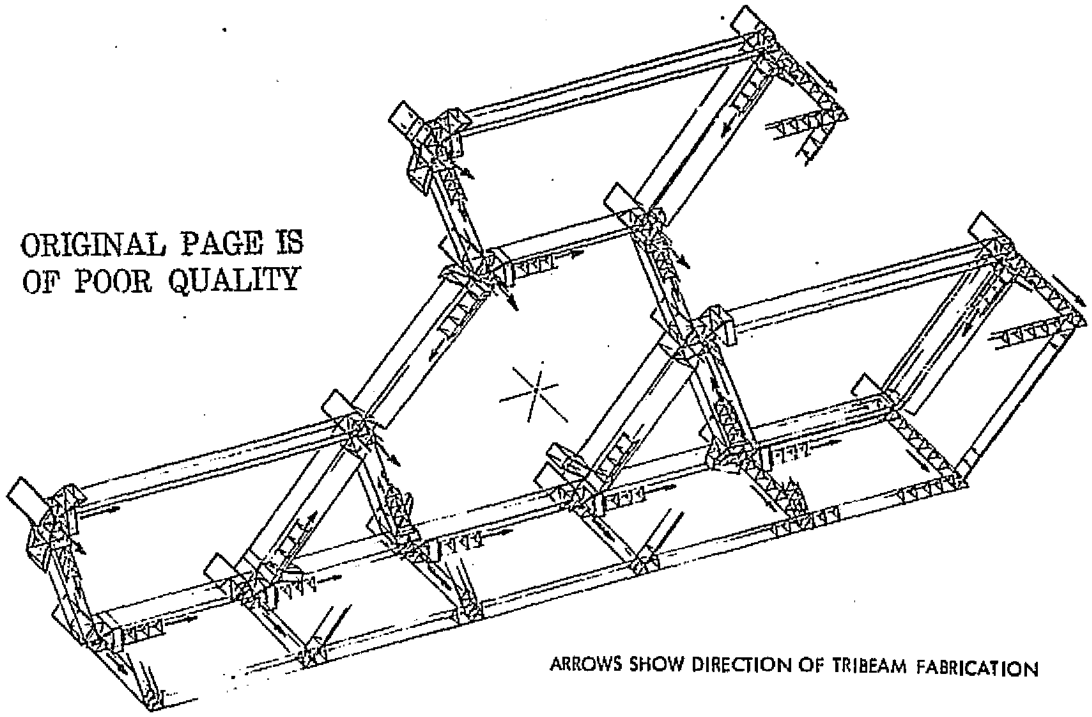


Figure 43. Satellite Tribeam Construction Jig

ORBITAL BASE CONCEPTS

Three orbital bases have been identified to support satellite construction, satellite operations and maintenance, and low earth orbit logistics. These concepts are described below.

<u>Satellite Construction Base</u>

Construction of the satellites takes place in GEO at its designated operational longitude. The concept for the GEO construction base is illustrated in Figure 44. Construction is accomplished almost entirely from the single assembly and fabrication fixture shown in the left side of the figure. A crew size of 640 has been established for accomplishing the construction in the scheduled time. The crew and their facilities are divided equally and are

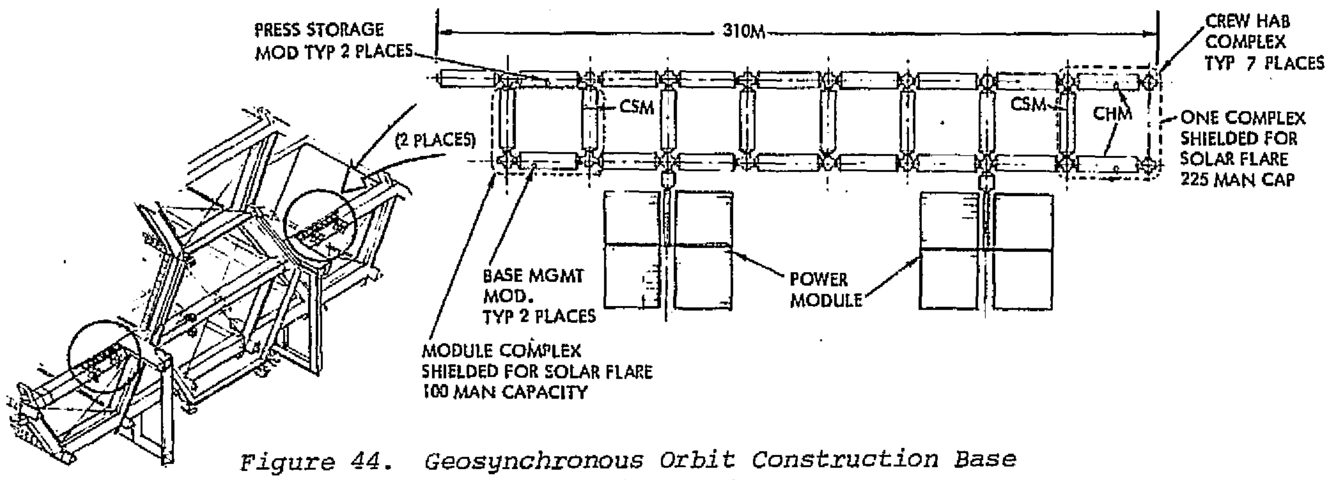


Figure 44. Geosynchronous Orbit Construction Base (Crew Size: 640)

located on each side of the hexagon portion of the fixture. One of these 320-men bases, shown in the figure, consists of 7 three-module crew habitability complexes plus two base management modules, two pressurized storage modules, and solar array power modules.

The modules of the crew habitability complex are described in more detail later (Figure 46). Each complex is composed of two crew habitability modules, each of which provides staterooms, personal hygiene facilities, and support subsystems for 24 crew members; and one crew support module which provides galley, recreational and medical facilities, and subsystems for the 48 crew members of the two crew habitability modules. Base management modules house the communications and control systems for the base and the construction facility. The pressurized storage modules include workshops for maintenance of construction facility elements and satellite hardware as required.

Seven of the modules (indicated by the dashed lines) are hardened against solar flare radiation and serve as temporary quarters for the entire crew when the base is subjected to that environment.

Operations and Maintenance Base

Figure 45 shows the permanent operations and maintenance base which is installed on each satellite prior to completion of construction. This base is located near the center of the satellite for best access to all parts of the satellite, and is installed subsequent to completion of the center structure as described in the discussion of the construction schedule. A maintenance crew of 20 has been estimated. The functions of the five modules which comprise the base are identified in the figure. The crew habitability module internal configuration is the same as for the construction base. The crew support module also has the same internal functions as the construction base, but occupies only half of the module, the other half being an integrated multi-crew member EVA preparation and airlock station.

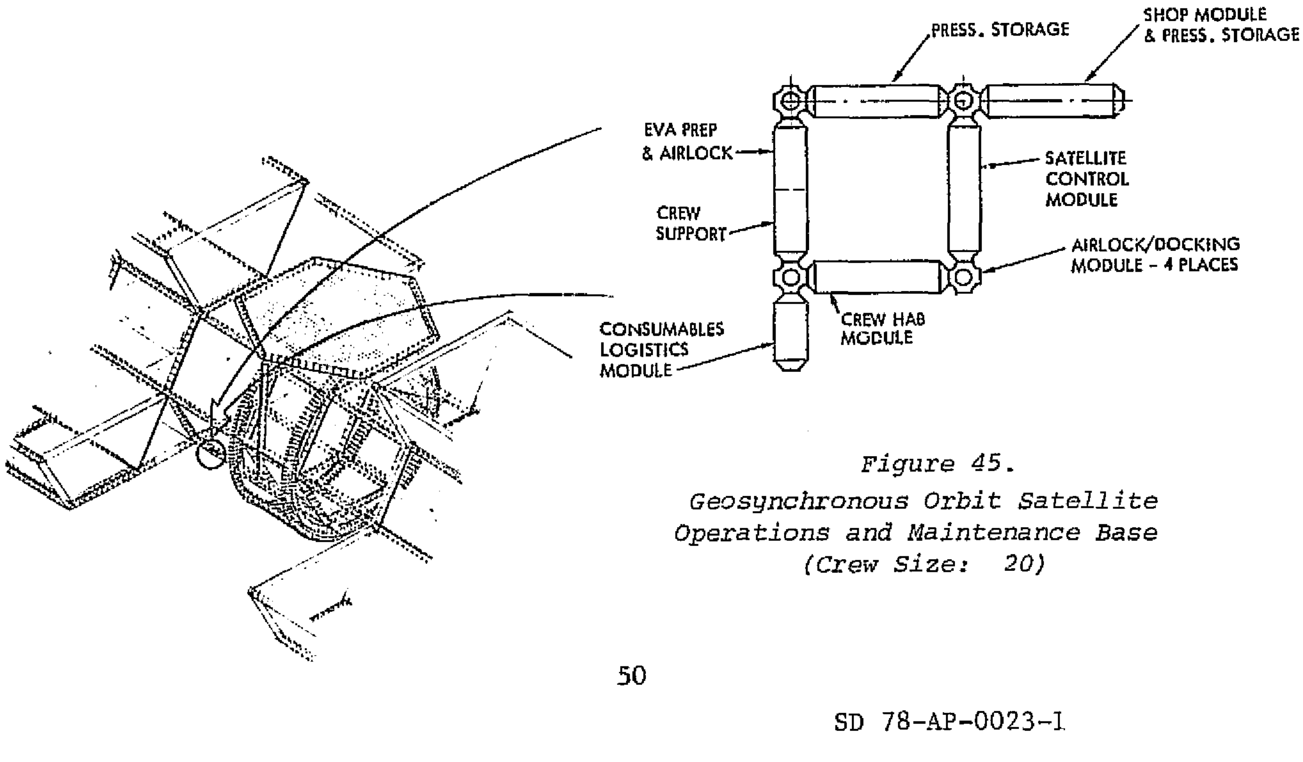


Figure 45.
Geosynchronous Orbit Satellite Operations and Maintenance Base (Crew Size: 20)

Low Earth Orbit Base

The LEO base personnel provide supervisory activities for transfer of up and down payloads between the HLLV and the OTV's and perform the scheduled maintenance required by the COTV (changeout of thruster screens). Figure 46 illustrates the concept for this base. It has one crew habitability module and one crew support module of the same configurations as the GEO construction base, except that six of the 30 staterooms are located in the crew support module. Direct transfer of crew and equipment between the HLLV and the OTV's is planned; however, multiple docking ports and excess subsystems capacity and power are provided for emergency staging support.

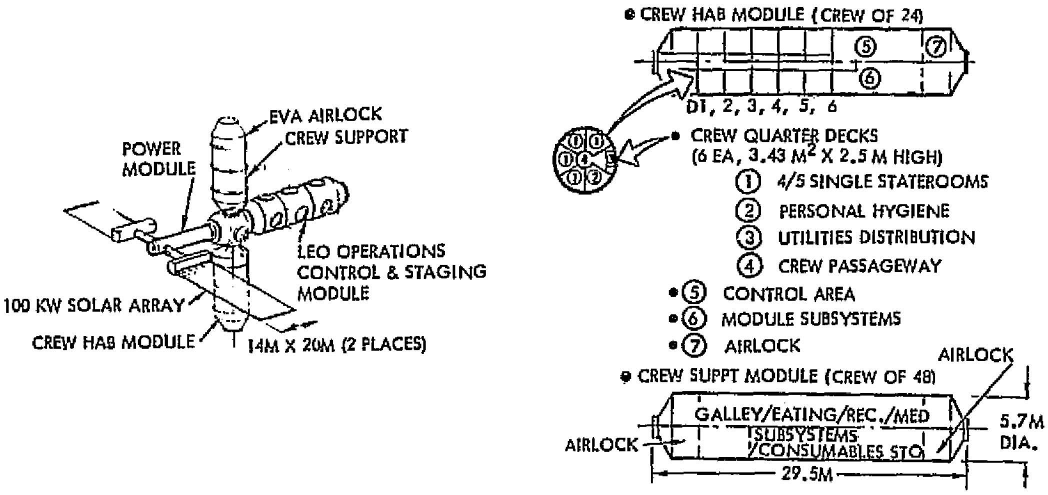


Figure 46. Low Earth Orbit Base (Crew Size: 30)

All base modules are of the dimensions shown for the crew support module. Module size and mass are designed for compatibility with HLLV cargo-carrying capability.

MASS FLOW TO ORBIT

Satellite mass flow requirements, categorized by major subsystems to support the construction schedule of one satellite, is shown in Figure 47. The initial mass requirement can be accommodated on one COTV and would be scheduled to arrive at the GEO site during the construction facility preparation, which occurs during the eight days prior to commencement of satellite construction. The schedule requires that all material be delivered in 72 days. However, it is planned to construct the antenna frame and commence installation of antenna components for the next SPS during the latter part of the construction schedule. Material delivery to support this construction is shown on the microwave antenna and rotary joint line, which extends past the 72 days.

A total of 409 HLLV flights is required to transport 37.2×10^6 kg, representing the mass of one SPS, to LEO. Ten different payload mixes, averaging 91,000 kg each, have been defined and sequenced to support construction needs. An HLLV launch schedule of eight flights per day has been postulated and is

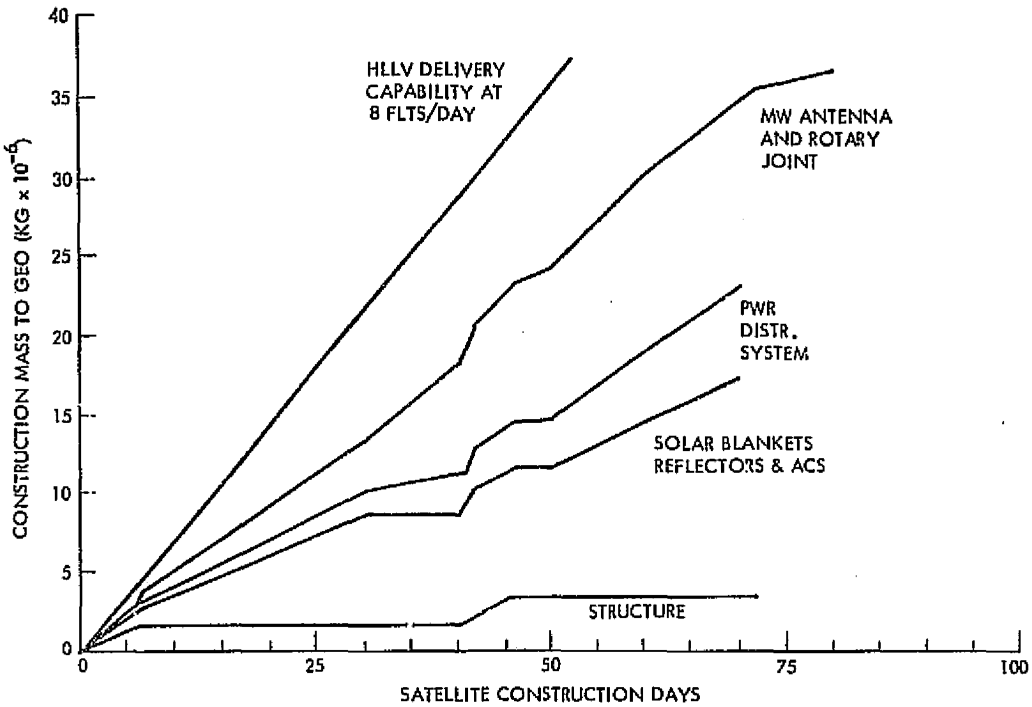


Figure 47. Mass Flow Demands for Satellite Construction

shown as the top line of the figure. The schedule is within the projected launch rate capability, considering other requirements such as maintenance material and crews. This results in total SPS mass delivery in 51 days—21 days ahead of the required completion—thus providing considerable margin for contingencies which could slow delivery rate.

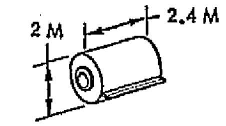
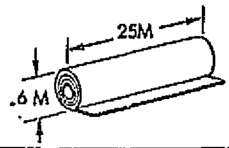
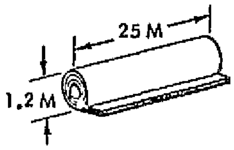
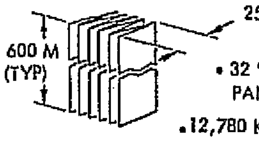
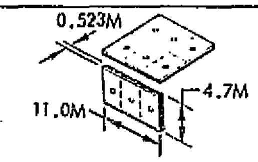
An analysis of cargo packaging was conducted to assure that the construction materials can be properly packaged in quantities consistent with construction requirements and in packages that fully utilize the payload weight capability of the HLLV, while not exceeding the volume constraints. Table 13 illustrates packaging concepts for major elements of the satellite. These package configurations, sizes, and specified quantities per satellite are designed for compatibility with the satellite construction concept and construction equipment described earlier.

The primary structure cassettes simultaneously feed each beam machine to produce the basic 2-m triangular beam elements used in construction of the 50-m girders. All cassettes contain sufficient material to complete half of the satellite structure, thus requiring replacement only once during satellite construction.

Each solar blanket roll is 750 m long—the length required for one bay. For a 600-m-wide bay, 22 of these 25-m-wide rolls are mounted side by side in the blanket layer and deployed simultaneously. End and side attachment materials and hardware are packaged separately.

The reflectors are 600 m wide and 800 m long per panel when deployed. When packaged, the reflectors have an accordian-fold 25 m wide. The resulting 25×600-m strip is then rolled for packaging as shown.

Table 13. Cargo Packaging

SPS ELEMENT	PACKAGING	PACKAGE DIMENSIONS	NO. REQUIRED	NOTES
STRUCTURES	CASSETTES OF ALUMINUM TAPES		1188	6 DIFFERENT TAPE LENGTHS 2500 KG AVE MASS
SOLAR BLANKETS	ROLLS		1632	750 M LENGTH/ROLL 7136 KG/ROLL
REFLECTORS	ROLLS OF FABRIC-HINGED ALUMINIZED KAPTON SHEET		144	 • 32 "HINGED" PANELS •12,780 KG/ROLL
MW ANTENNA WAVEGUIDE PANELS	SUB ARRAYS		6993	• ALL SUBARRAYS HAVE SAME OVERALL DIMENSIONS • 10 DIFFERENT POWER MODULE SIZES - QUANTITY VARIES WITH SIZE • SUBARRAY MASS (AVE) = 716 KG

The 6993 waveguide panels are the lowest density payload item and, therefore, become a major driver in packaging and scheduling payloads. Based on the average shipping dimensions and mass for each panel given on the table, a maximum of 22 panels for a total mass of 15,750 kg can be carried in the HLLV cargo bay.

In addition, klystrons (which do not present a packaging problem) are a major payload item. The microwave antenna contains a large number of subarrays that, in turn, are composed of up to 50 power modules. Each power module has a klystron which is shipped to GEO separately and inserted after the subarray has been secured to the antenna. Each klystron has an average volume of 0.092 m^3 and weighs 45 kg; 135,864 are required for each satellite.

PROPELLANT PRODUCTION

For each kilogram of mass to orbit, 16 kg of propellant are needed for the HLLV, assuming the use of a horizontal-takeoff, single-stage-to-orbit concept. Of this propellant, the mass ratio of oxygen to hydrogen is 1.7:1. Although oxygen comprises the greatest mass, production of hydrogen presents the greatest problem.

Several processes were considered for the production of hydrogen. Of these, coal gasification and electrolysis of water appeared to warrant in-depth analysis. These processes were analyzed to assess relative production costs and, for coal gasification, the method of transport of coal or hydrogen to the launch site. It is assumed that electrolysis can be accomplished in the launch site area, negating the need for long-distance transportation.

The major results of this evaluation are presented below in Figure 48 and Table 14. Figure 48 compares the costs of liquid hydrogen delivered at the launch site. Little difference in cost exists between producing the hydrogen at the coal mining site and shipping hydrogen to the launch site versus shipping coal (as slurry or on a train) to the launch site and producing hydrogen at the launch site. Electrolysis costs (assuming 10-mil/kW-h) are about twice the cost of coal gasification. As shown in Table 14, electrolysis has other advantages, of which environmental considerations are most important.

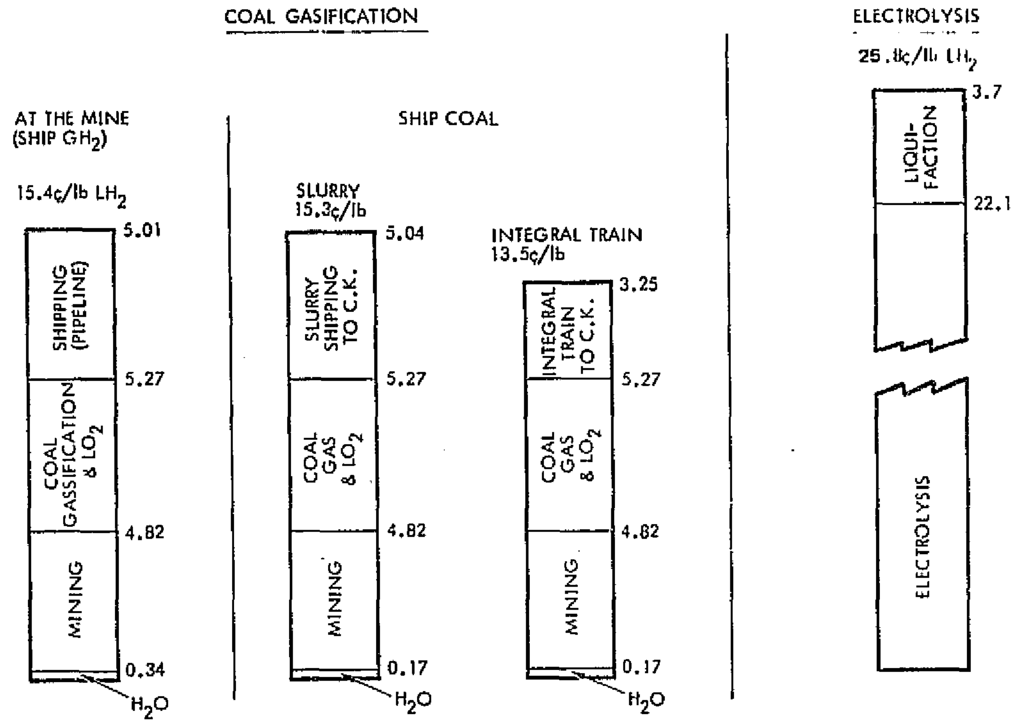


Figure 48. Preliminary Propellant Production Cost Comparisons

Table 14. Comparative Summary

	POSTIVE FACTORS FOR:	
	COAL GASIFICATION	SPS ELECTROLYSIS
• ENERGY REQUIRED	X	
• COSTS	X	
• ENVIRONMENTAL CONSIDERATIONS		X
• TRANSPORTATION REQUIREMENTS		X
• HYDROGEN ECONOMY		X
• MULTI-PURPOSE FACILITY		X

These data need to be incorporated into an economic assessment to determine impacts on total transportation costs. The synergism of the SPS concept and electrolysis would appear appealing if the overall cost impacts are not significant.

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RECTENNA CONSTRUCTION

Major elements of a 5-GW rectenna site located at approximately 34°N latitude are depicted in Figure 49. In order to minimize electrical wiring from the rectenna panels, two electrical switchyards are employed, each with its own converter and relay building. A rail spur line would be utilized, predominantly for the construction phase, to bring in gravel for the access roads and concrete plants and transformers for the switchyards. The four concrete plants would be removed after serving their function. The rectenna farm of 10×13 km contains 814 rows of rectenna panels tilted 40 degrees from the horizontal, providing an active intercept area of 78.54 km^2. Since an individual panel is 12.24×14.69 m, some 436,805 panels have to be assembled on site and erected.

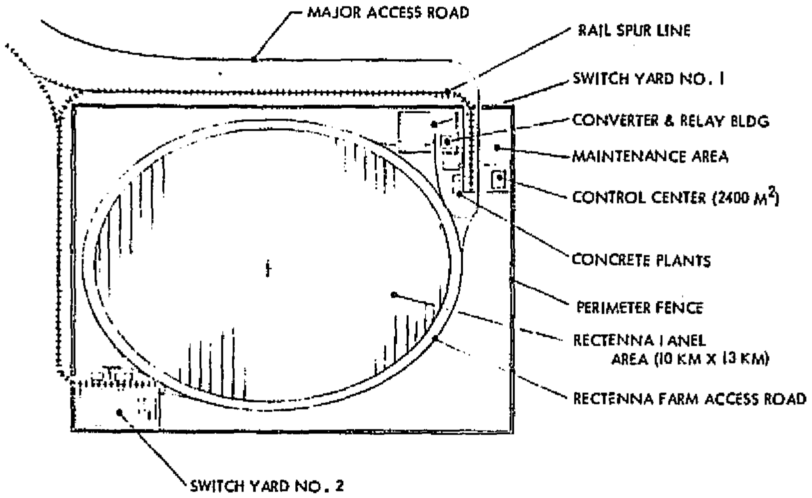

Figure 49. Rectenna Site Elements

Since the selected rectenna panel concept is comprised of solid sheets 1.35 cm thick, they are relatively insensitive to weather. However, because they are solid, high wind loads (up to 90 mph) must be considered in the construction of the support structure. Overall array deflections must be less than 3 cm. The support structure shown in Figure 50 employs preformed hat sections, standard I-beams, and 3.5-inch-diameter tube traces. The I-beams and braces support the structure on concrete piers.

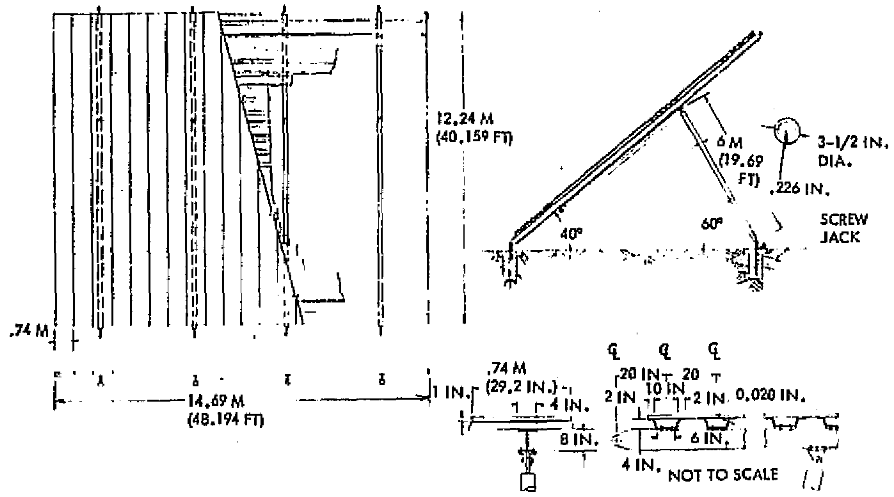

Figure 50. Rectenna Array Support Structure

Assembly of rectenna panels represents the major construction time challenge. The large numbers dictate the need for an assembly and erection concept like that shown in Figure 51. Fundamentally, the concept is a mobile assembly jig which, after having completed its share of the construction operations, can be disassembled and transported to another rectenna site. The assembly jig can be loaded to contain 10 sets of rectenna panel elements. Since each set weighs 2200 kg (~4800 lb), the 10 sets can be delivered to the jig on a single flat-bed truck. After the truck crane lifts off a completed rectenna panel (see insert) from their loaded locations at the side and end of the jig, the hat sections and I-beam tube braces are conveyed into place. Stops are used to assure

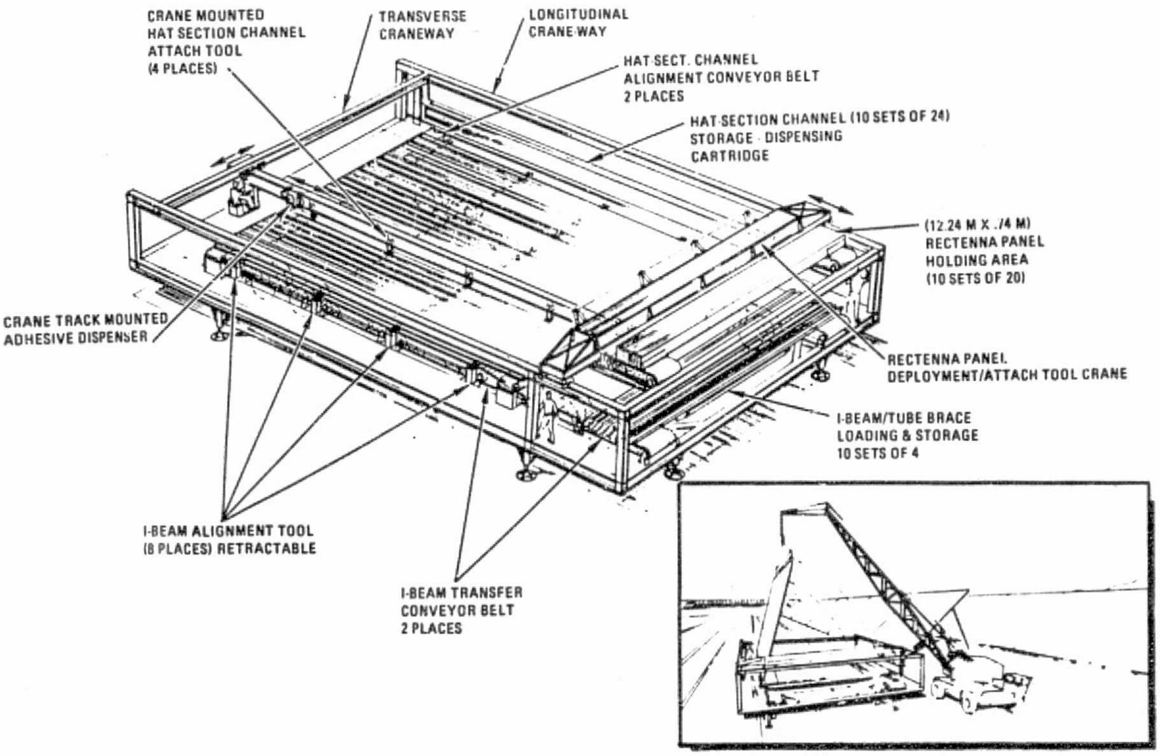

Figure 51. Rectenna Panel Assembly Concept

exact positioning and alignment. The manned truck-mounted crane unit then passes over the jig, securing the hat sections to the I-beams and laying down the adhesive for the rectenna panels. These operations consume approximately 21 minutes. Next, the rectenna panel crane moves longitudinally across the jig, placing each of the twenty 0.74-m-wide panels onto the completed structural frame. A geared eccentric roller on this crane provides the pressure to secure the rectenna panels to the frame. Wiring harnesses are then installed and the hoist sling is attached from the truck crane for removal of the completed unit. Ten array panels could be assembled in one eight-hour shift, but the number of assembly jigs is based on one assembly per hour. Installation of the completed panel on concrete piers is estimated to take about 20 minutes. One truck crane and installation crew should be able to work with two assembly jigs at a time.

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Impacts of rectenna siting alternatives on ease of integration into the power grid, power management, and cost have not yet been accomplished. This analysis requires an interaction between NASA and DOE activities. NASA must define the alternatives for rectenna locations, and DOE then must determine the impact of alternatives on the power grid. Three apprcaches to rectenna location that should bracket the possibilities are: (1) distributed, but near the load centers; (2) regional clusters; and (3) single location for all rectennas.

EARTH LOGISTICS

Two major areas of logistics were considered—logistics at the launch site and logistics at the rectenna site.

Launch Site Logistics

Launch base facilities must provide for (1) receiving, storage, and processing of material and propellants; (2) storage of HLLV's sufficient for initial operations; (3) refurbishment and checkout of returning HLLV's; and (4) personnel handling and administration. Figure 52 indicates the required facilities and shows the interfacility relationship for material and personnel processing.

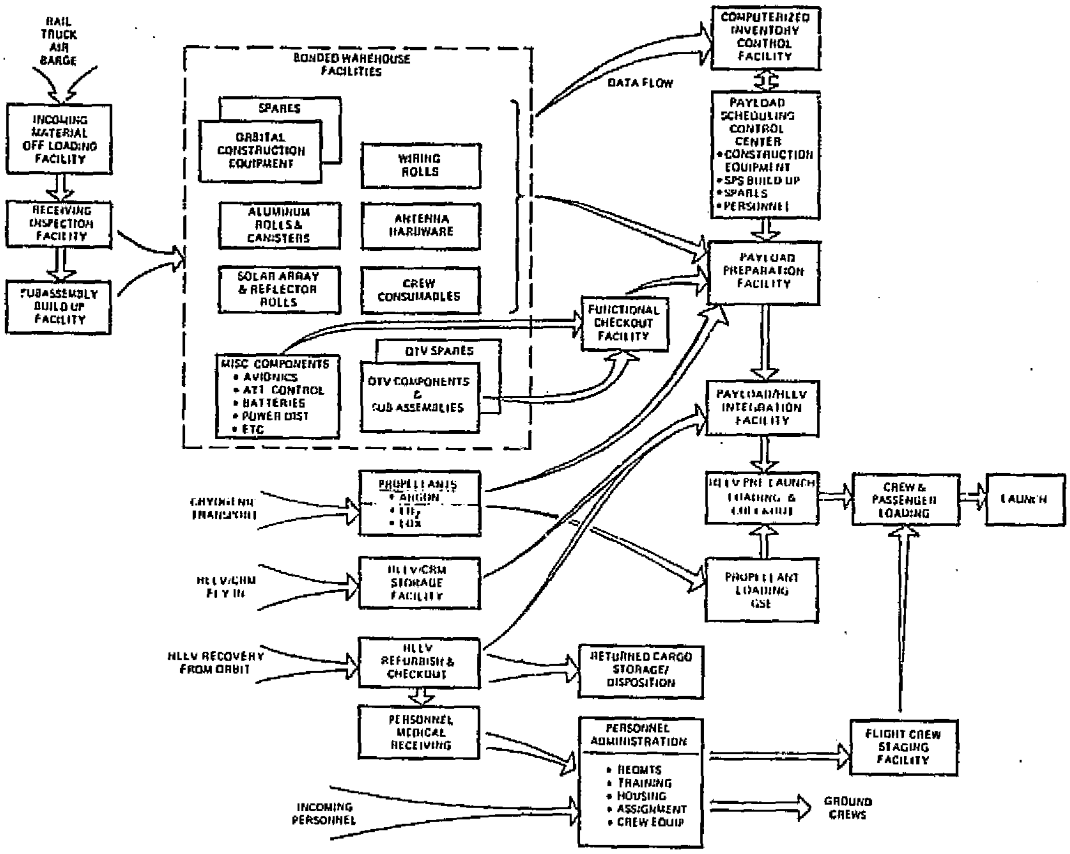

Figure 52. Launch Site Logistics

Incoming material (rail, air, etc.) is offloaded, subjected to receiving inspection, entered into the computerized inventory control system, and then stored in the appropriate warehouse facility. The maximum rate of incoming railroad traffic required to support construction and maintenance of space-based program elements is approximately 5500 cars annually (propellants not included) and occurs in the 30th year of the program.

Bonded warehousing is provided for SPS construction material and spares and for COTV construction material. An initial requirement of 190,000 m^2 of floor space, ultimately increasing to 267,000 m^2, has been established.

When scheduled by the Payload Scheduling Control Center, material (construction material, consumables, and spares as required) is transferred to the payload preparation facility for packaging and arrangement into payload units on 6×30-m pallets. Electronic modules and other selected components are functionally tested prior to packaging. The packaged payloads are then transported to and loaded on the HLLV prior to propellant loading and final HLLV checkout. Personnel comprising part of a payload will enter their crew module in the payload bay shortly before launch.

The propellant storage facilities provide for cryogenic storage of HLLV propellants and for argon which will be shipped to low earth orbit for COTV utilization.

Since initial HLLV flight requirements are substantial (approximately 550 flights during the first year), a fleet buildup will be required prior to initiation of the orbital phase of the program. A facility to store HLLV's during the buildup period must be provided. This facility also will serve as a refurbishment area for HLLV's returning from orbit.

Personnel administration and planning is a vital element in the overall base function. Incoming personnel must be trained and assigned to either flight or ground crews. Personnel returning from 90 days in orbit must undergo medical processing and then be reassigned to ground activities before returning to orbit. The continuous growth in number of both base personnel and space crews throughout the 30-year program precipitates the requirement for extensive facilities for medical, housing, training, and administration.

Finally, provision is made for processing and disposing of large amounts of packaging materials, and failed and damaged hardware which will be returned from orbit by the HLLV's.

Rectenna Site Logistics

In order to meet the rectenna site construction schedule, construction masses must be supplied to the assembly and support equipment at rates which meet or exceed their demands. These mass flow demands are shown in Figure 53 by material type and as two types of demand: delivery-to-the-site demands, and intra-site demands. Delivery-to-site requirements are lower since preconstruction buildup will allow (overall) approximately 12 months for satisfying these logistics demands. Intra-site requirements (for the same total masses) must be accomplished over a nine-month period. Approximately 390 truck trips/day must be handled at the site. In terms of vehicle flow on a good highway, this is a relatively modest demand; but at the site, approximately 40 unloading docks will be required to handle the traffic. Although the daily intra-site mass flow demands are higher, they are more easily handled since a truck at the site can make a number of short trips per shift.

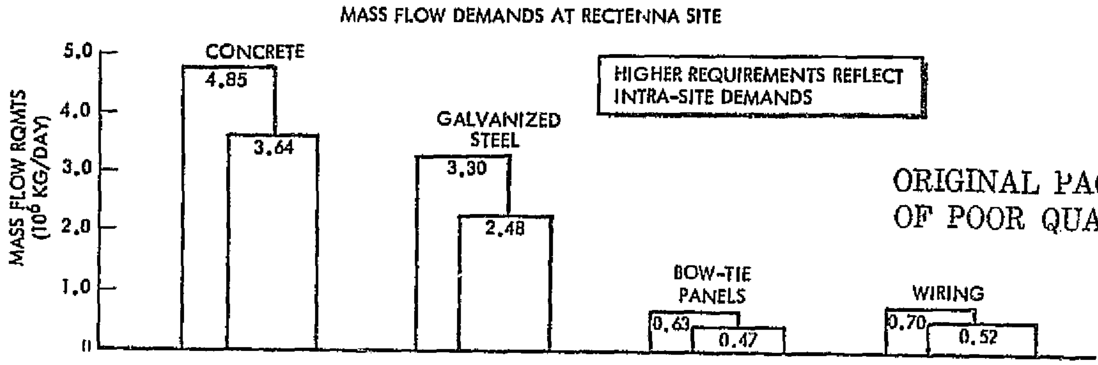

Figure 53. Rectenna Site Logistics

It is of interest to note that the material flow at the rectenna site is much greater than the flow at the launch site. As mentioned previously, 5500 railroad cars per year are required to bring materials (peak requirement); whereas, the 390 truck trips/day at the rectenna site are equivalent to a rate of 36,500 railroad cars per year.

PROGRAMMATICS

Programmatic analyses included (1) an identification of key issues and development of a plan for their resolution, (2) development of an overall plan for technology verification and identification of the major verification elements, (3) preparation of a broad plan for the development phase, and (4) assessment of SPS economics. Figure 54 presents an interlocking overview of these planning areas, illustrating the major phases and relationships to some of the most important supporting hardware elements. The first phase, from 1978 through 1980, is primarily concerned with analytical and low-cost ground experimental analyses aimed at the total or partial resolution of major issues. The second phase, from 1981 through 1987, is the technology verification phase. At the completion of this phase, technology readiness will have been demonstrated. From 1988 through about 1996, a large-scale (about 1-GW) satellite and related ground hardware are constructed and operated to demonstrate end-to-end SPS capability. This satellite and ground hardware are then upgraded to a full 5-GW SPS, and operation of the first SPS occurs by the end of 1998. This overall scenario was used as the basis for SPS planning described in the following sections.

KEY ISSUES

A review of the DOE and NASA in-house and contracted studies was conducted to extract areas of concern. These previous and on-going studies had identified

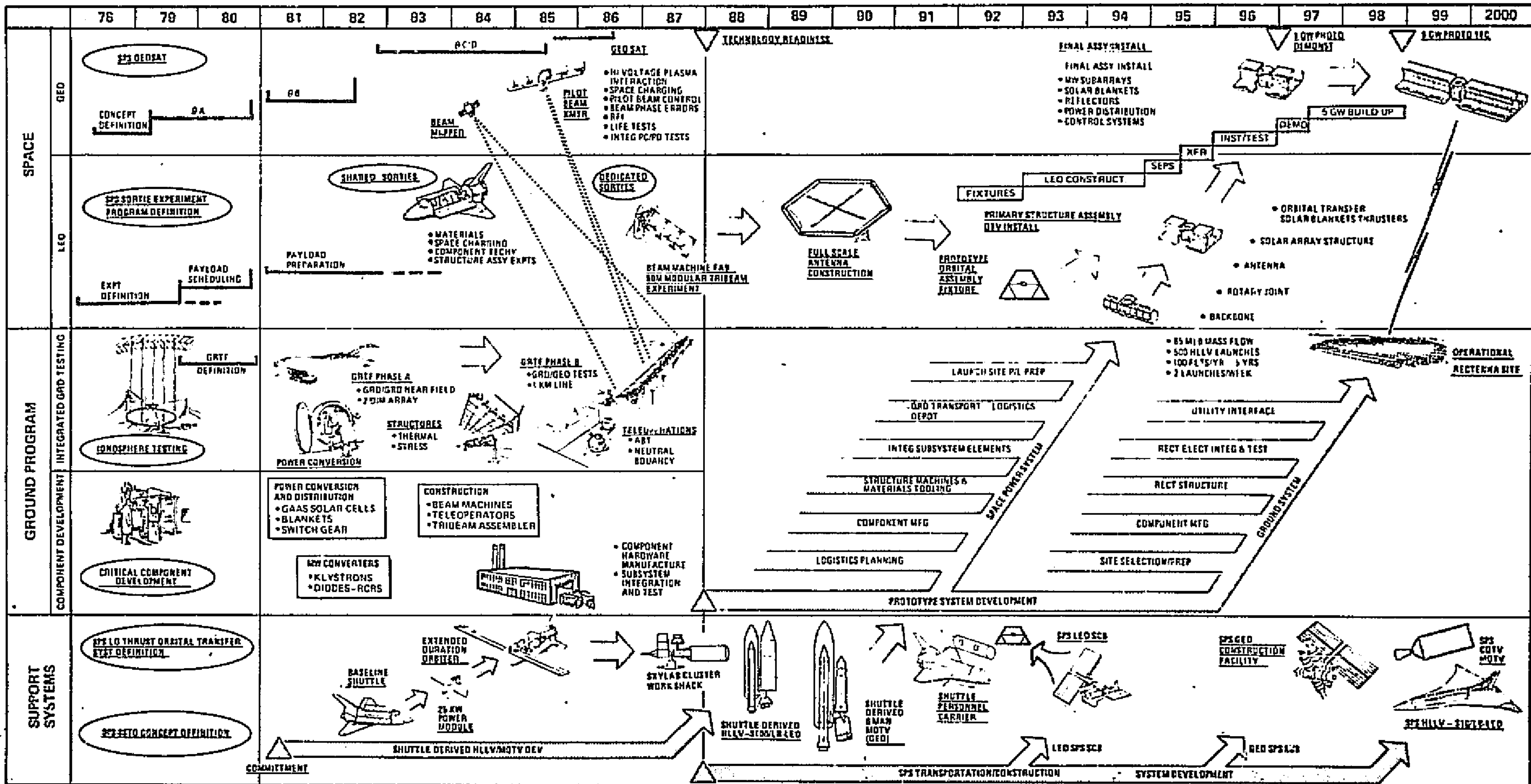

Figure 54. Program Overview

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several issues covering the spectrum of SPS activities. These issues were consolidated into a composite list which was used as a data base from which to derive the most critical issues.

A set of criteria was developed as a guide in evaluating the issues. These criteria consisted of categorizing the issues into one of the following three levels of criticality:

- Level 1 - Potential "show stoppers"
- Level 2 - Potential of serious impact
- Level 3 - Potential of undesirable impact

A Level 1 issue was defined as an issue which, if a negative result were determined or if there were a failure to resolve the issue, could result in the SPS program being labeled as unfeasible. If these issues were not resolved, or a work-around developed, they would be labeled as "show stoppers" and as a result the SPS program would more than likely be discontinued. These issues are called *key issues*. For example, if the capital needed to finance materials, equipment, labor, etc., could not be obtained, the SPS program would not get to the operational phase.

A Level 2 issue was defined as an issue which, if a negative result were determined or if there were a failure to resolve the issue, could result in serious impact to the SPS program. For example, if the solar cell cost was significantly higher than current projections, there might be serious impacts to the SPS program since a significant portion of the satellite cost is attributed to the cost of solar cells.

A Level 3 issue was defined as an issue which, if unresolved, would result in undesirable impacts to the SPS program. For example, crew safety is considered a necessity but if the current plans for crew safety could not be achieved, then surely work-arounds could be developed to provide the safety requirements without significantly impacting the program.

Table 15 presents the issues subdivided, based on the above Level considerations and into areas of economic viability, technical feasibility, and environmental acceptability. In the area of economic viability, three top-level considerations were identified—the ability to achieve cost targets, the cost of other energy sources, and the limitations of key resources (material or human). In the area of technical feasibility, the major concerns are the availability of the required technology in the SPS time period and the ability of the system, subsystem, or elements to meet performance goals. Finally, the impact of the SPS on the environment and the ability to meet existing or potential environmental standards was considered to be an area where issues may exist.

Specific information required for resolution of the key issues (Level 1) was developed and a planned overall approach for resolution was identified. Summary results of these analyses are presented in Table 16. As shown, 60% of the technical issues can be resolved with analysis only; 10% require only ground testing for resolution; and the remaining 30% require space experiments or demonstrations for resolution. Table 16 also shows that 85% resolution of the issues can be accomplished with the expenditure of about $4 billion and prior to development of a prototype.

Table 15. Program Issues and Concerns

CRITICALITY	ECONOMIC VIABILITY	TECHNICAL FEASIBILITY	ENVIRONMENTAL ACCEPTABILITY
LEVEL 1— POTENTIAL SHOW STOPPERS (KEY ISSUES)	CAPITAL INVESTMENTS TRANSPORTATION COST TO ORBIT FRONT-END DDT&E RESOURCE AVAILABILITY COMPETITIVE COST OF ENERGY	PHASE CONTROL LAUNCH RATES ANTENNA POINTING & CONTROL ORBITAL ASSEMBLY	MICROWAVE EXPOSURE STANDARDS MW IMPACT ON OZONE LEVELS & UV RADIATION LAUNCH VEH IMPACT ON OZONE LAYER SPACE RADIATION LIMITS TO CREW
LEVEL 2— POTENTIAL SERIOUS IMPACT	LIGHTWEIGHT BLANKET PRODUCIBILITY OTV PERFORMANCE CHARACTERISTICS OPERATIONS/MAINTENANCE COST MW ELEMENT LIFE FAILURE RATES PWR CONVERSION DEVICES LIFE/DEGRADATION ATT CONT & STATIONKEEPING THRUSTER PERFORMANCE AND LIFE RECTENNA LAND REQUIREMENTS SYSTEMS COMPLEXITY PAYLOAD PACKAGING DENSITY REFLECTOR FILM DEVELOPMENT ON-BOARD ENERGY STORAGE	DC/RF CONVERTERS WAVEGUIDES SATELLITE POINTING & CONTROL SHUTDOWN/STARTUP OF MW ANTENNA LAUNCH VEHICLE SIZE SPACE MAINTENANCE PROCESSES ORBITAL TRANSFER OF LARGE SPACE STRUCTURES POWER DISTRIBUTION SWITCHING TECHNOLOGY/CAPACITY HIGH-TEMPERATURE HEAT EXCHANGERS PROPELLANT RESUPPLY IN ORBIT RELIABLE FLUID CONTAINMENT REFLECTOR FILMS FLATNESS CONTROL VOLTAGE & CURRENT REGULATION MICROWAVE BEAM DISPERSION ANALYSIS	PUBLIC ACCEPTANCE OF SPS RFI EMI HIGH-VOLTAGE SPACE CHARGING PLASMA INTERACTIONS LAUNCH VEHICLE NOISE & SONIC BOOMS GEO ORBIT AVAILABILITY SPACE COLLISIONS ENERGY BALANCES OTV EMISSIONS
LEVEL 3— POTENTIAL UNDESIRABLE IMPACT	ASSIGNMENT OF MW FREQUENCY LAUNCH VEHICLE RECOVERY/REFURBISHMENT ON-GRND PWR FLUCTUATIONS & STORAGE REFLECTOR FILMS DEGRADATION RECTENNA ELEMENT LIFE/FAILURE RATES/ MAINTENANCE OIL USAGE (LAUNCH VEHICLES) LAUNCH SITE(S) LAND REQUIREMENTS INTERNATIONAL EMBARGOES/CRISES GSE REQUIREMENTS SECURITY REUSABLE PARTS	CONSTRUCTION BASE LOGISTICS POWER CONDUCTION TERRESTRIAL LOGISTICS FUTURE NASA PROGRAMS RECTENNA OPERATIONS DEV PROGRAM SCHEDULE & FLEXIBILITY RECTENNA/UTILITY INTERFACES ENGINEERING AVAILABILITY RECTENNA INFORMATION MGMT SYSTEMS SATELLITE INFORMATION MGMT PROCESSING MANUFACTURING CAPABILITIES/DEMANDS	SAFETY & CONTROL OF LAUNCH VEHICLES ORBITAL CREW SAFETY POLLUTANTS FROM MINING & MANUFACTURING TERRESTRIAL WORKERS HEALTH & SAFETY MW EFFECT ON ECOLOGY, SOIL, WATER, AND ATMOSPHERE POLLUTANTS FROM TRANSPORTATION OPNS FAILED HARDWARE (ON SPS) DISPENSATION LAND USE NEAR RECTENNA

Table 16. Resolution of Technical Issues

	GROUND		SPACE		%
	SYST DEF OPNS ANAL PROGRAMMATIC	SUBSYSTS COMPONENT TECHNOLOGY	SORTIE	PROTO	
ANALYSIS	40	20			60
GROUND		10			10
SPACE			15	15	30
% TECHY RESOLUTION	40	30	15	15	≈ 85% RESOLUTION WITH COMPREHENSIVE GROUND DEV/SORTIE PROGRAM
% FUNDING	T / $.02B	8% / 2.5B	5% / 1.5B	87% / 26.0B	

VERIFICATION PLAN

Two activities lead to SPS technology readiness by 1987. From 1978 to about 1980, small-scale experimental research and analytical activities are accomplished. From 1981 to 1987, verification of SPS technology occurs. In developing the technology verification plan, the basic constraint was to devise a low-cost program with a reasonable chance for success and adequate lead times. This led to an approach that made maximum use of ground testing and maximizes use of the Shuttle with minimum modification.

Major areas requiring verification include:

- Interaction of the microwave beam with the ionosphere, causing radio frequency interference.
- Proof of microwave transmission system concept.
- Proof of satellite assembly concepts.
- Verification of component performance.
- Verification of key system problem areas such as high-voltage space plasma interactions and space charging.

Figure 55 illustrates these areas and their approximate schedule and element relationships. The critical components shown at the bottom of the figure are potential areas of concern for early development to assure that the hardware is available in time for ground and space testing. Initial ground testing of the components and subsystems will be accomplished in thermal-vacuum facilities. Early space testing will be accomplished in low earth orbit, using shared Shuttle sortie flights. Testing of the components and subsystems at geosynchronous orbit will be accomplished using a dedicated satellite system (GEOSAT), which will be described later.

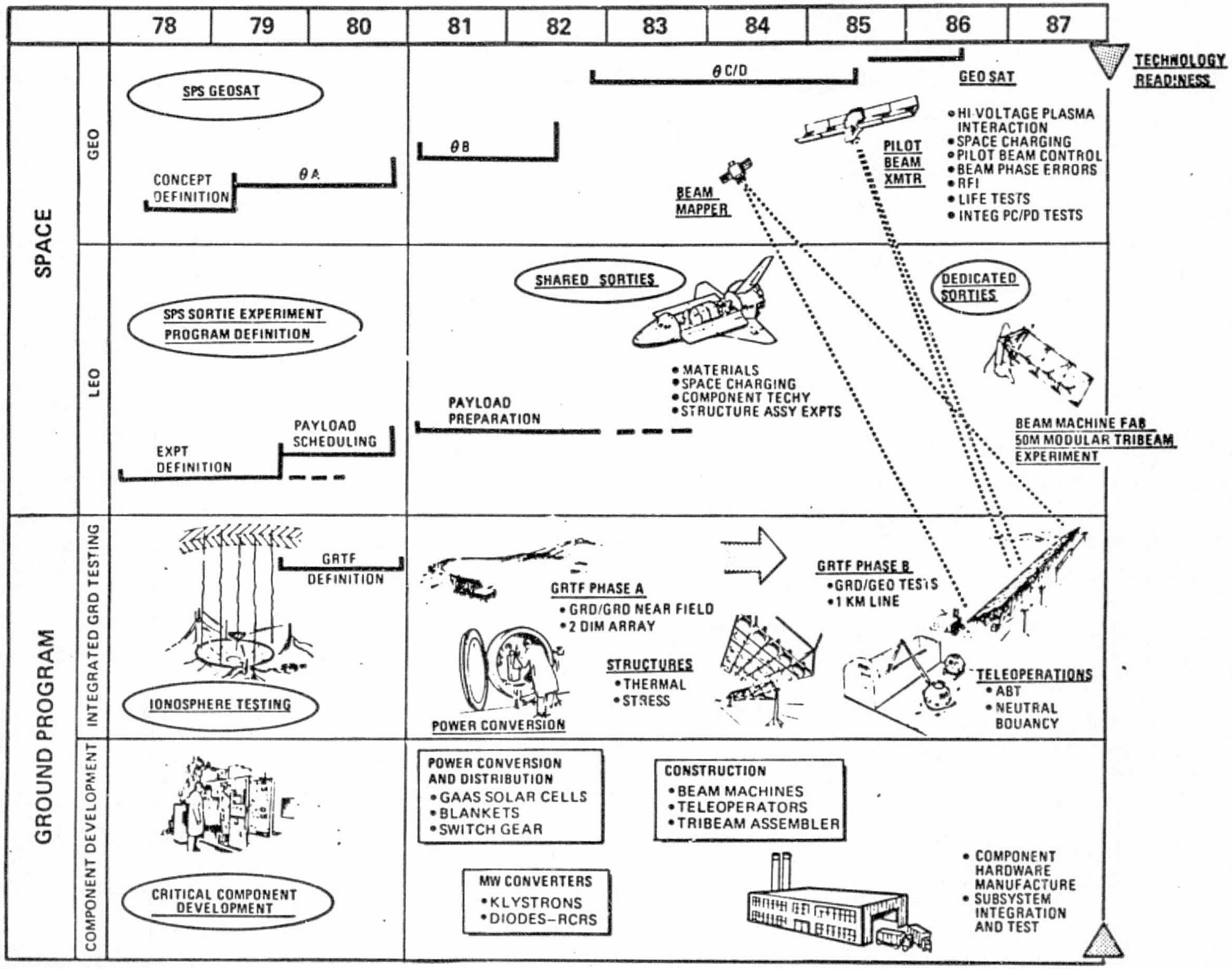

Figure 55. SPS Technology Advancement Plan Elements

One of the major potential problems—interaction of the microwave beam with the ionosphere—requires early experimentation using existing or modified existing facilities to the maximum to achieve local intensities of at least 23 mW/cm^2. Microwave transmitting facilities such as those located at Aricibo and Platteville (when modified) may be sufficient to provide the required beam intensities at the proper frequencies. Existing radar facilities would be used to assess the extent of radio frequency interference. These data are essential to provide design requirements data for the microwave antenna.

Proof of concept for the microwave transmission system can be accomplished using ground hardware to a maximum extent. A ground radiation test facility (GRTF) will be developed to be used in conjunction with a two-satellite system located in geosynchronous orbit (GEOSAT) to provide system test data. Initial GRTF testing will utilize a relatively small (0.1 km long) antenna array for ground-to-ground near-field microwave testing. This linear array will later be increased to 1.0 km in length for proof-of-concept testing in conjunction with the GEOSAT system. This is essentially an "upside down" test; i.e., hardware normally in space (the microwave antenna) is located on the ground, and hardware normally located on the ground is at geosynchronous orbit (the pilot beam transmitter). This approach results in a low geosynchronous orbit mass and allows the complex portion of the system that requires potential adjustments by man to be located on the ground.

The two-satellite GEOSAT system concept, shown in Figure 56, may be used to test several SPS subsystem elements. The small subsatellite is used to map

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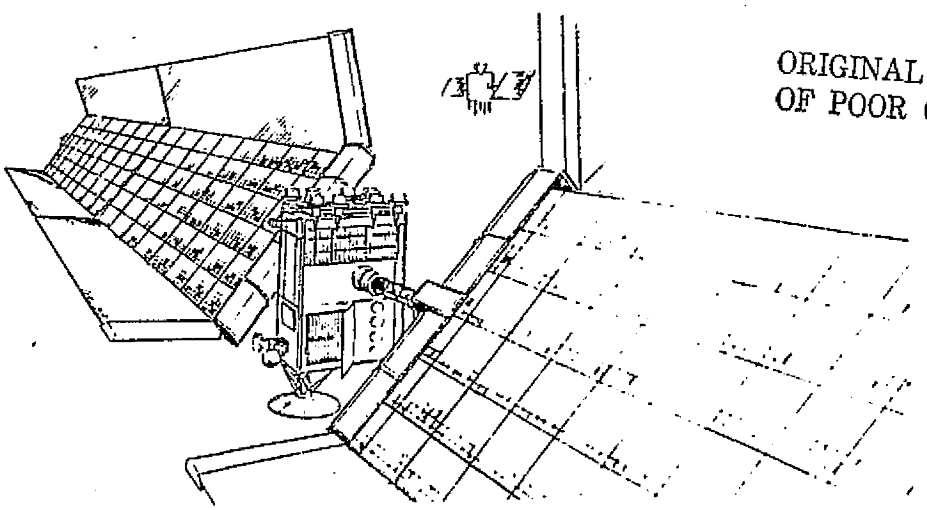

Figure 56. SPS GEOSAT System Elements

the microwave beam generated by the GRTF to assess beam quality. The large satellite contains the pilot beam which provides the retrodirective signal to the 1-km linear array on the ground. Also contained in the large satellite are the following:

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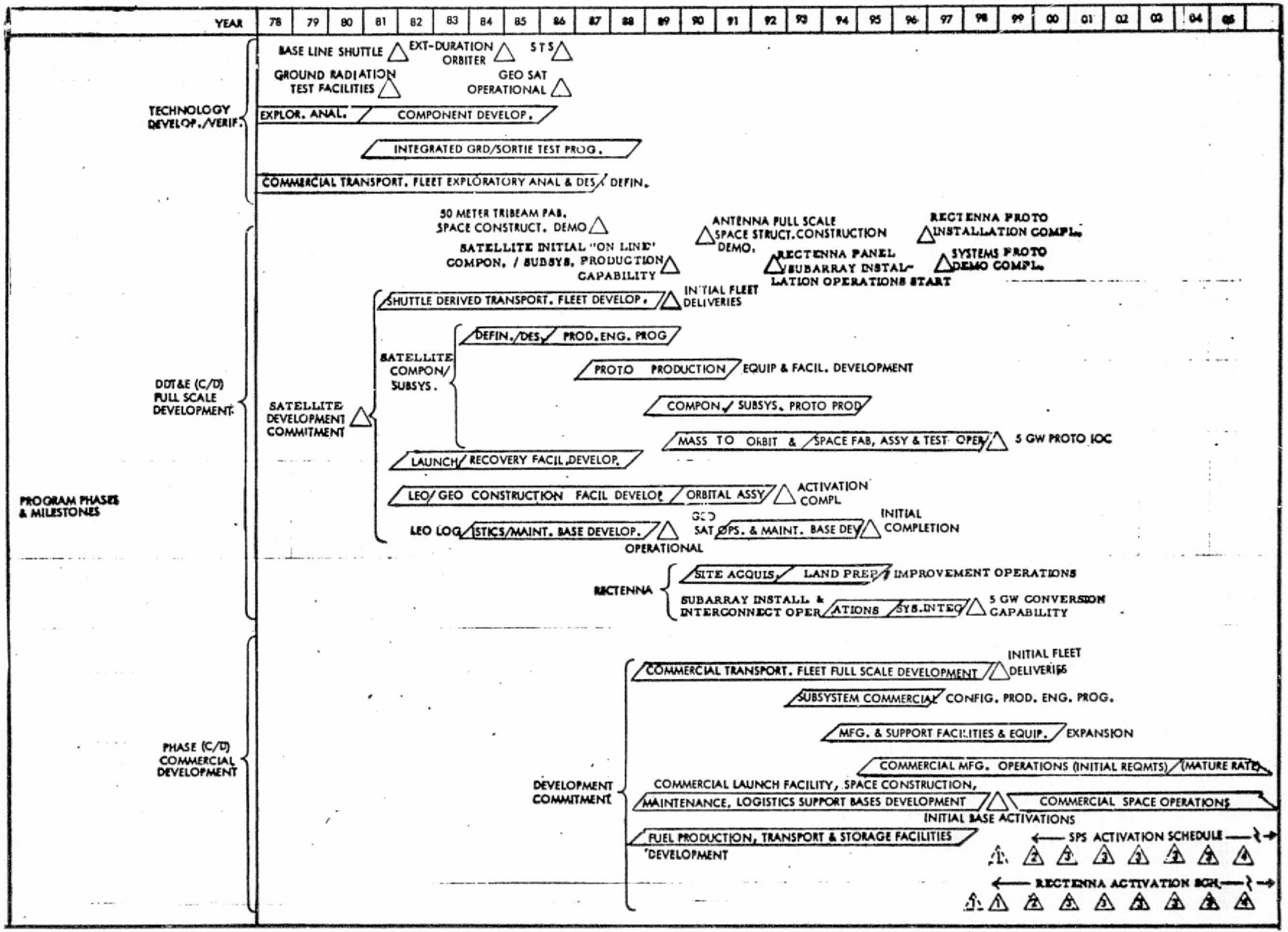

Figure 58. SPS Summary Program Phases and Milestones

Rockwell International
Space Division

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ECONOMICS

Economic analyses included development of a data base for determining cost parameters, cost estimating, cost probability determination, risk analysis, and economic feasibility.

Cost estimates for DDT&E, theoretical first-unit (TFU), investment per satellite system (average satellite system cost), and replacement capital investment/operations and maintenance are presented in Figure 59.

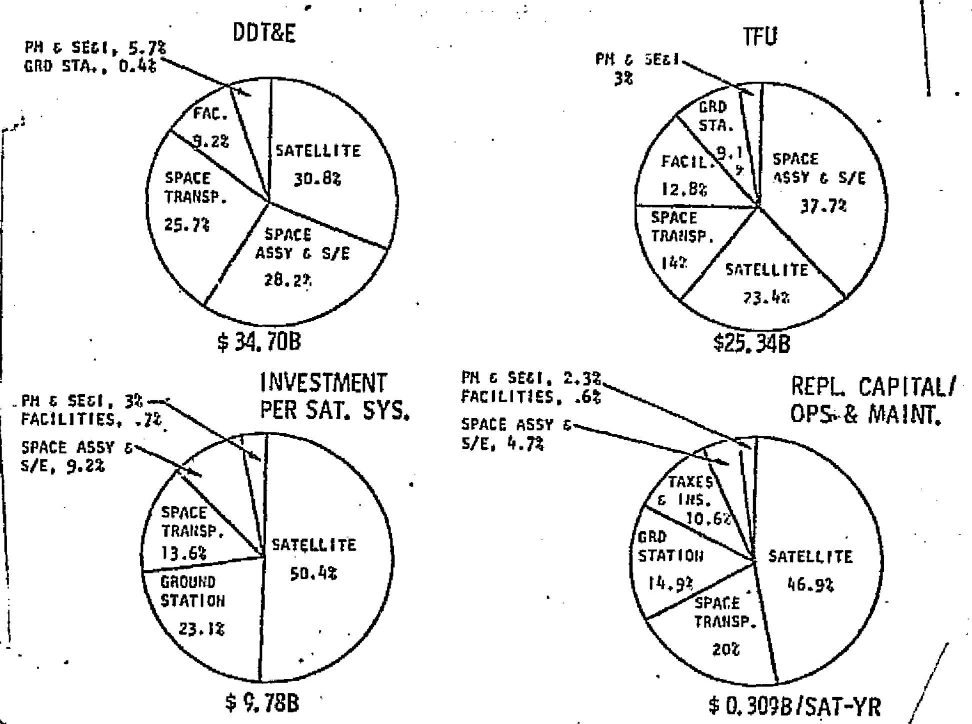


Figure 59. SPS Cost Breakdown

The DDT&E phase consists of the one-time effort associated with designing, development, and evaluating the components, subsystems, and systems required for the SPS project. All DDT&E effort associated with SPS-related support systems such as space transportation, facilities, and space assembly and support equipment necessary to accomplish the satellite DDT&E phase is also included. Total project development cost through the first full 5-GW satellite (TFU) is $60.0 billion (DDT&E is $34.7 billion and TFU is $25.3 billion). The satellite portion of the DDT&E cost is $10.7 billion. Cost to develop the space assembly and support equipment is $9.8 billion, and space transportation DDT&E is $8.9 billion.

The TFU cost breakdown reflects a somewhat different makeup of costs when compared to DDT&E costs. TFU estimates include the full dollar assessment for an initial satellite buildup including space transportation fleets (HLLV and OTV's), initial space assembly and support equipment requirements, and the facilities needed to establish the SPS operational capability. This means that the TFU cost includes elements with a lifetime capability of building more than one SPS system. In this regard, analyses will show that space assembly and support equipment represents the largest portion of total TFU costs—$9.6 billion. Satellite system costs are $5.9 billion; space transportation, $3.6 billion; and ground facilities, $3.3 billion. In the space assembly and support equipment TFU cost, the LEO/GEO satellite construction base makes up 53 percent of the total space assembly and support equipment estimate.

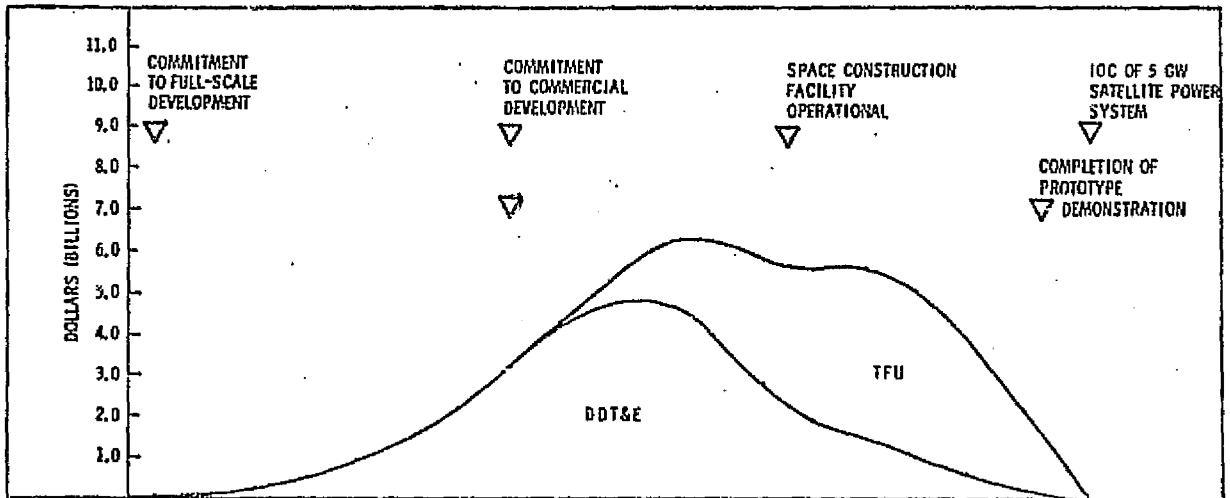
Investment per satellite is equivalent to the average unit cost of the total SPS requirement (TFU plus Satellites 2 through 120). This total average cost of $9.8 billion includes $4.9 billion for the satellite, $2.3 billion for the ground station (rectenna), $1.3 billion for space transportation, and $0.9 billion for space assembly and support equipment. The total average investment cost per 5-GW satellite yields an investment cost of $1956/kW.

SPS replacement capital and operations/maintenance phases have been combined and are estimated at an annual cost of $0.31 billion per satellite year or $9.3 billion over the 30-year operational period. Satellite requirements comprise $0.15 billion of the cost, or 46.9 percent.

Figure 60 graphically displays the funding requirements and peak year distributions for DDT&E and TFU. DDT&E costs peak at just over $4.7 billion during the years 10 and 11 (1989 and 1990). This time period corresponds to the activation time of the space construction base, orbital support equipment, and satellite construction fixtures. The TFU costs peak at around $4.1 billion in years 15 and 16 (1994 and 1995), which is the time period associated with construction, assembly, and test of the first full 5-GW satellite.

The economics of SPS are an essential element in the evaluation of a concept for the generation, transmission, and distribution of electrical power to the consumer. Studies were conducted to identify financial arrangements and practical approaches for the funding and operation of a satellite power system organization, and conclusions resulted in the evolution of a single concept for the funding and operation of the SPS. Organizationally, a national SPS (government-owned) authority would be created to undertake the space segment and a combination of utility consortia would undertake the ground segment. The scope of activities/responsibilities of the space segment would include the satellite, space assembly and support facility, launch, and space transportation systems. Ground segment responsibilities would include rectenna sites, utility interface, and transmission/distribution. Space segment funding would be furnished by the government, and ground segment financing would be furnished by the utility consortia. The consortia would then purchase electrical power delivered to the rectenna and reimburse the government for space segment investments. This approach is illustrated in Figure 61.

In concert with this financial and operational concept of managing the SPS, cash flow analyses and microeconomic determinations were conducted using the 5-GW GaAlAs SPS point design, by considering specific evaluation criteria for profitability, investment/costs funding, and revenues. Financial and operational models on such items as income statements, assets-employed



YEAR	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	TOTAL DOLLARS (Billions)
	1980	1981	1982	1983	1984	1985	1986	1987	1988	1969	1990	1991	1992	1993	1994	1995	1996	1997	1998	1999	
DDT&E	.013	.082	.221	.502	.936	1.497	2.220	3.220	4.170	4.758	4.764	4.033	2.815	1.915	1.496	1.054	.673	.289	.047	0	34.705
TFU							.	.001	.087	.585	1.382	2.256	3.041	3.718	4.162	4.060	3.275	2.066	.708	0	25.341
TOTAL	.013	.082	.221	.502	.936	1.497	2.220	3.221	4.257	5.343	6.145	6.289	5.856	5.633	5.658	5.114	3.948	2.355	.755	0	60.046

Figure 60. Annual DDT&E and TFU Costs

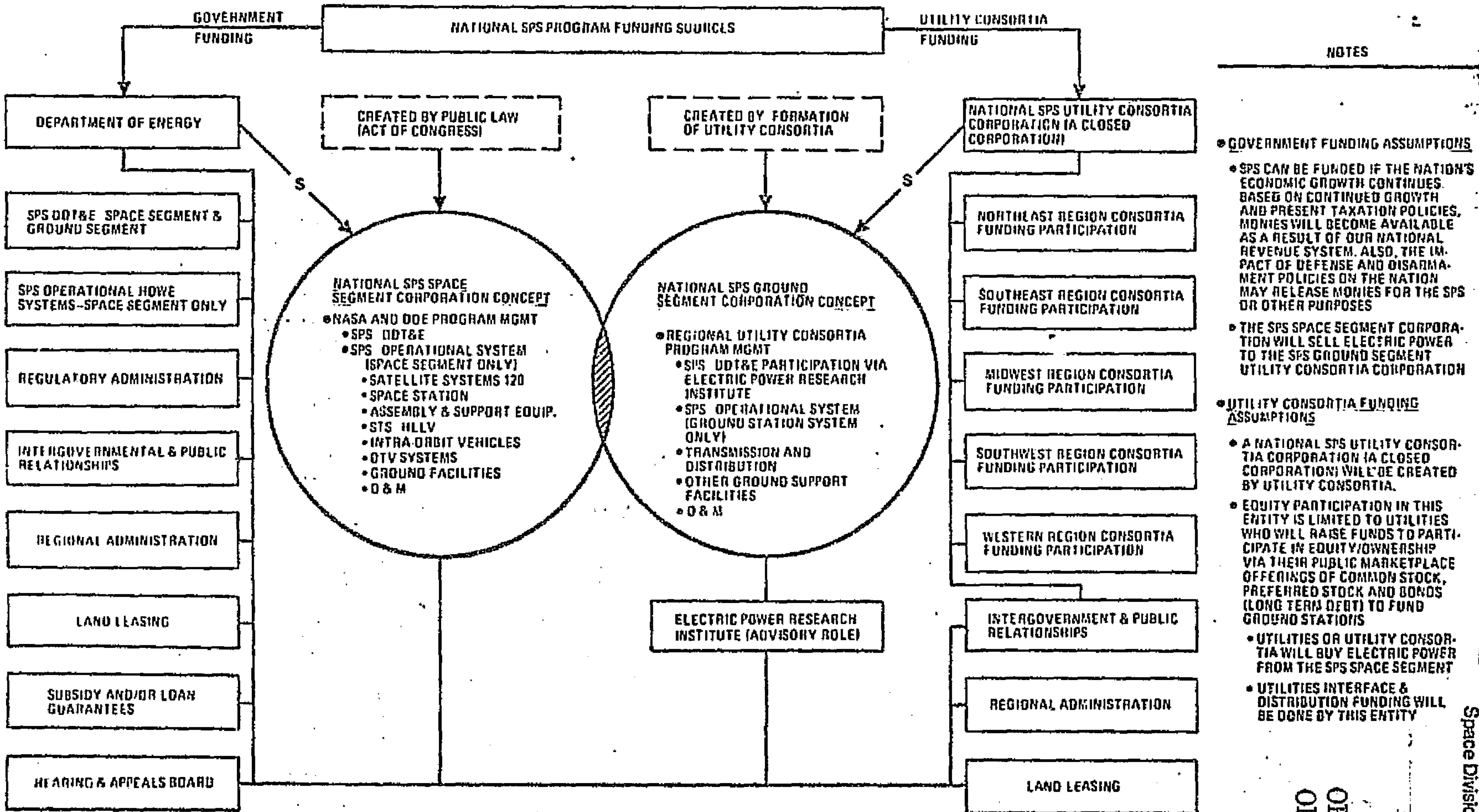


Figure 61. Funding Concept for a National Satellite Power System Program

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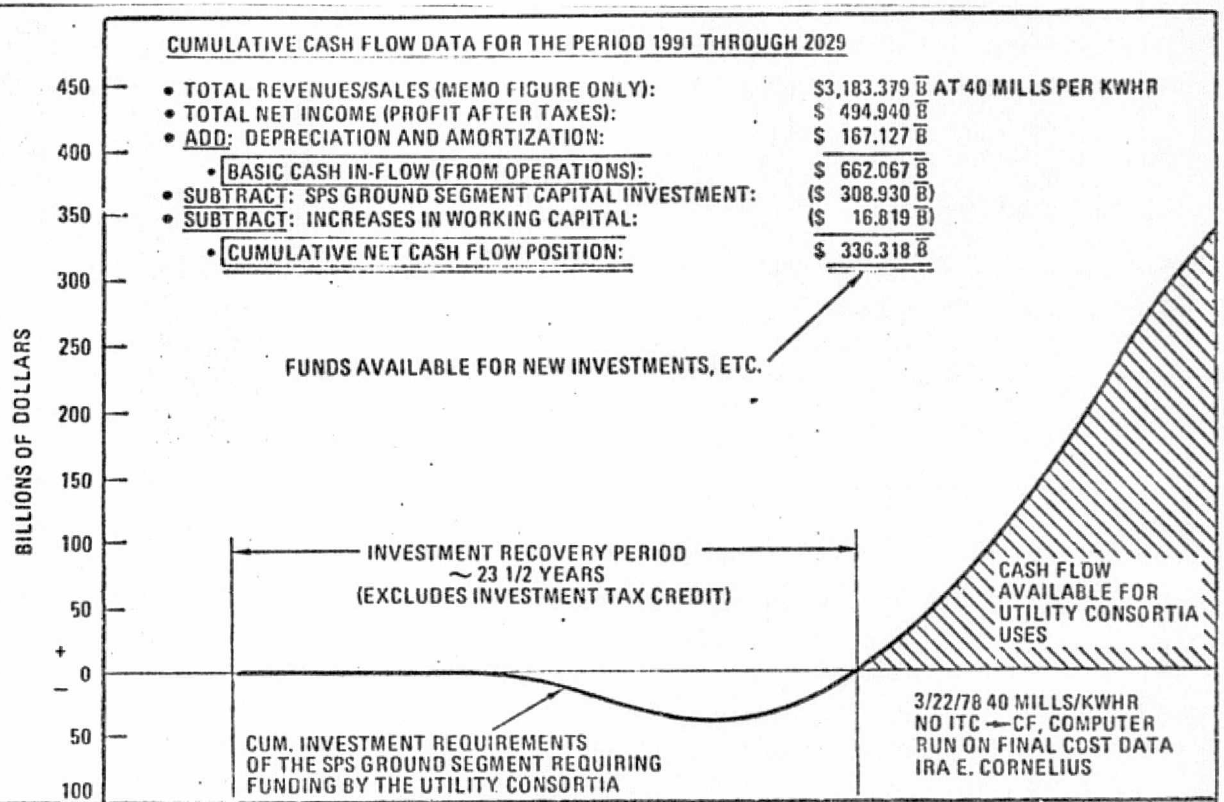
statements, and cash-flow statements were developed for an evaluation of a National SPS Space Segment Authority (federally owned) and the SPS Ground Segment Utility Corporation. Cash-flow performances and investment recovery, based on 40 mills per kilowatt hour without an investment tax credit, would result in an investment recovery for the ground segment by the year 2015, as shown in Figure 62. Although the investment recovery period shown in this figure is 23-1/2 years, the recovery period for the time where significant investments occur is about 12 years. These investment characteristics are considered to be financially feasible.



NO. OF OPERATIONAL RECTENNAS	–	–	–	–	–	1	6	15	26	38	50	62	74	86	98	110	120
CUMULATIVE NET CASH FLOW ($B)	–0–	–0–	(0.083)	(0.570)	(1.447)	(4.007)	(16.088)	(29.958)	(37.383)	(33.498)	(15.763)	15.634	60.579	118.997	190.836	276.056	336.318
SELECTED YEARS	1988	1989	1991	1993	1995	1998	2000	2003	2006	2009	2012	2015	2018	2021	2024	2027	2029

Figure 62. SPS Ground Segment Utility Consortia Corporation Concept—Cumulative Cash Flow and Investment Recovery Schedule at 40 Mills/kWh (excludes investment tax credits)

CONCLUSIONS

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SATELLITE CONCEPT

- ✓ The GaAlAs photovoltaic concept is recommended as the current preliminary baseline satellite concept.
- ✓ Silicon photovoltaic and Rankine cycle solar-thermal concepts are viable alternatives.

SATELLITE CONSTRUCTION LOCATION

- ✓ Geosynchronous orbit is preferred for construction of the satellite.

TRANSPORTATION SYSTEM CONCEPT

- ✓ A horizontal takeoff and landing (one- or two-stage) airbreather/rocket HLLV concept is preferred for earth-to-LEO transportation.
- ✓ Vertical takeoff options are viable alternatives.
- ✓ An argon electric orbit transfer vehicle is preferred for cargo transport from low earth orbit to geosynchronous orbit.
- ✓ A chemical, LH_2/LO_2, two-stage orbit transfer vehicle is preferred for crew transport from low earth orbit to geosynchronous orbit.

RECTENNA CONCEPT

- ✓ A stripline rectenna array is the current preferred concept.

PROGRAMMATICS

- ✓ The SPS concept appears technically feasible and economically competitive.
- ✓ A verification program with primary emphasis on ground test articles is recommended.

RECOMMENDATIONS

RECOMMENDATIONS

RECOMMENDATIONS

The SPS system studies should be continued to provide support for near-term SPS program milestones; i.e., NASA-MSFC recommendations to DOE/NASA of baseline SPS concept during August 1978, and MSFC program plan recommendations to DOE/NASA during February 1979.

Specific areas that need to be addressed include the following:

- ✓ Update of preliminary baseline SPS concept recommended to DOE/NASA on January 24, 1978.
- ✓ Feasibility of solid-state electronics antenna approach and its impact on SPS concept.
- ✓ Definition of satellite, rectenna, and support systems constructability.
- ✓ Identification of supporting space and ground systems concepts.
- ✓ Continued definition of transportation system elements.
- ✓ Continued overall assessment of SPS operations with development of in-depth data in areas of significant concept, program, or cost impact.
- ✓ Continued evolution of experiment/verification programs, including experiment definitions and verification elements concepts.
- ✓ Continued program analysis with emphasis on the development of cost data in key areas such as solar cell manufacturing, rectenna elements, microwave devices, and transportation.

In addition to the above SPS system studies, it is also recommended that a significant effort be initiated in areas requiring advanced technology and experimental effort. Some of the key areas include preliminary microwave transmission system concepts verification, microwave transmission environmental assessment, and GaAlAs solar cell development (including replacement materials for GaAs substrates, such as synthetic sapphire).

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