SOLAR POWER SATELLITE
BUILT OF LUNAR MATERIALS

Study conducted by:

SPACE RESEARCH ASSOCIATES, INC.

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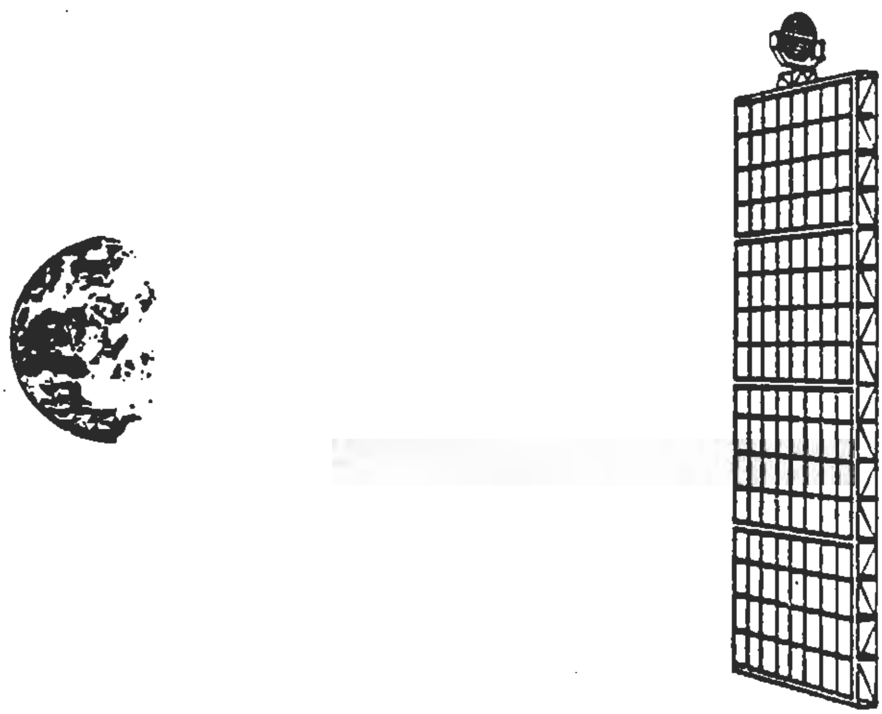

FINAL REPORT

Sept. 21, 1985

Study conducted for:

SPACE STUDIES INSTITUTE

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FOREWORD

This final report was prepared by Space Research Associates, Inc. (SRA) for the Space Studies Institute. It presents the results of work done between June 1984 and July 1985. Interim reports were presented to SSI in September 1984, January 1985, and June 1985.

Significant contributions to this study were made by SRA personnel, five advisers, a consultant on lunar geology, and several volunteers from Seattle chapters of the L5 Society.

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SRA thanks the members of the advisory board for their time and their many helpful suggestions. The advisory board's experience and judgement provided valuable guidance for the study team. A preliminary summary of the study was presented at the 1985 Princeton Conference on Space Manufacturing, which resulted in constructive input from many of the attendees. Dr. Gillett's advice on the availability of relatively rare lunar materials was of great help. The work of the volunteers, especially Art LaPella, significantly improved the clarity and consistency of the reports. Thanks are also given to Joel Sercel of JPL for several helpful conversations and references.

1. EXECUTIVE SUMMARY

High cost has been a major obstacle to development of a Solar Power Satellite (SPS) system. A major part of this cost is the expense of transporting material from Earth to geosynchronous orbit (GEO).

The energy required to transport lunar material to GEO is less than 8% of that for Earth material. In addition, launch from the Moon should be more efficient than launch from Earth due to low lunar gravity and the effective lack of a lunar atmosphere. Thus, the cost of delivering materials from the Moon to GEO might be about one-fiftieth of the cost to deliver equivalent materials from Earth.(4) This suggests that the use of lunar materials could significantly reduce the cost of building an SPS system.

General Dynamics Convair has studied the use of lunar resources for SPS construction.(1) The rules of that study allowed only minor changes to an earlier Boeing design(2) which had assumed that all materials came from Earth. Thus, the resulting design was not optimized for the use of lunar materials. Even so, it required only 10% as much Earth mass as the Boeing design.(1)

This document is the final report of Space Research Associates' SPS design study emphasizing minimal use of Earth-supplied mass to achieve low cost. The ground rules of the study are listed in Table 1-1. The study objective was to provide a basis for comparison of alternative SPS concepts and subsystems in the context of lunar material utilization.

TABLE 1-1
STUDY GROUND RULES

- Any reasonably abundant chemical element present in lunar material could be mined, refined, and transported to the SPS construction site. All other materials must be transported from Earth.

- The primary design requirement is minimal non-lunar mass. Low total mass is only a minor concern.

- Scope of the study does not include design of transportation systems, materials processing, or assembly systems.

- To enhance commercial feasibility, undemonstrated technology is to be avoided in the design wherever possible.

- Only microwave transmission of power is to be considered. The peak intensity at the ground is not to exceed 300 W/m^2.

Elements abundant on the Moon are aluminum, calcium, iron, magnesium, oxygen, silicon, and titanium. Less abundant (< 1%) but potentially available are chromium, potassium, manganese, and sodium; though using them does not seem unreasonable, there is assumed to be some technological or economic risk in planning to use these materials. A special case is nickel, which is a component of some micrometeorites. Though present in only tiny amounts in lunar soil, these micrometeorites are high quality steel and should be easy to separate magnetically from the soil.(7) Thus, using nickel steel is assumed to be a reasonably good risk.

1.1 SUMMARY OF RESULTS

This section presents the broad, satellite-level results of the study, followed by technical results for each major satellite system.

It is found that an SPS can be designed which uses less than one percent as much non-lunar material as the Earth baseline SPS, a factor of ten improvement over the results of the General Dynamics study. The total mass of the system is about eight percent greater than that of one made from Earth-derived materials. The best design uses silicon photovoltaic cells for power conversion. Its structure is primarily aluminum and uses aluminum oxide coatings for thermal control. Radiators are more than 99 percent composed of lunar material. A flywheel system is used for energy storage during eclipses. The design is suitable for largely automated construction.

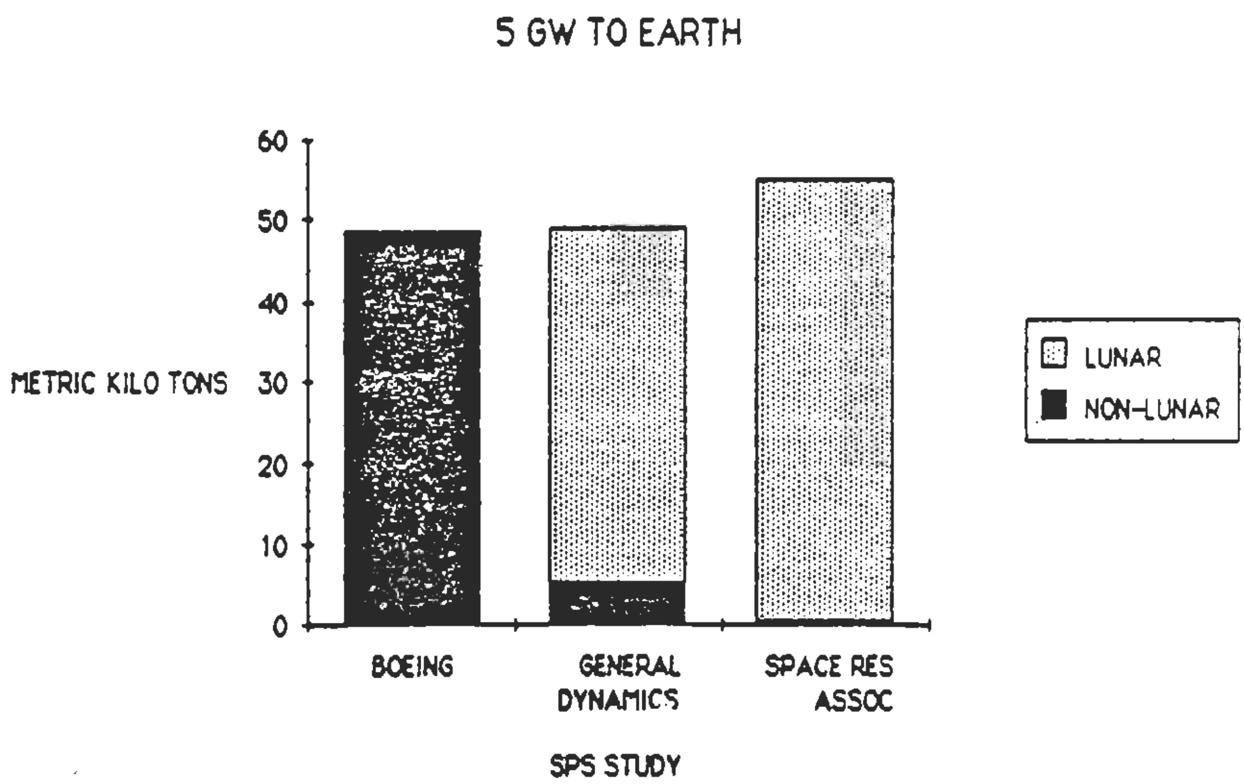

FIGURE 1.1-1, TOTAL MASS TO NON-LUNAR MASS COMPARISON

Figure 1.1-1 compares total mass and non-lunar mass of the Boeing SPS design, the General Dynamics SPS design, and the Space Research Associates SPS design. Figure 1.1-2 compares the estimated cost per SPS for each design, based on the assumption that non-lunar material is fifty times more costly than lunar material. The cost units are equivalent to thousands of metric tons of lunar material. Growth and contingency analyses were outside the scope of this study, so the mass estimate for the SRA design assumes the same growth factor, 26.7%, as the Boeing design.

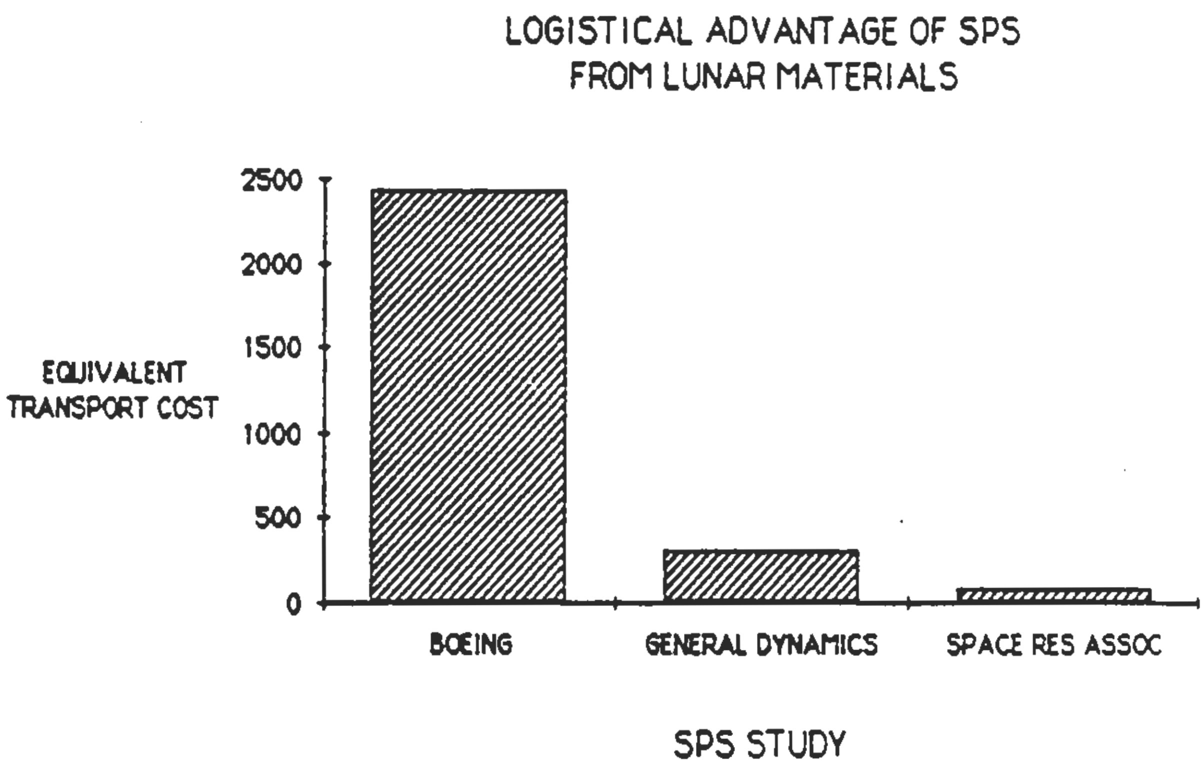

FIGURE 1.1-2, ESTIMATED SPS COST COMPARISON

An alternative to silicon cells is to use gallium arsenide cells with solar concentrators. This roughly triples total SPS mass, but non-lunar mass remains less than two percent of that for the Earth baseline. Four other power conversion systems were investigated: thermophotovoltaic (TPV), Brayton, Rankine, and Stirling. These were found to use significantly more non-lunar material than either the silicon system or the gallium arsenide system.

It appears that a lens made of aluminum mesh could economically increase the effective aperture of the SPS antenna. If this proves to be correct, the lower limit on SPS power could be as low as 200 MW per satellite. This is an area for further study.

1.1.1 Solar Power Conversion Systems

Solar power conversion systems include subsystems which collect and intensify solar radiation and those which dissipate heat from the power conversion process. Since design of these subsystems is largely independent of the conversion systems they support, they are discussed separately after the power converters.

Systems for converting sunlight into electricity may be categorized as those which require active cooling and those which do not. Two systems which use passive cooling were investigated: silicon planar and gallium arsenide concentrator. Four systems which require active cooling were investigated: thermophotovoltaic (TPV) conversion, Brayton cycle, Rankine cycle, and Stirling cycle. At least one SPS design was developed and evaluated using each system. As stated above, the passively cooled systems were found to require significantly less non-lunar material. Total specific mass and non-lunar specific mass for each power conversion system are compared in Figs. 1.1-3 and 1.1-4. These figures do not include masses of other SPS subsystems.

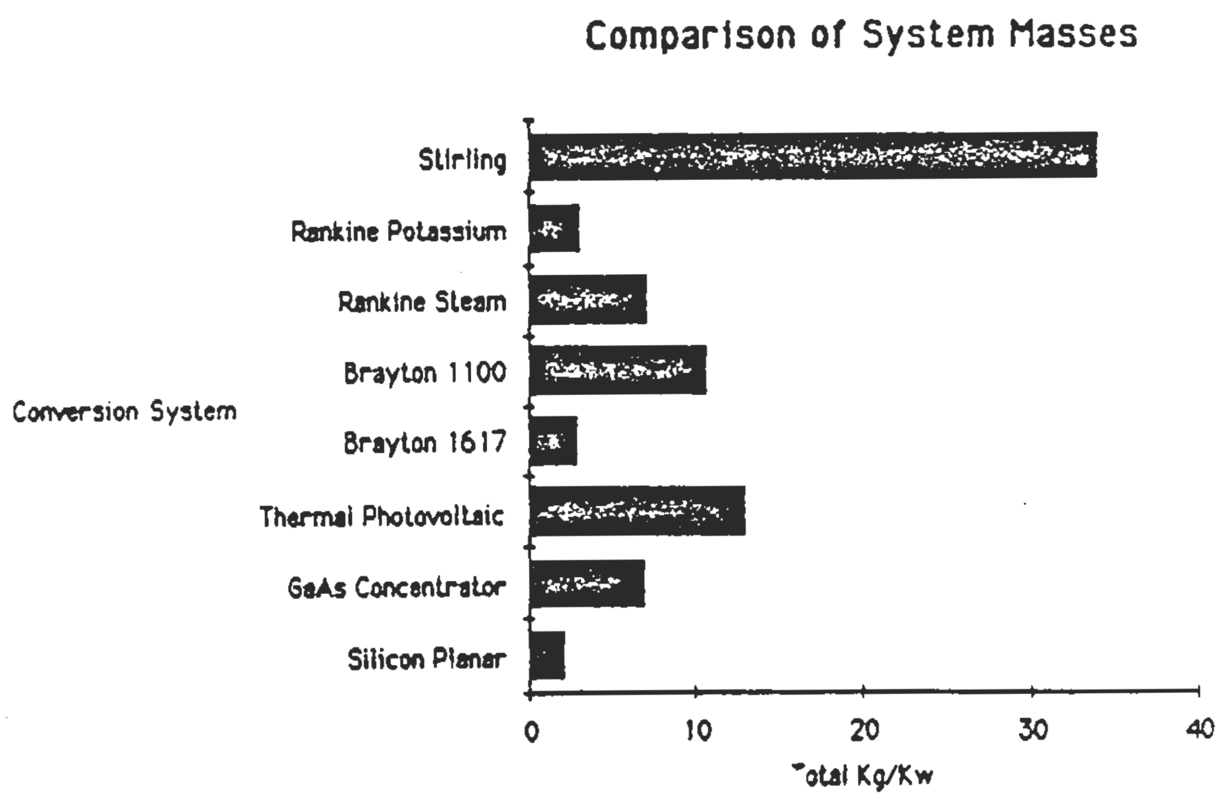

FIGURE 1.1-3, TOTAL SPECIFIC MASS COMPARISON

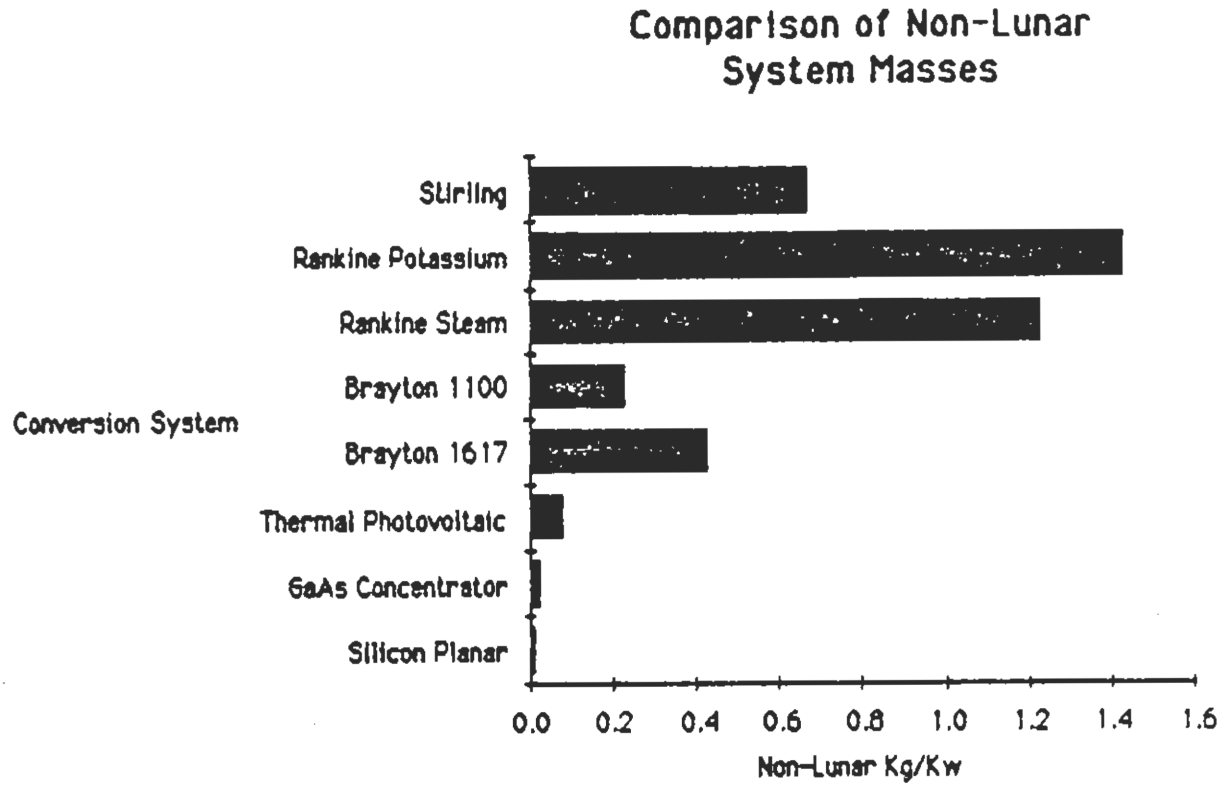

FIGURE 1.1-4, NON-LUNAR SPECIFIC MASS COMPARISON

1.1.1.1 Silicon Planar

In the silicon planar configuration, non-concentrated sunlight falls onto a planar array of silicon cells. Cells are passively cooled by radiating from their own surfaces. They are partly protected from harmful radiation by thick silica glass windows and substrate. Cells are periodically annealed to repair damage from energetic particles.

Previous silicon planar SPS designs have used plastic tape to connect solar cell panels in a flat array. The design developed in this study uses aluminum wire to connect and support the panels, reducing non-lunar mass. The panels are angled to face the sun more squarely, which improves efficiency.

The silicon planar design uses virtually no non-lunar materials. It is also the least massive conversion system overall, as shown in Figures 1.1-3 and 1.1-4. The mass shown may be optimistic by a factor of two if radiation damage cannot be repeatedly annealed out of silicon cells, but the silicon planar concept would still require the least non-lunar material.

1.1.1.2 Gallium Arsenide Concentrator

The gallium arsenide (GaAs) concentrator concept uses GaAs cells to produce electricity from concentrated sunlight. Neither gallium nor arsenic is abundant on the Moon, but GaAs cells can be used with highly concentrated sunlight. Thus, the mass of the cells - and hence the mass of non-lunar material - need be only a small fraction of the SPS mass. GaAs cells are more resistant to radiation damage than silicon, but still require periodic annealing or thick cover glass.

The design used here has a Cassegrain optical system with a concentration ratio of 260. Aluminum primary and secondary mirrors focus sunlight onto a small GaAs cell which is fixed to the center of the primary mirror. Excess heat from the cell is radiated from the back surface of the primary, which constrains the design to use many small units. The cell operates at 200 C, producing 4.04 watts per unit. As shown in Figures 1.1-3 and 1.1-4, this design is second only to silicon planar in minimizing non-lunar materials. Further optimization of the design could probably reduce the GaAs system's total mass by a factor of at least two.

1.1.1.3 Thermophotovoltaic (TPV)

A TPV converter moves the spectral peak of concentrated sunlight from the visible region to the infrared, which is more efficiently converted to electricity by photovoltaic cells. The cells require active cooling.

A parametric model of a TPV converter using silicon photovoltaic cells was developed. The result of optimizing this model for minimal non-lunar mass is shown in Figures 1.1-3 and 1.1-4. The high mass of the TPV system is due to active cooling of the silicon cells. The cooling system also accounts for more than 90% of the non-lunar mass.

An advantage of TPV conversion over other photovoltaic systems is that the cells are enclosed within a substantial structure which protects them from harmful radiation, so no annealing is required. TPV technology is advancing rapidly, so it should be considered in future SPS studies. TPV would be especially appealing if a cooling system with very little non-lunar mass were developed.

1.1.1.4 Brayton Cycle

High temperatures in the Brayton design yield high efficiency and low total mass but make lunar material substitution difficult due to the need for advanced materials. The degree of lunar material substitution possible was conservatively estimated for two different temperature cycles. The hot cycle has a turbine inlet temperature of 1617 K; the cold cycle, 1100 K. Both use helium as the working fluid. Estimates of total specific mass and non-lunar specific mass for the two temperature ranges are shown in Figures 1.1-3 and 1.1-4.

The low temperature cycle is found to use less non-lunar material, but the increase in total mass is forty times larger than the decrease in non-lunar mass. Since neither this figure nor the fifty-to-one cost advantage of lunar materials is very precise, neither temperature cycle has a clear cost advantage over the other.

1.1.1.5 Rankine Cycle

Two working fluids, potassium and steam, were evaluated for the Rankine engine. Each fluid was evaluated with two temperature ranges. The more promising specific mass estimates are shown in Figures 1.1-3 and 1.1-4. The high temperature steam cycle was found to require least non-lunar material among the Rankine designs, but the Rankine cycle requires much more non-lunar material than any other option.

1.1.1.6 Stirling Cycle

Performance of the Stirling engine was estimated using the goals of a current NASA demonstration project. As shown in Figures 1.1-3 and 1.1-4, its specific mass is rather high. However, the Stirling requires little non-lunar material because of its low operating temperatures. The state of Stirling engine technology is advancing rapidly, so it should not be ruled out for SPS applications.

1.1.1.7 Concentrating Reflectors

Standard technology for large reflectors in space is vapor deposited aluminum on facets of Kapton plastic. A similar design which uses vapor deposited aluminum on facets of aluminum foil was selected. This requires virtually no non-lunar material.

1.1.1.8 Radiator Systems

Three radiator concepts which might be used to dissipate heat from active cooling systems are the heat pipe radiator (HPR), the liquid droplet radiator (LDR), and the moving belt radiator (MBR). The heat pipe radiator was selected for all active cooling needs in this study because it represents least technological risk and requires little non-lunar material. The moving belt radiator may have even lower non-lunar material requirements, but was considered too risky to use in this study. Nonetheless, the MBR concept is promising and deserves further study.

1.1.2 Structural Material

Aluminum was found to be the best structural material for this study because it is well understood, requires few non-lunar alloying agents, has good structural properties, and has relatively low mass per unit strength. Passive thermal control systems should suffice to keep eclipse-related thermal expansion within acceptable bounds.

1.1.3 Power Distribution

The Power Distribution System (PDS) controls, conditions, and transmits power from the conversion system to the microwave transmission system. It also provides energy storage for eclipse periods and handles fault detection and isolation. Major PDS subsystems are discussed below.

1.1.3.1 Main Buses

Four options were considered for the main power buses: sheet aluminum conductors, refrigerated buses, superconducting buses, or microwave transmission. The sheet aluminum option requires essentially no non-lunar material, and resistance losses can be made arbitrarily small by increasing the width or thickness of the bus. It is also the simplest to manufacture. Thus, the sheet aluminum option was selected.

1.1.3.2 DC-DC Converters

Previous studies(1,2,5) have used an oscillator-transformer-rectifier system for DC to DC conversion at the microwave antenna. Transformers are about 98% efficient, making the overall conversion process about 96% efficient. Transformers require active cooling, so non-lunar material is needed for pumps and coolant. Modern power electronics provide DC-DC conversion at over 98% efficiency without using transformers.(6) This increased efficiency reduces non-lunar mass needed for cooling and reduces the mass of the power conversion system, so this conversion technology was selected.

1.1.3.3 Energy Storage

Stored energy is required during eclipses to maintain essential systems and to keep RF amplifier cathodes hot. Three energy storage technologies were considered: nickel-hydrogen, sodium chloride, and counter-rotating flywheels wrapped with glass fiber. The flywheel system requires much less non-lunar material per stored kilowatt-hour than the other systems. The energy density of a glass-wrapped flywheel was conservatively estimated from the best demonstrated performance of a steel flywheel.

1.1.3.4 Electrical Slipring

All major SPS studies to date have used a slipring-and-brush assembly as the electrical rotary joint. Typically these designs have used silver molybdenum sulfide brushes on coin silver to minimize abrasion, drag, and electrical losses. However, this calls for a large mass of non-lunar materials. The silver in the sliprings (10.6 metric tons for a 5 GW SPS) could cause supply problems on Earth.

To decrease the mass of non-lunar material in the electrical slipring, a thin plating of coin silver on an aluminum slipring was selected. This design reduces the requirement for a 5 GW SPS from 11.8 metric tons of coin silver to 1.2 metric tons and gives equivalent electrical performance with less mass. It also reduces the total slipring mass.

1.1.4 Microwave Transmitter

The microwave transmitter includes RF amplifiers, waveguides, power distribution systems, phase control systems, thermal control systems, and structure. It may also include microwave optics such as a lens or reflector. Major components of the microwave transmitter are described below.

1.1.4.1 RF Amplifiers

Two RF amplifiers, a klystron and a magnetron, were redesigned for minimal use of non-lunar material. The magnetron was found to contain more non-lunar material, but its higher efficiency reduces the size of the power conversion system. Thus the magnetron is preferred if the power conversion system contains much non-lunar material, and the klystron is preferred if the power conversion system contains little non-lunar material. Using this criterion, the klystron is appropriate for the silicon planar and gallium arsenide concentrator systems; the magnetron is preferred for all others.

A third RF amplifier concept, the gyrocon, was dropped from consideration due to its low state of development.

1.1.4.2 Waveguides

The waveguide designed by General Dynamics(3), which has a conductive coating of aluminum in a foamed glass waveguide, was selected. It uses no non-lunar materials and remains within the close length tolerance over a wide temperature range. The alternative, an all-aluminum waveguide, could not maintain the required length in the severe thermal environment of the microwave transmitter.

1.1.4.3 Microwave Lens

A microwave lens design was developed so that SPS systems smaller than 2 GW could be economical. The lens is a Fresnel lens made of layers of aluminum wire mesh, and requires very little non-lunar material. Assuming that a reasonable design limit is that the lens constitutes no more than 50% of the SPS mass, then a power level as low as 200 MW can be achieved. Further study is required to verify the effectiveness of the lens design.

1.2 REFERENCES

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5. Hanley, G. M., Satellite Power Systems Concept Definition Study, Vols. 1-7, Rockwell International, Downey, Cal., NAS8-32475, Sept. 1980.

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7. Gillett, Steve, "Elements on the Moon", 1984.

2. SOLAR POWER CONVERSION SYSTEMS

Solar power conversion systems include transducers which convert sunlight to electrical energy, cooling systems to dissipate waste heat from the conversion process, and optical systems which filter or concentrate sunlight. Several power conversion concepts considered in this study require large concentrating reflectors and large active cooling systems. These subsystems are discussed in sections 2.7 and 2.8. Primary structure is discussed in chapter 3; it is not included in the mass estimates for any of the conversion systems.

Most of the power conversion designs discussed in this chapter are sized to produce 9 GW of electricity at the SPS bus. This is assumed to correspond to 5 GW of usable electricity on the ground, i.e., overall electrical efficiency is assumed to be 55.6%. This assumption is quite conservative, since earlier studies have achieved higher efficiency.

Two categories of solar power converters were considered in this study: photovoltaic (PV) systems and heat engines. A third category, thermionic conversion, was not considered because of its low efficiency, high mass, and large non-lunar material requirements.

Photovoltaic systems convert light energy directly to electrical energy. In all photovoltaic systems considered in this study, electricity is produced when photons of light form electron-hole pairs in a semiconductor material. The photovoltaic systems considered here are silicon planar, gallium arsenide concentrator, and thermophotovoltaic (TPV). Silicon planar was found to require least non-lunar material of all conversion systems considered, as shown in Table 2-1.

TABLE 2-1 PERFORMANCE OF POWER CONVERSION SYSTEMS

System	Max. Temp.	% Eff.	Total kg/kW	Non-lunar kg/kW
Si Planar	N/A	14.34	2.22	0.0000001
GaAs Conc.	N/A	9.5	6.93	0.0104
TPV	N/A	20.5	13.0	0.08
Brayton	1617 K	43	3.01	0.43
Brayton	1100 K	35	10.77	0.234
Rankine, steam	1644 K	42	6.80	1.23
Rankine, potass.	1600 K	31	3.14	1.47
Stirling	720 K	25	34.0	0.66

Photovoltaic systems require less non-lunar material than heat engine systems and are simple to design and build. The semiconductor cells must be protected from radiation or periodically annealed to repair radiation damage. The study team assumed that repeated annealing is effective, but was unable to determine whether this is the case. This matter is discussed further in sections 2.1 and 2.2.

Heat engines convert thermal energy into mechanical energy which can be converted to electricity by a generator. Solar power systems based on heat engines use concentrated solar radiation to heat a working fluid. Some of the energy of the working fluid is extracted by a mechanical device such as a turbine and the remainder is rejected by a radiator. Use of these systems in space can offer good efficiencies but can also present problems with complex machinery, temperature limitations, and maintenance.

Three types of heat engines were evaluated in this study: Brayton cycle, Rankine cycle, and Stirling cycle. A fourth type of heat engine, the magnetohydrodynamic (MHD) generator concept, appeared attractive due to mechanical simplicity and high theoretical efficiency. This concept was not thoroughly investigated, as it was felt that the high temperature required in the working fluid would make constructing such an engine from lunar materials impractical. A variation on the MHD concept, the liquid metal MHD generator, may prove suitable for construction from lunar materials after further development.

The Brayton cycle was evaluated at two ranges of temperatures. The lower temperature range had lower non-lunar mass despite lower efficiency and higher total mass. The high temperature cycle required many non-lunar materials in hot and/or fast moving parts of the engine. Neither has a clear cost advantage, assuming a cost ratio of fifty to one for non-lunar vs. lunar materials. Many estimates of lunar material substitution for the Brayton system were based on intuitive assumptions, since detailed information on complex high-temperature machinery often could not be found.

The Rankine cycle was studied using two different working fluids: potassium and water. The steam cycles gave better efficiencies but required more mass. The least non-lunar mass for the Rankine was achieved with the high temperature steam cycle. For both working fluids, the low temperature cycles required more non-lunar mass and more total mass. As for the Brayton system, many estimates of lunar material substitution in the Rankine were based on judgement rather than data.

The Stirling engine has only recently been seriously considered for space power systems. It is potentially easier to construct from lunar materials than either Brayton or Rankine, but is more massive overall than either. Because relatively little has been done with the Stirling as a space power system, this study considers only one temperature cycle of the Stirling engine.

2.1 SILICON PLANAR

2.1.1 Introduction

The silicon planar concept uses a large, modular array of silicon solar cells facing the sun. The sunlight is not concentrated nor is any attempt made to alter the spectrum. The cells are cooled by radiation from both surfaces. The cells are protected from harmful particle radiation by a thick window in front and a thick substrate behind. A modular box-frame structure maintains the shape of the solar array.

Because the silicon planar concept does not require concentrated sunlight, it has an advantage over all other power conversion systems in that the SPS does not have to be precisely oriented to face the sun. Rather than being oriented perpendicular to the ecliptic plane, the SPS can be oriented perpendicular to the plane of its orbit. This results in reduced stiffness and control requirements, so the structure and attitude control systems are less massive than for other conversion systems.

Several other studies have extensively reviewed the silicon planar option(1 - 4). Most details presented here are based on the Boeing reference design(1,2). Lunar material substitutions for the cell covers, substrate, and cell interconnects are based on the studies by General Dynamics(3) and MIT(4,p3.5). Cell panels and the panel support system have been redesigned for better efficiency and reduced non-lunar mass.

Silicon solar cells have been used for power production on satellites for many years. A 1984 Space Shuttle flight demonstrated a large solar array which is similar to the proposed SPS array. The vast amount of research that has been associated with silicon cells in space has resulted in a comparatively low technological risk for the silicon planar option. However, the useful lifetime of silicon cells in the GEO environment is questionable because repeated effective annealing silicon cells has not been demonstrated. High temperature silicon cells being developed now may solve this problem.(11)

If repeated annealing cannot be effective, some redesign of the silicon SPS will be needed. Two possible changes are to use thicker cover glass and substrates, or to use a larger array area. Using a larger array is preferred, since doubling array size gives the same end of life (EOL) performance as increasing cover glass thickness by a factor of ten.(12)

2.1.2 Design Description

The individual solar cells are 50 micron wafers of single crystal silicon fitted with aluminum contacts. These cells are connected into an 18 x 14 cell array which is placed between two plates of silicon dioxide glass to form a panel (Fig. 2.1-2). The glass serves to protect the cell from radiation damage as well as provide structural support. The 75 micron front plate is the optical cover and the 50 micron back plate is the cell substrate (Fig. 2.1-1). The optical cover is grooved to refract light around the grid fingers, increasing the effective efficiency by approximately 10%(2,p56).

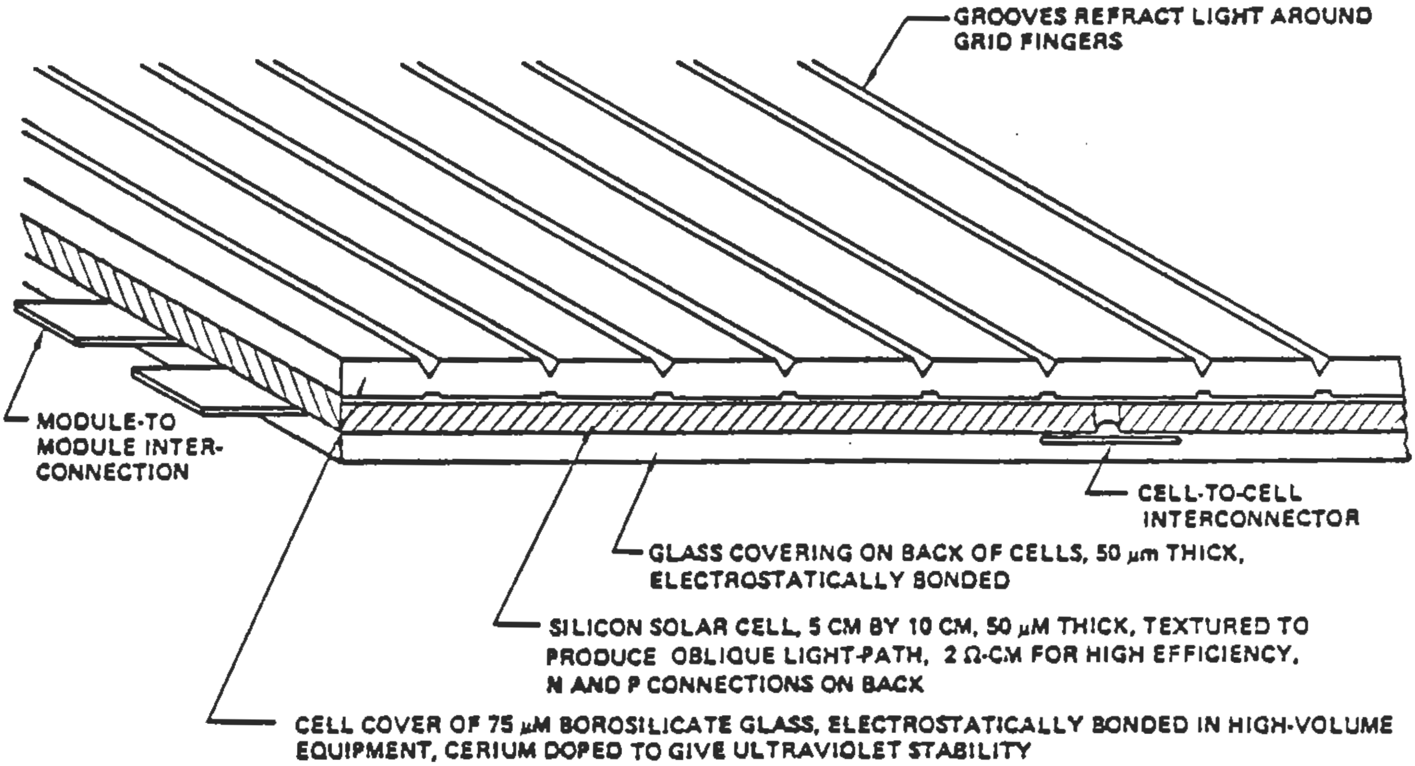

FIGURE 2.1-1, PANEL CROSS SECTIONAL VIEW

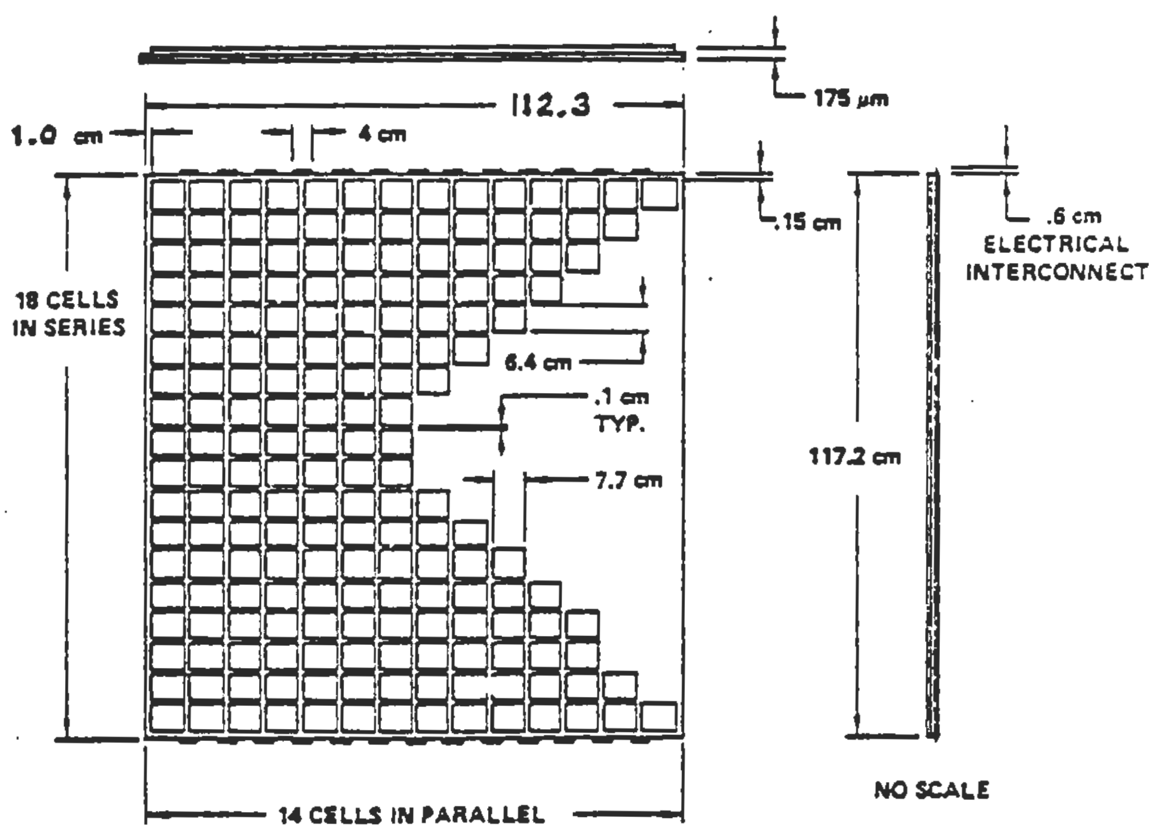

FIGURE 2.1-2, PANEL DIMENSIONS, TOP AND SIDE VIEWS

In earlier studies, the panels were connected and supported by Kapton tape. Substitution of aluminum tape for Kapton was considered, but the tape adhesive still required a large non-lunar mass. It was decided to use a grid of aluminum wires to support the panels. Panels are mounted in front of the grid. Unlike earlier studies, this design puts no tensile stress on the panels.

Cell substrate plates are 2 cm wider than cover plates, with holes drilled near the two sides. Support rods from the grid wires are connected to the panels at these holes. The panels are canted at 12 degrees to the SPS's longitudinal axis (Fig. 2.1-4) to reduce solstice cosine losses. The cells never face more than 12 degrees away from the sun because the SPS is rotated 180 degrees at each equinox. At edges of the grid the support wires are connected to catenary cables which attach to the primary bay structure (Fig. 2.1-3).

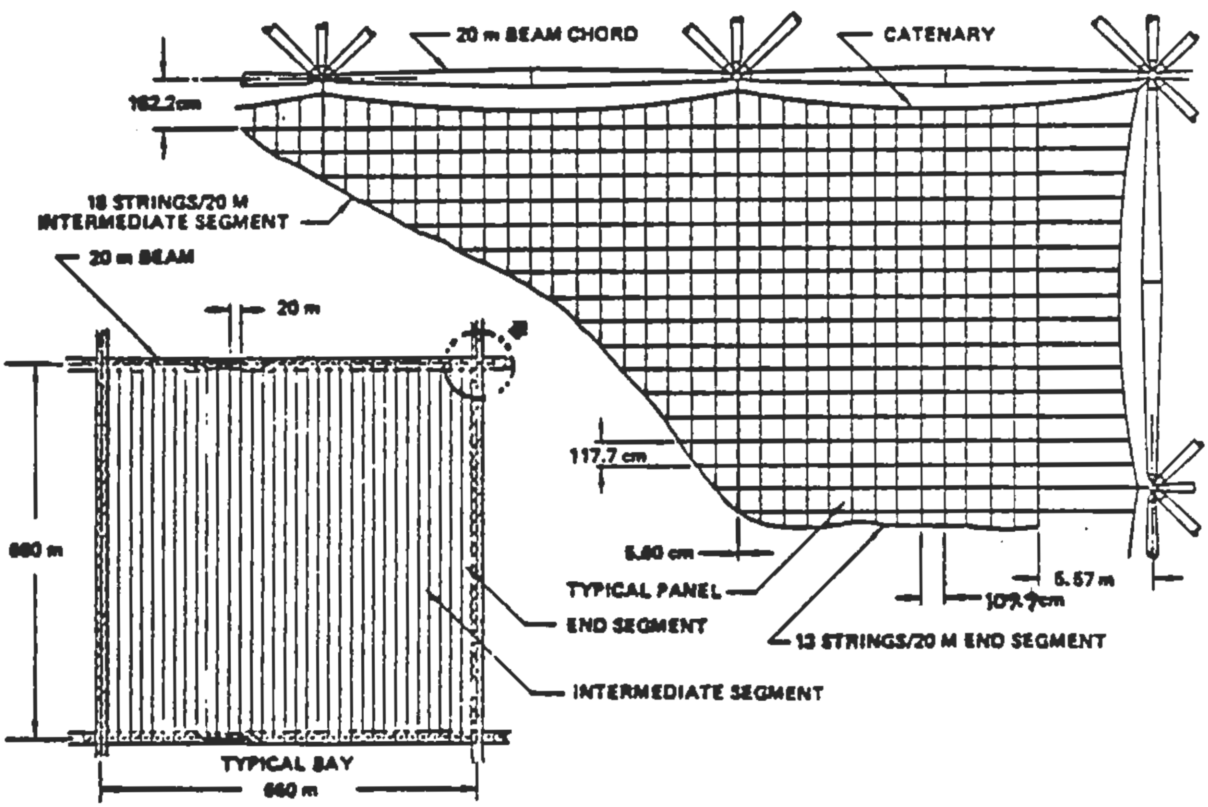

FIGURE 2.1-3, PANEL ARRAY AND PRIMARY BAY STRUCTURE

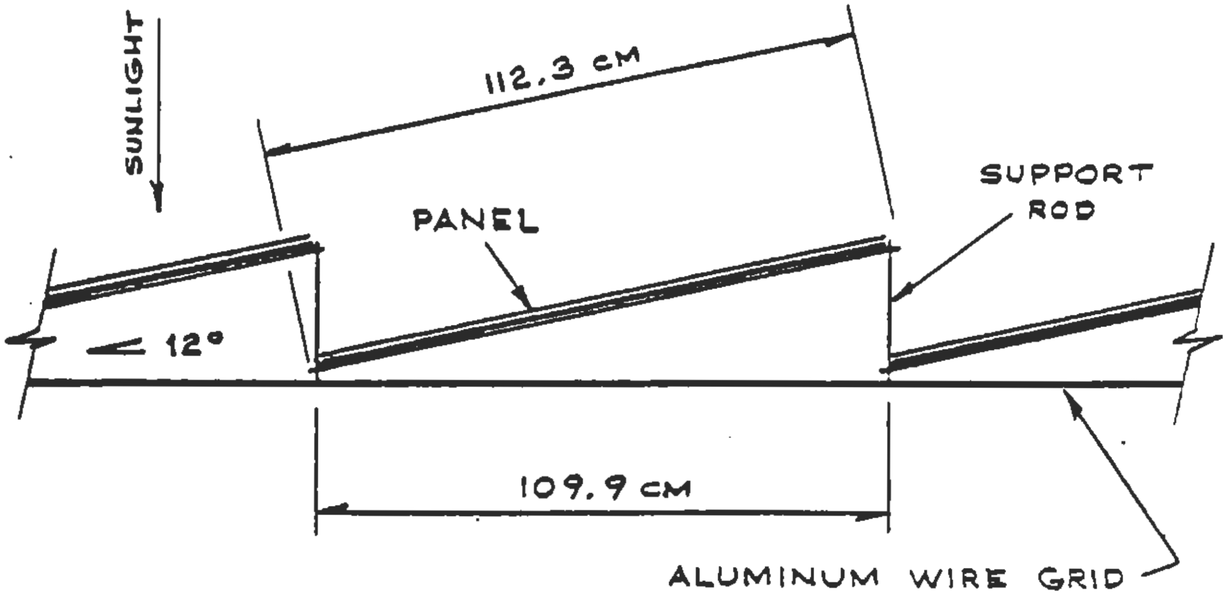

FIGURE 2.1-4, PANEL DETAIL

It is assumed that the silver grid fingers of the original design can be replaced with aluminum, based on Wang's results(5,pp580-583). Wang found that silver contacts could be replaced by copper if a layer of aluminum were used to prevent copper from diffusing into the silicon. In this design, the aluminum layer is thicker so no copper is needed.

The base efficiency for these cells is assumed to be 15.75% at AMO (Air Mass 0, refers to the solar spectrum unaltered by atmosphere, such as would be found at geo-synchronous orbit) and 25C, as in the Boeing study(2,p57). Allowing for increased efficiency due to the grooved cover plate and losses due to mismatch, radiation degradation, etc. (see 2.1.4.1) the minimum efficiency is taken to be 14.34%.

2.1.3 Design Summary

Table 2.1-1 presents a design summary for an SPS which delivers 9 GW to the spacecraft bus. The design is based on standard panels from the Boeing reference design which have been modifieu to include various material substitutions. The panel design is detailed in the technical discussion that follows. Note that the mass summary includes the catenary tensioning system but that no attempt has been made here to estimate the mass of the supporting structure.

TABLE 2.1-1 MASS ANALYSIS OF 9 GW SILICON PLANAR SYSTEM

ITEM	TOTAL MASS (kg)	COMPOSITION	NON-LUNAR MASS(kg)
Cell Covers	7.93E+6	Fused Silica	0
Solar Cells	5.50E+6	Silicon	<1
Substrate	5.42E+6	Fused Silica	0
Cell Interconnects	5.50E+5	Aluminum	0
Support Wires	3.45E+5	Aluminum	0
Connectors	3.24E+4	Aluminum	0
Tensioning System	1.20E+5	Aluminum	0
TOTAL MASS	1.99E+7		<1

Non-lunar specific mass: 1.1E-7 kg/kW
Total specific mass: 2.2 kg/kW

2.1.4 Silicon Planar Technical Discussion

2.1.4.1 Cell Efficiency Modifiers

The silicon cells used in this report are assumed to operate with the same base efficiency as those used in the Boeing study(2,p57). The loss factors are also assumed to be the same, except for the summer solstice cosine loss. The efficiency values and losses used are shown in Table 2.1-2.

The figure for non-annealable radiation degradation is grossly uncertain. The value from the Boeing study was assumed for comparison purposes.

TABLE 2.1-2 SILICON PLANAR EFFICIENCY MODIFIERS

MODIFIER	EFFICIENCY FACTOR
Grooved Cover Plate	1.10
Blanket Factors (String I2R, UV Losses, & Mismatch)	0.9453
Summer Solstice Cosine Loss (12 degrees)	0.9781
Aphelion Intensity Factor	0.9675
Temperature Losses (36.5 C @ Summer Solstice)	0.9540
30 Year Non-Annealable Radiation Degradation	0.970
Total Modifier	0.9106
Basic Cell Efficiency @ AMO, 25 C	15.75%
Effective Cell Efficiency	14.34%

2.1.4.2 Silicon Solar Cells

The Boeing reference design (1,2) assumed that 50 micron thick silicon solar cells with an efficiency of 15.75% would be available by 1985(2,p52). This has proved to be somewhat optimistic and such a cell has been projected for 1990 in this study.

The state of the art in high efficiency silicon solar cells is changing so rapidly that any attempt to predict future achievements is risky at best. In any event it seems unlikely that standard silicon cell technology will produce efficiencies in excess of 20% for this application. This would imply that the reference cell used in this study is conservative by a factor of no more than 30% (this assumes, of course, that the projected values for the reference cell are in fact achieved). In the silicon planar option, the solar conversion system constitutes a majority of the overall system mass. Any significant improvement in the efficiency of silicon cells would, therefore, be of great value. The non-lunar mass would not be greatly affected, however, as the conversion system accounts for only a small percentage of the SPS total of non-lunar material.

Of potential significance is the maturing of some alternate silicon cell approaches such as multi-gap and amorphous cells. Amorphous cells, to date, have achieved only relatively low efficiencies but are much less massive than standard cells(9,p67). Multi-gap cells offer higher efficiency but at the cost of higher mass(9,pp100-103). In either case it is the mass efficiency that will finally determine their relative merit. While neither of these two technologies are competitive now, it seems likely that one or both will become so in the future.

A good example of current laboratory achievements in silicon cell technology appears in a paper by S. Matsuda et al (6). The National Space Development Agency of Japan and the Sharp Corporation collaborated to produce BSFR (Back Surface Field, Back Surface Reflector) silicon cells approximately 50 microns thick that operate at 13.23% efficiency (at AMO and 28 C). In addition these cells were exposed to a 1 Mev electron fluence of 1E+15 e/cm2 after which they were annealed at 60 C for 24 hours thereby restoring them to 82% of BOL (Beginning of Life power).

2.1.4.3 Silicon Planar Dimensions and Masses for a 5 GW SPS

This section presents the numerical details of the silicon planar design and the references from which they are taken.

Solar Cells(4,p3.5)

Material = Single Crystal Silicon
Thickness = 50 microns
Length = 7.7 cm
Width = 6.4 cm

Cover Plate(4,p5.73)

Material = Fused Silica
Thickness = 75 microns
Length = 1.17 meters
Width = 1.10 meters

Substrate(4,p5.73)

Material = Fused Silica
Thickness = 50 microns
Length = 1.17 meters
Width = 1.12 meters

Panel to Wire Connectors

Material = Aluminum
Radius = .28 mm
Length = 1.3 meters per panel

Support Grid Wires

The radius of these wires was chosen to yield the same total mass as the aluminum tape considered earlier. This radius is quite conservative; each wire bears a tension of less than 6 newtons.

Material = Aluminum
Radius = .688 mm
Length = 2.30 meters per panel (average)

Cell Interconnects(4,p5.73-5.81)

Material = Aluminum
Mass Estimate = 1.47E-2 kg/panel

Tensioning System(4,p3.4)

Material = Aluminum
Mass Estimate = 3.21E-3 kg/panel

2.1.4.5 System Performance

Cells per Panel = 252
Cell Area per Panel = 1.24 square meters
Efficiency = 14.34%
Solar Constant = 1353 W/m^2
Panel Output = 240.6 W
Power Required to Bus = 9.0 GW
Panels per SPS = 37.4 million
Panel Area = 49.23 square kilometers

2.1.4.6 Mass Analysis

Densities (kg/m3)

Aluminum: 2.7E+3
Fused Silica: 2.2E+3
Silicon: 2.36E+3

Item	Panel Mass(kg)	System Mass(kg)
Photovoltaic Cells	1.47E-1	5.50E+6
Cover Plate	2.12E-1	7.93E+6
Substrate	1.45E-1	5.42E+6
Interconnects	1.47E-2	5.50E+5
Support Wires	9.23E-3	3.45E+5
Panel to Wire Connectors	8.65E-4	3.24E+4
Tensioning System	3.21E-3	1.20E+5
TOTAL	5.32E-1	1.99E+7

2.1.5 Silicon Planar References

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2.2 GALLIUM ARSENIDE CONCENTRATOR SYSTEM

2.2.1 Introduction

The gallium arsenide (GaAs) concentrator option is based on replacing the bulk of the solar cells used in a planar configuration with an aluminum concentrator. Such a system would typically have a concentration ratio in the 100 - 1000 range and is therefore suitable to a strategy of minimizing the use of non-lunar materials. Although GaAs is not available on the Moon, its ability to operate efficiently at high concentration ratios and high temperatures allows a system to be designed which employs minimal amounts of non-lunar mass (less than one percent).

Like silicon, GaAs cells must be protected from radiation or must be periodically annealed. GaAs is more likely to be annealable than silicon, but questions remain as to whether the process is workable.(18) An advantage of the GaAs concentrator concept is that its mass is not greatly affected by whether GaAs cells can be repeatedly annealed. Two factors account for this: the high radiation resistance of GaAs (roughly three times that of silicon) and the relatively small size of the cells. Thus, very thick cover glass could be added without significantly affecting the mass (total or non-lunar) of the SPS. It was assumed here that GaAs could be annealed.

Three major areas were considered in this analysis: the concentrator, the cooling system and the optimization of concentrator dimensions. These areas are addressed in the technical discussion (section 2.2.4), along with supporting comments on the cell itself, the structure and manufacturing.

2.2.2 Design Description

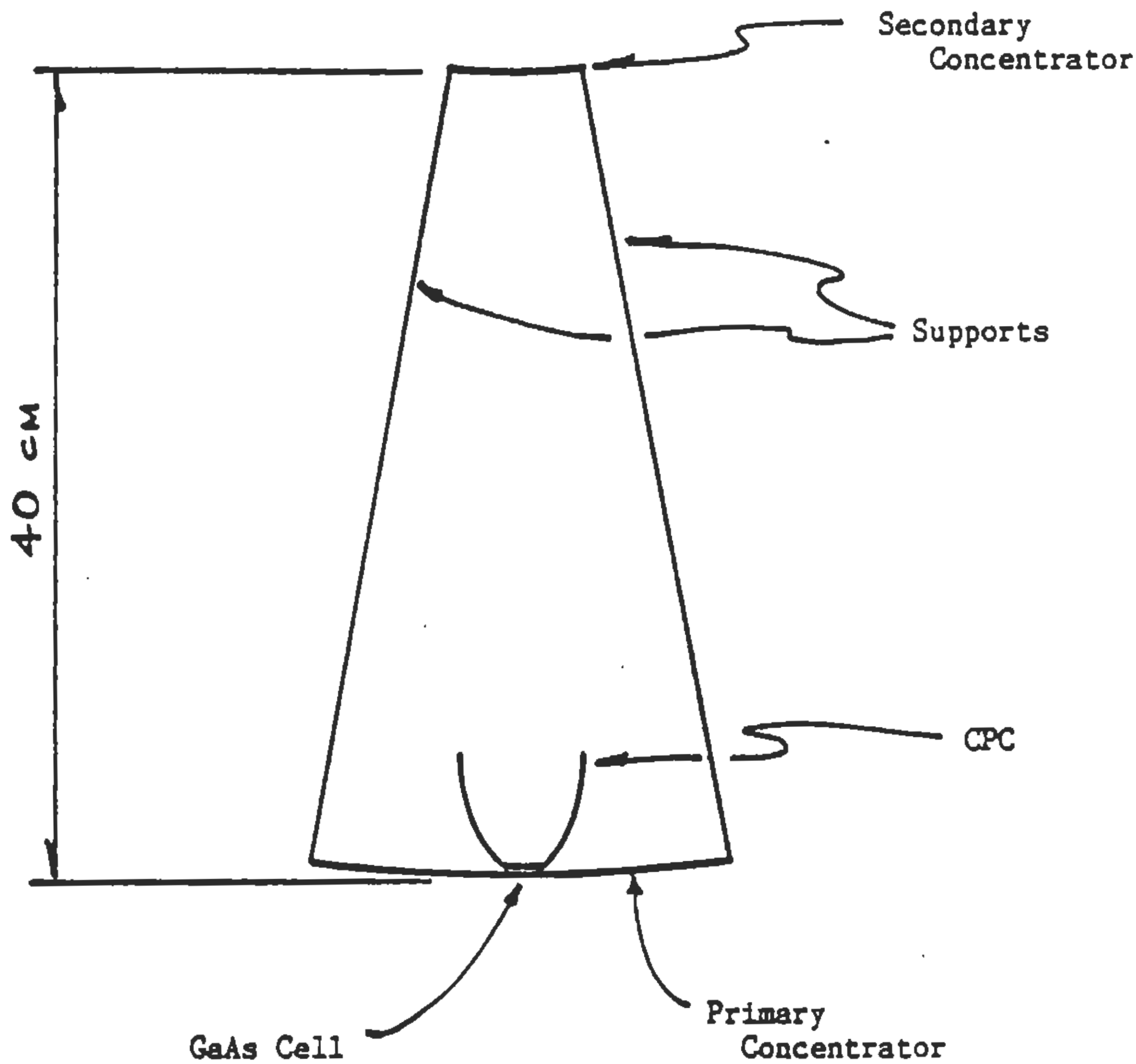
The concentrator system used in this study consists of a parabolic dish for the primary reflector and a hyperbolic mirror for the secondary reflector. This makes up a typical Cassegrainian concentrator. In addition a small Compound Parabolic Concentrator (CPC) has been added around the photovoltaic cell as a tertiary reflector. This, along with an oversized secondary reflector, is designed to compensate for imperfections in the mirrors which cause dispersion in the reflected light. (See Fig. 2.2-1)

The secondary reflector is located at a point where the radius of the image striking it is the same as that of the photovoltaic cell. This point is very near the focal point of the primary reflector. The two reflectors are connected by four aluminum legs.

The Ga's photovoltaic cell is attached directly to the primary reflector which serves as a radiator. The Design Summary allows for a thin layer of possibly non-lunar material between the cell and the primary reflector to serve as an insulator and/or adhesive.

All three reflectors are made of aluminum and have a coating of vapor deposited aluminum to enhance reflectivity. In addition the back surface of the primary reflector has been anodized to increase the emissivity. This layer of aluminum oxide greatly increases the capacity of the radiator and is a significant fraction of the concentrator mass (see 2.2.4.3.1).

The concentrators are supported by a wire grid with the same mass per unit area as that of the silicon planar configuration.



Side view diagram - not to scale.

FIGURE 2.2-1, GALLIUM ARSENIDE CONCENTRATOR SYSTEM

2.2.3 Design Summary

Table 2.2-1 summarizes the results from the technical discussion that follows. The overall efficiency is 9.5%. Some other key values are 0.84 reflectivity for the mirrors, 0.8 emissivity for the radiator and 15 percent efficiency of GaAs cells operating at 200 C and with a concentration ratio of 260. The mass calculations are based on 2.228 billion cells each supplying 4.04 watts to the power distribution system for a total of 9.00 gigawatts.

TABLE 2.2-1 MASS ANALYSIS OF 9 GW GALLIUM ARSENIDE CONCENTRATOR SYSTEM

ITEM	TOTAL MASS (kg)	COMPOSITION	NON-LUNAR MASS(kg)
Primary Reflector	4.72E+7	Aluminum	0
Secondary Reflector	1.88E+6	Aluminum	0
Secondary Supports	2.29E+6	Aluminum	0
Tertiary Reflector	3.01E+6	Aluminum	0
High Reflectivity Coating	2.36E+5	Aluminum	0
High Emissivity Coating	6.95E+6	Aluminum Oxide	0
Glass Cover Plate	1.18E+5	Fused Silica	0
GaAs Photovoltaic Cell	7.15E+4	GaAs	7.15E+4
Insulator/Adhesive	2.23E+4	Unspecified	2.23E+4
Support Wires	5.23E+5	Aluminum	0
Tensioning System	1.04E+5	Aluminum	0
TOTAL MASS	6.24E+7		9.38E+4

Non-lunar specific mass: 1.04E-2 kg/kW
Total specific mass: 6.93 kg/KW

2.2.4 Gallium Arsenide Technical Discussion

2.2.4.1 System Performance

Solar Constant	1.353 kW/m2
Collector Area	3.142E-2 m2
Collector Efficiency	9.5%
Cell Output	4.04 W
Power Required to Bus	9.00 GW
Collectors per SPS	2.228 billion
Total Collector Area	70.0 square kilometers

2.2.4.2 Concentrator Design

Many types of concentrators were considered, however, the Cassegrainian appears to be far superior in light of the design limits. Lenses were ruled out because they tend to be massive and may require large amounts of non-lunar material. A simple parabolic dish reflector offers some advantage over the Cassegrainian, however, it would add considerable difficulties to the design of the cooling system because the radiator would be facing the sun and would be separate from the primary reflector. The Compound Parabolic Concentrator (CPC) also has advantages over the Cassegrainian, in particular its tolerance to pointing error and its one piece construction.

These benefits, however, are far outweighed by its disadvantages. The CPC, particularly at high concentration ratios, is a long and therefore massive device. In addition its shape does not lend itself to effective radiation of heat and a secondary radiator would be needed. As a result, unless pointing accuracy or precision construction prove to be insurmountable problems for the Cassegrainian, the CPC is disqualified on grounds of weight.

2.2.4.3 Radiator Design

Approximately 80 percent of the light absorbed by the solar cell must be disposed of as waste heat. Two methods of achieving this have been considered in detail. The first, a disk radiator, has the advantage of all aluminum construction and simplicity of design but limits the system to small cell sizes and/or relatively low concentration ratios. The second option, heat pipes, allow the use of larger cells and higher concentration ratios but they require significant amounts of non-lunar material and have a considerable technological risk associated with them.

Both radiator systems are compatible with a simple annealing scheme. A sheet of reflective material is positioned behind cells to be annealed. This reflects heat back to the radiator, raising the temperature of the cell. This annealing system has not been designed in detail, but should require virtually no non-lunar material.

2.2.4.3.1 Disk Radiator

The major advantage of the disk radiator is that it is already present in the form of the primary reflector. By making the primary reflector perform double duty as both reflector and radiator it is possible to cool the solar cell without any additional structures.

As aluminum alone is not an efficient radiator (15,pp2-7), it is necessary to coat the back surface of the primary reflector with a high emissivity material. Aluminum oxide was chosen for this purpose because it has a reasonably high emissivity, consists totally of lunar materials and is easily applied by anodization. The thermal and mechanical properties of Al2O3 are, however, considerably different from those of aluminum. It was necessary, therefore, to require that the radiator be thick with respect to the Al2O3 coating. It is estimated that 25 microns of Al2O3 will be required to give the desired emissivity of .8 (14,pp1198-1201), however this emissivity value is heavily dependent on the method of anodization. A precise knowledge of this value would be essential for a more accurate determination of the mass efficiency of the concentrator.

2.2.4.3.2 Heat Pipes

Two different configurations were considered for the application of heat pipes. The first involved placing several heat pipes as ribs in the primary reflector. This had all the advantages of using the primary reflector as the radiator, most notably little or no reradiation between unit cells.

Unfortunately there were also several problems with this design. The very complex temperature gradient in the radiator was felt to present severe problems to the design of the primary reflector. The resulting complex thermal expansion pattern and fatigue resulting from thermal cycling seem likely to degrade the optics. In addition, as the radiator design (radial heat pipes on a one sided disk) is inefficient, it was necessary to make the dish relatively thick (and therefore massive) to minimize the temperature drop. Finally, it appeared that mass efficiency increased significantly as the number of heat pipes increased and as concentrator size decreased. Optimizing in this way alleviated the first two problems to some degree but also decreased the minimum heat pipe radius. For a unit cell radius of 10 - 100 cm the heat pipe dimensions became so small as to be impractical. Larger heat pipes could be used but that would incur a severe mass penalty. For these reasons focus was placed on the second option, a single finned heat pipe projecting axially back from the primary reflector. The major benefits of this approach are simplicity and economy of scale (using a larger heat pipe). At the same time, however, analysis of the reradiation became difficult. Reradiation between fins (taken here to be coaxial disks) on a single pipe was easily evaluated (6,p635) but for radiation between concentrators no simple approximation was found. As a standard of comparison it was estimated that 25 percent of the heat emitted from a heat pipe/fin assembly would be reabsorbed by other units. Using this assumption and foregoing the requirement that the fins be thick relative to any high emissivity coating gave mass efficiency results that were approximately equivalent to that of the disk radiator.

In addition to the problems mentioned above there were some difficulties encountered that were common to both applications of heat pipes. The first was transferral of heat from the comparatively large solar cell to the much smaller radius heat pipe. The most promising method seemed to be wrapping the evaporator portion of the pipe into a spiral disk which would cover the back of the solar cell. There is not much information, however, on complex geometries for heat pipes and this application remains to be proven acceptable. A second problem lies in the choice of a proper structural material / working fluid combination for this temperature (assumed to be approximately 470 K). Water seems to be the most likely candidate for a working fluid, particularly when striving for a low non-lunar mass. Unfortunately, few lunar materials are compatible with water at temperatures above 100 C (1,pp103-106) because generation of even small amounts of oxides or non-condensable gas can seriously impair the performance of a heat pipe(1,p103). There is evidence to suggest that some steels are workable but this is not yet sufficiently documented(1,pp110-111). This leads to a final problem: the lifetime of the heat pipe. Currently most space applications require lifetimes of 5 to 10 years and even that has proved to be a real challenge for heat pipe design(1,pp108-110). The 30 year lifetime required by the SPS will place a much greater strain on the heat pipes as well as on the verific¬tion process.

2.2.4.4 Construction

For analysis, the primary reflector is approximated as a circular dish. In practice, however, it is more likely to be constructed as a square or hexagonal section of a panel to facilitate assembly. As these panels will serve as radiators they will have a relatively high operating temperature and will therefore be subject to a fair amount of thermal expansion. In addition, due to the occasional eclipsing of the SPS, there will be thermal cycling. This combination will result in stress on the secondary reflector as well as the SPS structure. This is expected to be one of the more severe problems to be dealt with in the mechanical design of the system.

The design used here assumes that the primary reflector has constant thickness for ease of manufacturing. A tapered disk would be more efficient as a radiator, reducing total mass by over half. Members of the Advisory Committee believe that an effective means of manufacturing tapered primary reflectors could be found. Thus, the total mass estimated here is probably conservative by a factor of two.

2.2.4.5 Optimization

The thickness of the parabolic dish was the limiting parameter for optimization of the concentrator size. System mass appeared to increase almost linearly with thickness, suggesting that thickness should be as small as practical. It was also necessary, however, to make the thickness large compared to that of the high emissivity coating. As a result the thickness was set at 0.25 mm (10 times that of the coating).

With thickness fixed, the maximum concentration ratio (CR) at which the cell can be effectively cooled is a function of disk radius. The maximum CR was found to increase as disk radius decreased. High concentration ratio corresponds to low non-lunar mass, so a small disk size is preferred.

Scanty data made it infeasible to optimize the solar cell efficiency with respect to concentration ratio and operating temperature. It was assumed that reasonable progress in technology could produce a GaAs cell 50 microns thick that would have an operating efficiency of 15 percent after all losses (see Table 2.1-2) at 470 K and with a concentration ratio in the 100 - 500 range. Hot spots within the solar cell were not considered in detail so an intermediate value was thought to be best for the concentration ratio. A CR of 260 was chosen as it was in the middle of the range and proved to be convenient for calculations. If the desired efficiency can not be achieved with this value a sub-optimum concentration ratio could be used. The mass of the solar cell is a very small fraction of the concentrator mass and if the concentration ratio was decreased by a factor of two or three the non-lunar mass would still be very good. Furthermore it would probably still be within the error bars of the values shown.

2.2.4.6 Gallium Arsenide Concentrator Parameters

Primary Reflector/Radiator

Design = Parabolic Dish
Material = Aluminum
Radius = 10 cm
Thickness = 0.25 mm
Focal Length = 40 cm

Secondary Reflector

Design = Hyperbolic Dish
Material = Aluminum
Radius = 3.15 cm
Thickness = 0.10 mm
Distance from Primary = 38.2 cm

Supports for Secondary

Number = 4
Length = 38.8 cm
Radius = .28 mm

Tertiary Reflector

Design = CPC
Material = Aluminum
Maximum Radius = 3.15 cm
Minimum Radius = 0.62 cm
Height = 3.0 cm
Thickness = 0.10 mm

High Reflectivity Coating

Location = All Mirror Surfaces
Material = Vapor Deposited Aluminum
Thickness = 1 micrometer
Reflectivity = 0.84

High Emissivity Coating

Location = Radiator Face
Material = Aluminum Oxide
Thickness = 25 micrometers
Emissivity = 0.8

Glass Cover Plate

Material = Fused Silica
Radius = 6.2 mm
Thickness = 0.2 mm

Photovoltaic Cells

Material = Gallium Arsenide
Radius = 6.2 mm
Thickness = 50 micrometers

Insulator/Adhesive

Location = Back Surface of Photovoltaic Cell
Material = Unspecified non-lunar
Radius = 6.2 mm
Thickness = 25 micrometers

2.2.4.7 Mass Analysis for a Single GaAs Concentrator Unit

The densities shown below were assumed to compute the component masses shown in Table 2.2-2.

Densities(kg/m3):

Aluminum: 2.7E+3
Aluminum Oxide: 3.97E+3
GaAs: 5.32E+3 (2,p305)
Fused Silica : 2.2E+3 (17,p68)

TABLE 2.2-2 MASS ANALYSIS OF A SINGLE GALLIUM ARSENIDE CONCENTRATOR

ITEM	MASS (kg)	COMPOSITION	kg/kW
Primary Reflector	2.12E-2	Aluminum	5.25E+0
Secondary Reflector	8.42E-4	Aluminum	2.08E-1
Secondary Supports	1.03E-3	Aluminum	2.55E-1
Tertiary Reflector *	1.35E-3	Aluminum	3.34E-1
High Reflectivity Coating	1.06E-4	Aluminum	2.62E-2
High Emissivity Coating	3.12E-3	Aluminum Oxide	7.72E-1
Glass Cover Plate	5.31E-5	Fused Silica	1.31E-2
GaAs Photovoltaic Cell	3.21E-5	GaAs	7.95E-3
Insulator/Adhesive **	1.00E-5	Unspecified	2.48E-3
Unit Lunar Mass	2.77E-2		6.86E+0
Unit Non-lunar Mass	4.21E-5		1.04E-2
Total Unit Mass	2.77E-2		6.87E+0
Non-lunar Fraction	.0015		

* Approximated by a truncated cone
** Mass is approximate.

2.2.4.8 Energy Conversion Process

Incoming Light

Intensity = 1.353 kW/m2
Blocked by Secondary Reflector: 10% (4,p109)
Absorbed by Primary Reflector: 16% (16,p1593)

Light to Secondary Reflector

Blocked by Tertiary Reflector: ~0
Absorbed by Secondary Reflector: 16% (16,p1593)

Light to Photovoltaic Cell

Absorbed by Tertiary Reflector: ~0
Reflected from Cell Face: 25% (3,p1616)
Converted to Electricity: 15% (3,p1616)
Rejected as Waste Heat: 60%

Thus the sunlight projecting onto the concentrator is converted to electricity with an overall efficiency of:

$$(0.90)*(0.84)*(0.84)*(0.15) = .095 \text{ or } 9.5\%$$

The 15% efficiency of the GaAs cell assumes the losses listed in Table 2.1-2.

2.2.4.9 High Reflectivity Coating

This study assumes a value of .84 for the reflectivity of vapor deposited aluminum. This is based on the value presented in a report by TRW (16,p1593) for 'advanced aluminum' technology. The reference does not define the term 'advanced aluminum' so the relationship between TRW's approach and vapor deposited aluminum is not clear. Due to similarity in the applications, however, and a general lack of information on this subject, this number was assumed. Other references(11,p267) and members of the Advisory Committee have suggested that a higher reflectivity, perhaps as high as .92, is probable for aluminum. Since the assumed coating absorbs (or transmits) nearly 30 percent of the incoming light, it is recommended that efforts to design a concentrator based SPS should include efforts to identify the maximum achievable reflectivity.

2.2.5 Gallium Arsenide Concentrator References

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2.3 THERMOPHOTOVOLTAIC (TPV) CONVERSION

Photovoltaic cells do not efficiently use the solar spectrum, which resembles the spectrum of a 6000K blackbody. Thermo-photovoltaic energy conversion refers to an energy conversion system which uses concentrated sunlight to maintain a radiator at about 2200K. Silicon solar cells can convert the lower temperature spectrum to electricity at a theoretical efficiency of up to 75%. Experimental efficiencies of 40% have been achieved.(4) A schematic diagram of the TPV process is shown in Fig. 2.3-1.

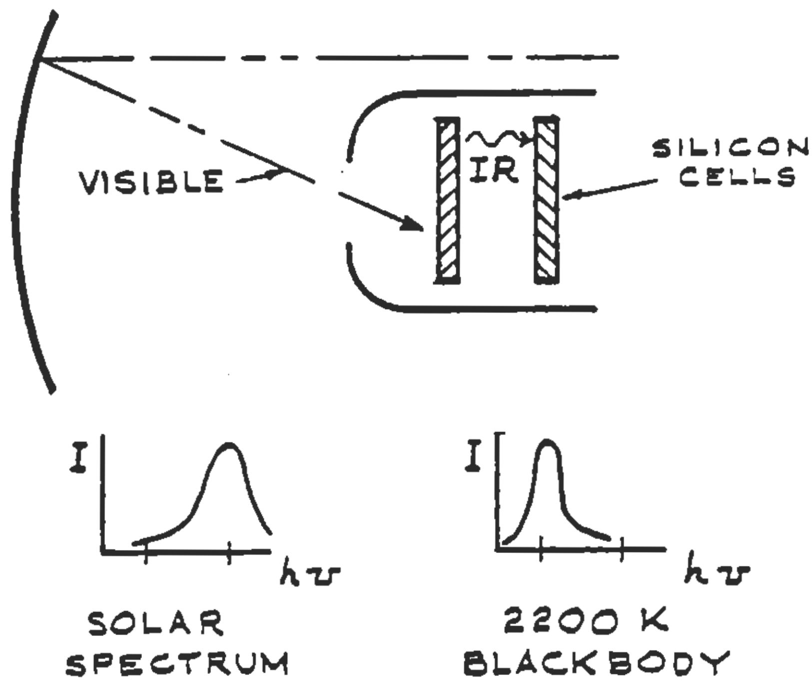

Diagram - not to scale

FIGURE 2.3-1, THERMOPHOTOVOLTAIC CONVERSION PROCESS

2.3.1 TPV Design Summary

Table 2.3-1 summarizes a design for a 5 GW SPS which requires 9 GW at the spacecraft bus. The design was developed usi~g a computer model whose parameters were optimized for minimal non-lunar mass. Structural mass for the radiator cavity is not included, but is assumed to be small compared to the concentrator mass. The maximum radiator temperature is limited to 2300K by the evaporation rate of tungsten.(5) The minimum non-lunar mass occurs at a cell temperature of 405 K, for an overall conversion efficiency of 20.5% from intercepted sunlight to electrical power.

TABLE 2.3-1 MASS ANALYSIS OF 9 GW TPV SYSTEM

ITEM	MASS (kg)	COMPOSITION	NON-LUNAR MASS (kg)
Cooling	113,000,000	Aluminum, steel, water	676,000
Concentrator	3,520,000	Aluminum	0
Radiator	30,600	Tungsten	30,600
Solar array	52,400	Silicon, Aluminum	<1
Cell backing	14,500	1-mil Kapton	14,500
	116,618,000		721,101

Non-lunar specific mass: 8.01E-2 kg/kW
Total specific mass: 12.96 kg/kW

A parametric model of the thermophotovoltaic conversion system has been developed (section 2.3.3). The model includes the solar concentrator and the cooling system. Output includes the total mass and the earth mass for each major subsystem and for the total TPV converter. Assumptions used in the model are described below.

2.3.2 TPV Design Description

The most massive subsystem is the cooling system. Heat pipe radiators and liquid droplet radiators were considered. For LDR, the heat rejection temperature (less than 400 K) would require silicone oil as the working fluid, so the non-lunar mass of the LDR was unacceptably high. Heat pipe radiators at this temperature can use water in the main coolant loop and as heat pipe working fluid. Heat pipe radiators were selected for the TPV design because of their more advanced development and lower non-lunar mass requirement.

The second most massive system is the concentrator. An all-aluminum concentrator like that described in section 2.7 was selected, so no non-lunar materials are required. The TPV model assumes the concentrator's specific mass does not change for different concentration ratios (CRs). This assumption is overly simplistic, but should not cause significant errors in this application (CR < 400).

Silicon solar cells were assumed because they can be nearly 100% lunar in composition and they can be manufactured in large quantities. Conventional cells are not optimum for TPV converters. TPV-optimized cells should be highly transparent to long-wavelength radiation and should have different spectral response than conventional cells. TPV cells should be thin for better cooling and for better long-wave reflection. They should be designed to operate at high radiant intensity. The design assumes cells which are designed for TPV conversion, but which are not close to the achievable limit. Note that cells used in a TPV converter do not require periodic annealing because the body of the converter completely surrounds the cells, protecting them from harmful radiation.

Solar cell interconnections are assumed to be aluminum. Detailed design of the interconnection pattern is beyond the scope of this report.

The cell backing provides thermal contact and electrical insulation between the cells and the cooling system. 1-mil Kapton was used by Horne(4), but lunar materials such as glass might be used instead.

The radiator is a thin sheet of fabric woven from tungsten wire. Evaporation limits the maximum operating temperature to less than 2300 K.(5)

An early version of the TPV design included a thick radiator with enough heat capacity to provide power during eclipses. This was rejected because it gives a mass penalty to the TPV concept and would make it difficult to directly compare the TPV concept with other power conversion systems. However, this option could make TPV more attractive if power during eclipses becomes a high priority.

The major problem with TPV conversion in space is cooling the cells. There are two facets to this problem: massive cooling systems, and solar cells which are too opaque at long wavelengths. High specific mass in cooling systems is a common problem in spacecraft. The poor transparency of cells at long wavelengths produces heat in the cells without producing power (see TPV Technical Discussion, section 2.3.3), so the cooling load is increased.

Three solutions are possible: find a cooling system which has a low specific non-lunar mass, develop a cell which is more transparent at long wavelengths, or develop a cell which is more tolerant of high temperatures. All three options are being investigated in industry and academe, so the status of TPV should improve.

2.3.3 TPV Technical Discussion

The primary characteristic of a photovoltaic (PV) cell is the band-gap energy, E_g, of the semiconductor material in the cell. For example, silicon has a band-gap of 1.13 eV. A photon's energy, E_p, may have any value. For a particular photon striking the cell, if $E_p > E_g$, then the photon will probably be converted to an electron-hole pair. From each electron-hole pair produced, at most E_g energy can be converted to electricity; the remainder of the photon's energy will appear as heat in the cell. Infrared photons with $E_p < E_g$ will usually pass through the cell without interaction, though some will be absorbed as heat energy in the cell.

Typically PV cells are used in normal sunlight, which is similar to the spectrum of a blackbody at about 6000 Kelvin. Most of the power of sunlight is carried by photons of about 2.7 eV. Thus, since only E_g/E_p of a photon's energy can be converted to electricity, most of the power of sunlight is wasted by typical PV cells.

In a TPV converter, sunlight is used to heat a radiator which illuminates PV cells. The radiator's temperature is much lower than 6000 K, so much of its radiated power is carried by photons whose energies are only slightly above the cells' band-gap. Thus, more of the power can be converted to electricity.

The efficiency of a TPV converter is dependent on many parameters. A spreadsheet model of a TPV converter has been developed to study the effects of these parameters, and to estimate whether a TPV converter would be practical with reasonable parameter values.

The model was built with two Multiplan(8) spreadsheets. The first, SPECTRUM, computes values which are dependent on the solar cell material (e.g., Si) and on the temperature of the radiator. The second, TPVMODEL, uses the values from SPECTRUM to compute masses and efficiencies of the whole converter system.

Table 2.3-2 is an example of part of SPECTRUM, using silicon cells and a 2300 K radiator.

TABLE 2.3-2 USABILITY OF 2300 K SPECTRUM FOR SILICON CELLS

1.13E+00 eV	E_g: band gap of cells
2.30E+03 K	T_rad: Radiator temp
5.68E+00	g_t: E_g/(k*T_rad), bandgap over average photon energy
2.68E+05 W/m^2	I_g: Intensity above band gap
2.17E+05 W/m^2	I_u: Usable intensity above band gap
1.59E+06 W/m^2	I_rad: Total intensity at radiator
1.69E-01	f_g: Fraction of power above E_g
8.10E-01	f_u: Fraction of power above E_g which is usable
1.37E-01	f_t: Fraction of total power which is usable

SPECTRUM uses a numerical calculation of a blackbody spectral power distribution. This distribution is integrated from the cell material's band-gap energy to infinity to give an estimate of I_g, the spectral intensity which could be converted to electron-hole pairs if the cells' quantum efficiency were unity. Another section integrates the spectral intensity times the ratio of the band-gap energy to photon energy to give an estimate of I_u, the potential electron-hole intensity that could be converted to electricity. These two values are divided by the total radiant intensity of the radiator to yield f_g, the fraction of the total power which can ideally be absorbed as electron-hole pairs, and f_u, the fraction of electron-hole pair energy which can ideally be converted to electricity.

Table 2.3-3 is an example of TPVMODEL, using the SPECTRUM model in Table 2.3-2.

TABLE 2.3-3 TPV MASS AND PERFORMANCE MODEL

9.00E+09	W	P_out: Power Output
4.38E+10	W	P_in: Power Input through window
4.37E+11	W	P_rad: Radiator Power
3.06E+05	M**2	A_rad: Radiator Area
3.06E+04	Kg	M_rad: Radiator Mass
3.06E+04	Kg	M_rad_e: Radiator Earth Mass
3.08E+10	W	P_cool: Power to Cooling
1.13E+08	Kg	M_cool: Cooling Mass
6.76E+05	Kg	M_cool_e: Cooling Earth Mass
3.52E+06	Kg	M_conc: Concentrator Mass
0.00E+00	Kg	M_conc_e: Concentrator Earth Mass
2.91E+05	M**2	A_cells: Cell Area
5.24E+04	Kg	M_cells: Cell Array Mass
1.45E+04	Kg	M_cells_e: Cell Array + Backing Earth Mass
1.16E+08	Kg	Total Mass
7.21E+05	Kg	Total Earth Mass
1.69E-01		f_g: Fraction of power which is above E_g
8.10E-01		f_u: Fraction of power above E_g which is usable
3.34E-01		f_e: Fraction of f_u which becomes electric power
4.50E-01		p: Avg. view angle fraction of cells from radiator
1.00E+00		emf_rad: Radiator Earth Mass Fraction
0.00E+00		emf_conc: Concentrator Earth Mass Fraction
6.00E-03		emf_cool: Cooling Earth Mass Fraction
1.00E-08		emf_cells: Cell Earth Mass Fraction
4.86E+01		A1: P_rad / P_out
3.42E+00		A3: P_cool/P_out
8.00E-03	Kg/W	A4: Cooling Spec. Mass (@60 C.)
4.87E+00		A5: P_in / P_out
1.00E-01	Kg/M**2	A6: Concentrator Spec. Mass
2.71E-01		A7: Cell efficiency above band gap
1.30E-01	Kg/M**2	A8: Cell Array Specific Mass
0.00E+00		A10: Cooling Power / Power to Cooling
9.20E-01		A12: Concentrator efficiency
FALSE		A13: Using thermal storage for eclipse power?
5.00E-02	Kg/M^2	A14: Cell Backing Specific Mass (1-mil Kapton)
1.35E+03	W/M**2	S: Solar Constant
5.00E+02	Suns	I_c: Cell Intensity, max 500 suns
4.05E+02	K	T_cells: Cell temperature
2.30E+03	K	T_rad: Radiator Temperature
1.59E+06	W/m^2	I_bb: Intensity at ideal blackbody surface
1.43E+06	W/M^2	I_rad: Intensity at actual radiator surface
9.00E-01		e: Radiator Emissivity
1.00E-01		R_c: Cell front surface reflectivity
8.50E-01		R_w: Average reflectivity of non-cell sky, incl. window
9.60E-01		t_r: long-wave reflectance through cells (twice)
1.64E-01		a: (1-r); fraction of radiator energy not recycled
3.86E+02		CR: required concentration ratio

TPVMODEL uses cell properties and their temperature dependence to estimate the fraction (f_e) of the potentially convertible power which actually becomes electricity at a given cell temperature. The remainder of the power above the cells' band-gap energy is part of the cooling load.

Other parameters used in TPVMODEL are the front-surface reflectivity of the cells, the emissivity of the radiator, and the optical configuration of the radiator cavity. A crucial parameter is the fraction of infrared power (from photons with energies below the band-gap) which passes through the cells and is reflected back to the radiator; absorbed infrared photons add to the cooling load but add nothing to the electrical output.

The principal outputs of TPVMODEL are the total mass and the total Earth mass of the TPV converter, together with the solar concentrator and the cooling system. Parameters used to support these outputs are the specific masses of various components and the estimated non-lunar mass fraction of each.

TPVMODEL has provisions for using thermal energy stored in the radiator to provide power during the eclipsed portion of the SPS orbit. Parameter A13, set to FALSE in this example, controls whether thermal storage will be considered.

2.3.4 TPV References

1. Solar Power Satellite: System Definition Study, Vol. 3, Phase I, Final Report, D180-25037-3, Boeing Aerospace Company, Seattle, 1979.

2. Lunar Resources Utilization for Space Construction, Final Report, Ed Bock study manager, General Dynamics Convair Division, 1979.

3. Wang, Jing-xiao, "An Investigation of Copper Contact for Use in Silicon Solar Cells", 17th IEEE Photovoltaics Specialists Conf., 1984, pp. 580-583.

4. Horne, W. E., "Solar Thermal Photovoltaic Electric Power Generator", Proc. of Miami International Conf. on Alternate Energy Sources, 1977.

5. Horne, W. E., Day, A. C., Greegor, R. B., Milliman, L. D., and Crabtree, W. L., "Solar Thermophotovoltaic Space Power System", Proceedings of the 15th Intersociety Energy Conversion Engineering Conference, 1980.

6. Advanced Propulsion System Concepts for Orbital Transfer Study, Final Report, Vol. 2: Study Technical Results, D180-26680-2, Boeing Aerospace Company, 1981.

7. Mattick, A. T., and Hertzberg, A., "The Liquid Droplet Radiator - An Ultralight Heat Rejection System for Efficient Energy Conversion in Space", 23rd Congress of International Astronautical Federation, 1981.

8. "Multiplan" is a trademark of Microsoft Corp., Bellevue, Washington.

2.4 BRAYTON CYCLE

2.4.1 Introduction

The results reported here are based on a design as similar as possible to that of a NASA MSFC report.(1) Conservative estimates were made for potential substitution of lunar materials.

The majority of the weight for the system is in the solar collection and heat rejection systems. It is straightforward to determine where lunar materials could be substituted in these systems.

It is more difficult to estimate the percentage of lunar materials for the Brayton engine because the contract report(1) includes no details on the alloys appropriate for the engine components and the weights for specific components. The NASA report information was used to calculate engine efficiency for two operating temperatures and then conservative estimates were made for potential use of lunar materials at each temperature.

2.4.2 Design Description

A simple schematic of the Brayton cycle is given in Fig. 2.4-1. This single regenerative Brayton cycle uses a solar concentrator to collect the solar energy and reflect it into a cavity heat absorber. The heat is absorbed by the gas through the heat exchanger tubing lined on the heat absorber's insulated shell. This heat transfer also helps reduce the temperature of the heat absorber itself. The hot gas then goes through the turbine, which converts some of the thermal energy to mechanical work by turning the compressor and the generator. The recuperator extracts more heat from the gas coming out of the turbine and transfers it to the cold side of the cycle. The waste heat is rejected by the radiator.

Helium was chosen as the working fluid to make use of the mass statements of the cycle investigated by NASA. According to one reference (2), a 60% Xe / 40% He mixture with a molecular weight of 80 was also considered. This mixture requires fewer stages for the turbomachinery, but would require a much larger heat exchanger due to reduced thermal conductivity. The size of the turbomachinery would also have to be increased (2) which is already a problem, even with helium. Thus, pure helium was selected as the working fluid.

The selected radiator system is a liquid metal radiator (LMR), i.e., a heat pipe radiator which uses sodium-potassium (NaK) as the coolant and water as the working fluid in the heat pipes. It is possible that an LDR system using lunar NaK as the working fluid would be suitable at some temperatures, but such a system was not selected due to its high technical risk.

Two different sets of cycle points were studied to show the effect of temperature on the mass statement. The first set was picked to emphasize higher efficiency and lower total mass. It uses the same cycle state points as (1). This cycle has a very high turbine inlet temperature (1617 K) which limits the use of lunar materials for components such as the turbine blades or the heat absorber.

The second set of temperatures has a more moderate turbine inlet temperature of 1100 K, which is seen in existing engines. This allows better utilization of the lunar alloys (Ni alloys, steel, Cr alloys) and metals like Al and Ti.

The efficiency of the cycles was calculated assuming:

Compressor efficiency = 0.875
Turbine efficiency = 0.920
Bearing loss & turbine disk cooling efficiency factors = 0.98

The compression and turbine efficiencies used here are from (1) and are viewed as possible but non-conservative values.

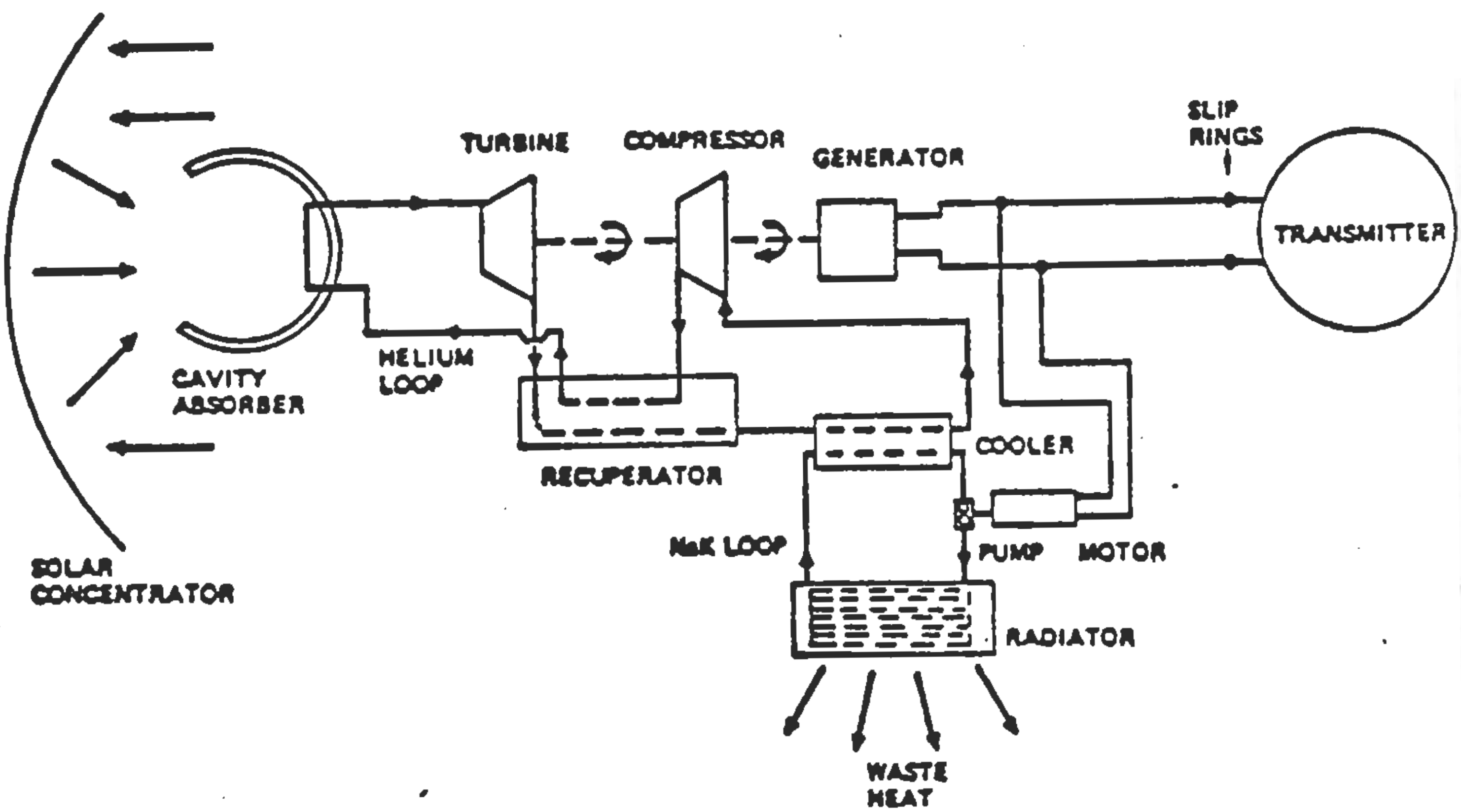

FIGURE 2.4-1, BRAYTON SOLAR CYCLE

Table 2.4-1 shows the state points and the efficiencies of each temperature set. The mass statement of each cycle is shown in Table 2.4-2.

TABLE 2.4-1 SELECTED CYCLE STATE POINTS FOR BRAYTON CYCLE

		High Temp.	Low Temp.	
TW	(Wall Temp)	1650	1122	K
TO	(Min. Gas Temp)	404	345	K
T1	(Comp. Outlet Temp)	603	515	K
T2	(Absorber Inlet Temp)	1108	754	K
T3	(Turbine Inlet Temp)	1617	1100	K
T4	(Turbine Outlet Temp)	1190	810	K
T5	(Cooler Inlet Temp)	685	580	K
TL1	(Minimum NaK Temp)	377	322	K
TL2	(Maximum NaK Temp)	644	550	K
ER	(Recuperator Effectiveness)	0.86	0.81	
Cycle efficiency		0.43	0.35	
Cycle pressure ratio		2.5	2.6	

TABLE 2.4-2 MASS ANALYSES FOR 9 GW BRAYTON SYSTEMS

High Temperature Cycle (Efficiency 43%):

COMPONENT	TOTAL MASS (kg)	NON-LUNAR MASS (kg)
Radiator system	1.64E+07	5.11E+04
Heat absorber	4.80E+06	2.38E+06
Recuperator/cooler	2.57E+06	6.46E+05
Concentrator	2.32E+06	0.0
Turbomachines	1.03E+06	7.73E+05
TOTAL:	2.71E+07	3.85E+06

Non-lunar specific mass: 0.43 kg/kW
Total specific mass: 3.01 kg/kW

Low Temperature Cycle (Efficiency 35%):

COMPONENT	TOTAL MASS (kg)	NON-LUNAR MASS (kg)
Radiator system	7.43E+07	1.32E+05
Heat absorber	5.88E+06	5.82E+05
Recuperator/cooler	1.29E+07	1.29E+06
Concentrator	2.76E+06	0.0
Turbomachines	1.03E+06	1.06E+05
TOTAL:	9.69E+07	2.11E+06

Non-lunar specific mass: 0.234 kg/kW
Total specific mass: 10.77 kg/kW

Table 2.4-2 was developed using the mass statement of the NASA report (1) as the primary reference. That report presented a solar Brayton cycle with 17 GW of output power to two transmitters. The efficiency of this cycle is 43.9%. Figure 2.4-2 shows the overall configuration of this SPS. The masses of the components of the NASA paper were corrected for material substitutions, design changes, and temperature ranges considered in this study. These adjustments were done with conservative assumptions. More accurate estimates of the masses of these components should be made if the Brayton cycle is considered further as an option for the SPS energy conversion system.

As shown in Table 2.4-2, the lower temperature cycle gives the lower specific non-lunar mass. However, the decrease in non-lunar mass is offset by a forty times greater increase in total mass. Since the cost ratio of non-lunar to lunar materials is estimated to be no better than fifty, there are no grounds for preferring one temperature cycle over another.

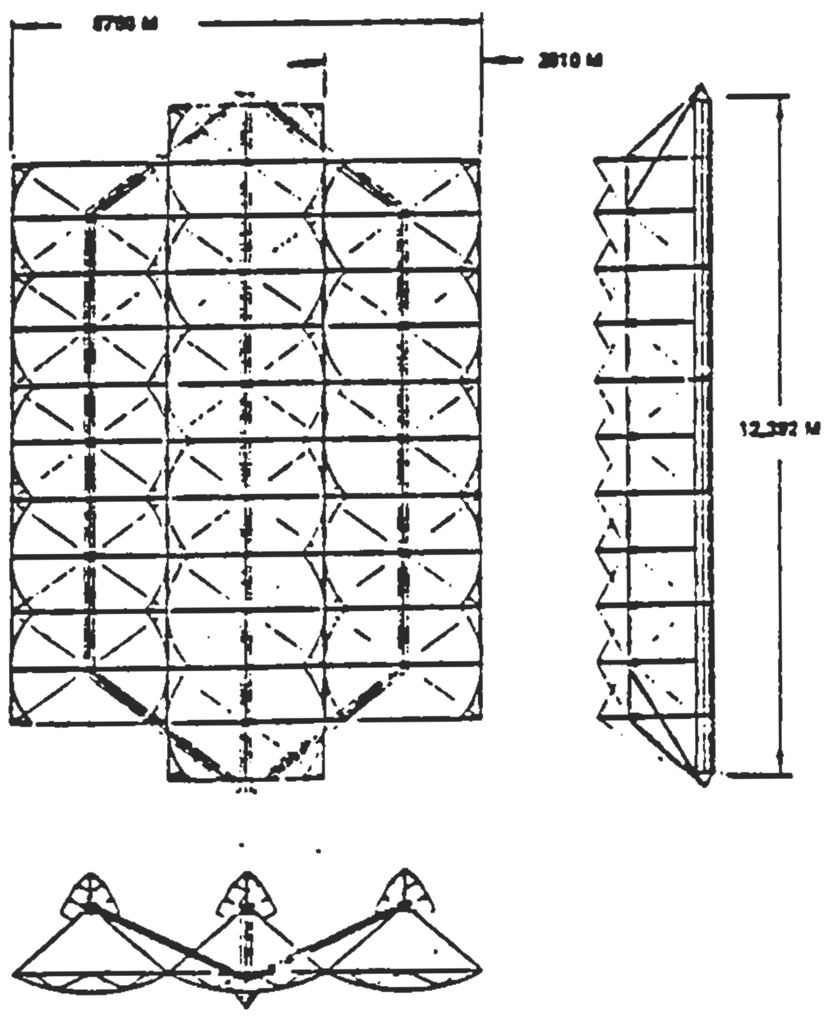

FIGURE 2.4-2, BRAYTON SPS CONFIGURATION

2.4.3 Brayton Cycle Technical Discussion

2.4.3.1 Radiator

In the high temperature Brayton cycle, 12.2 GW of waste heat must be rejected into space to produce 9 GW of electricity. The inlet and outlet temperatures are 644 K and 377 K respectively. The low temperature cycle must reject 16.9 GW of heat, with high and low temperatures of 550 K and 322 K.

One of the 16 modules of the 10 GW Brayton SPS configuration is shown in Figure 2.4-3. The cavity absorber with the engines is in the center. The concentrator is below with its concentrated sunlight entering the aperture. Radiator panels surround the absorber.

The basic element of the radiator panels is the heat pipe. For the range of temperatures used here, a water heat pipe is a reasonable choice. This heat pipe can be made with thin stainless steel and an internal stainless steel wick.

Figures 2.4-4 and 2.4-5 show more details of the HPR configuration and its panels. Stainless steel is the primary material used in this radiator system. Water is used as the working fluid, which is estimated to have about 0.2% of the total panel mass. Hydrogen in the water and carbon in the steel (0.3%) are assumed to be the only non-lunar materials.

The mass breakdown for this radiator for the 9 GW high temperature Brayton cycle is shown in Table 2.4-3.

TABLE 2.4-3 RADIATOR MASS FOR 9 GW BRAYTON CYCLE

Total Radiator Mass = 16.4E+06 kg
Mass of panels with water = 1.02E+07 kg
Mass of water = .002 x 1.02E+07 = 2.04E+04 kg
Non-lunar Mass =(.1 x .020 + (16.4 -.020)x.003)x1E+06 = 5.11E+04 kg

2.4.3.2 Concentrator

An all-lunar aluminum concentrator like that described in section 2.7 is assumed. The mass of the concentrator for both temperature cycles is shown in Table 2.4-4.

TABLE 2.4-4 CONCENTRATOR MASS FOR 9 GW BRAYTON CYCLE

High Temperature Brayton Cycle:
Total Concentrator Mass = 2.32E+06 kg
Non-Lunar Mass = 0

Low Temperature Brayton Cycle:
Total Concentrator Mass = 2.76E+06 kg
Non-Lunar Mass = 0

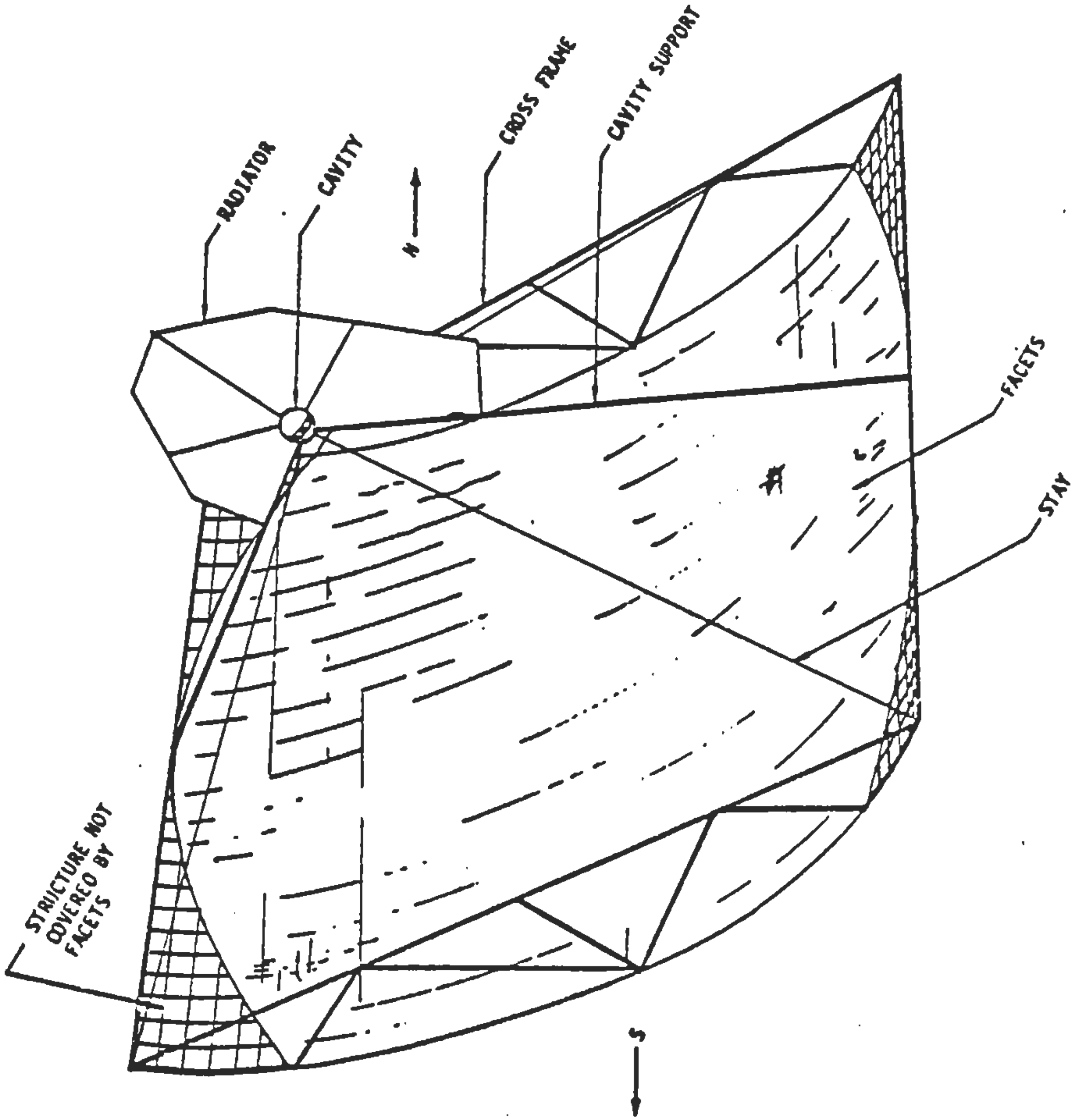

FIGURE 2.4-3, TROUGH TYPE MODULE

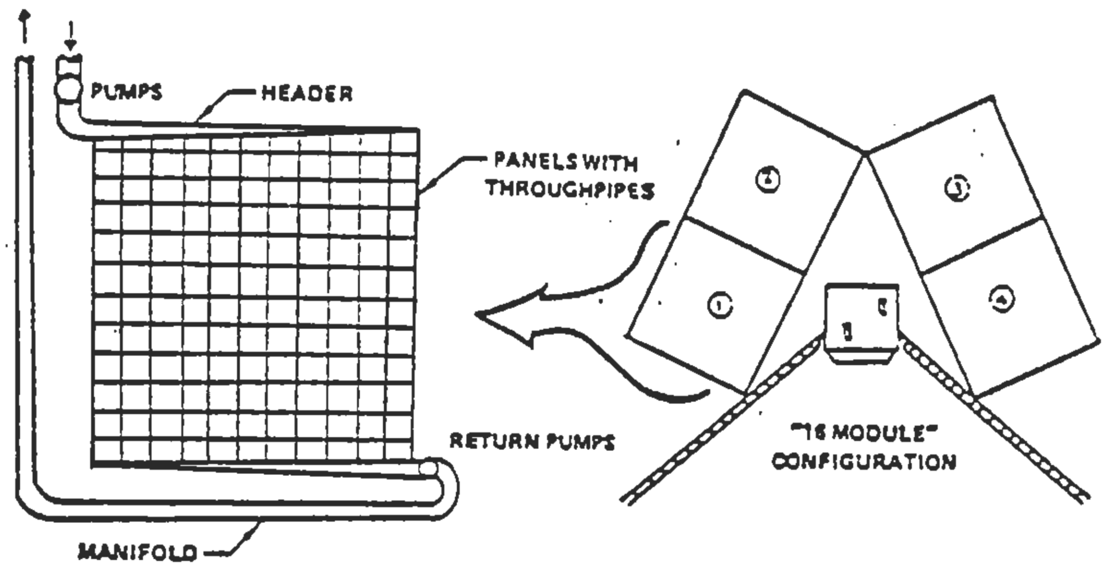

FIGURE 2.4-4, RADIATOR CONFIGURATION

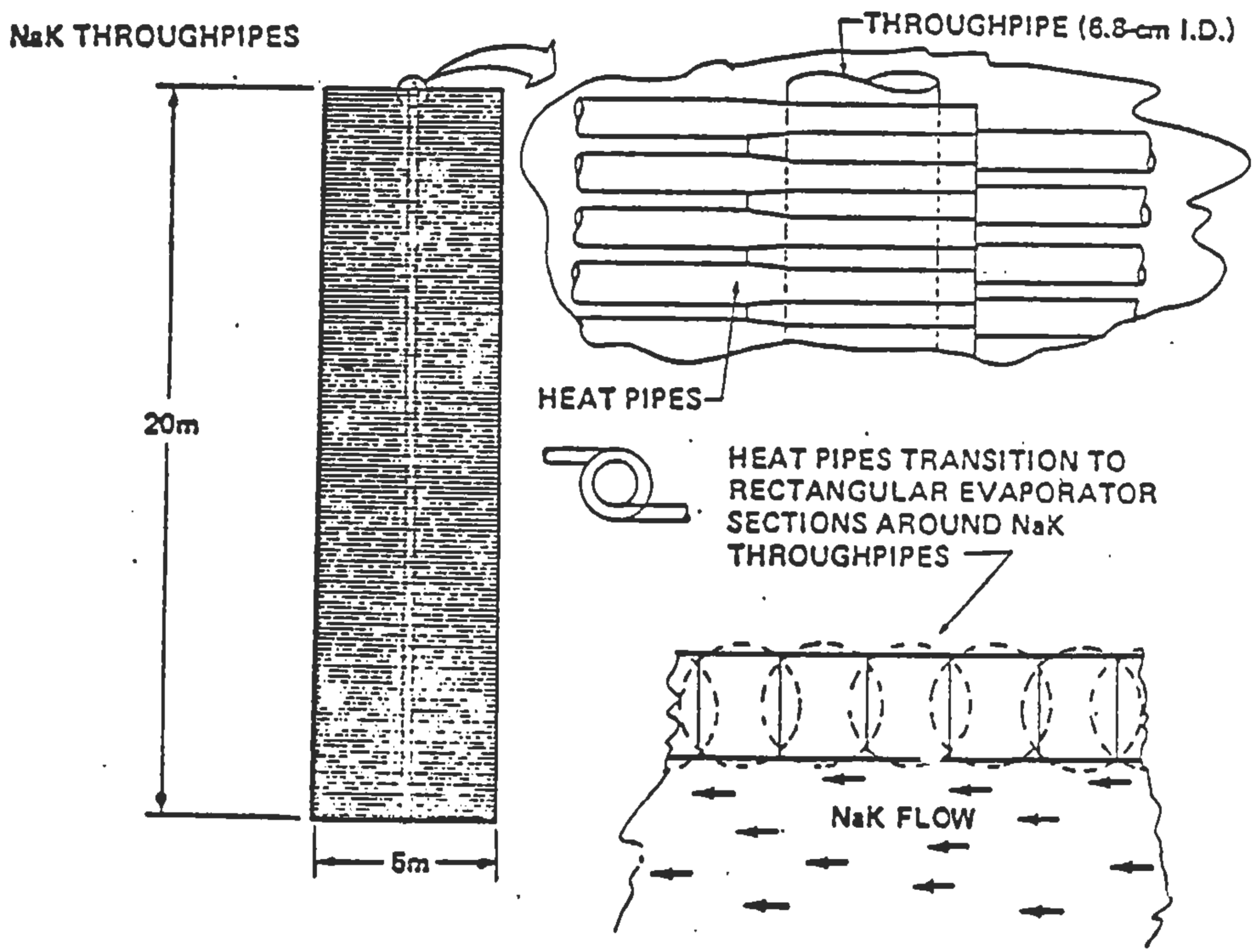

FIGURE 2.4-5, HEAT PIPE PANEL DETAILS

2.4.3.3 Heat Absorber

The system uses a cavity heat absorber which collects heat from the concentrator and transfers it to the helium flowing through the pipes on its panels. A typical absorber panel is shown in figure 2.4-6.

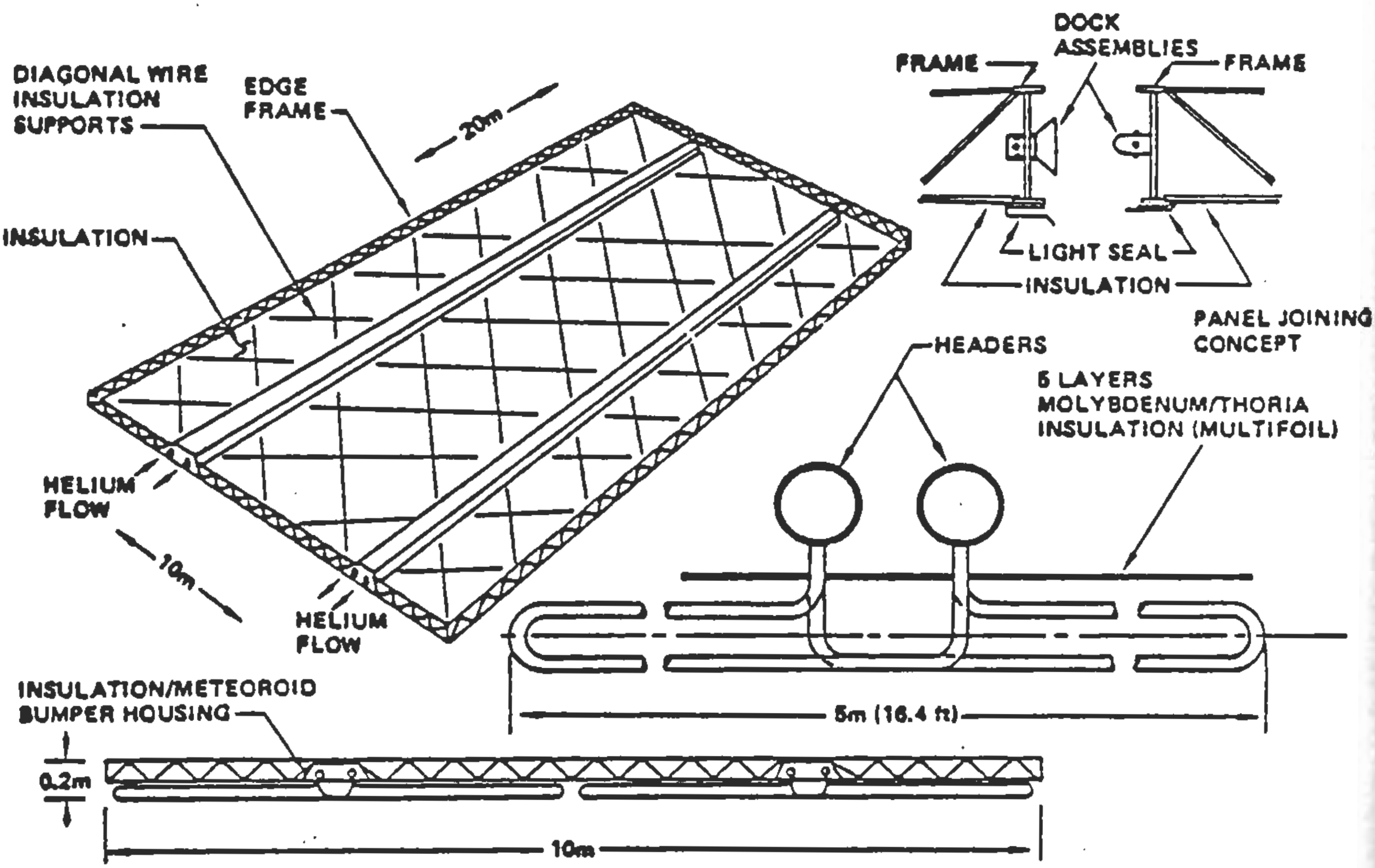

FIGURE 2.4-6, CAVITY HEAT ABSORBER PANEL

The heat absorber tubing loops are mounted on the cavity interior side. The skin of the panel is multi-layer high temperature insulation, stabilized by a crisscross wire pattern. Figure 2.4-7 shows one cavity which provides 1.1 GW to the SPS bus. The dashed outline in Figure 2.4-8 represents the cavity absorber. The manifolds which conduct the hot and cold helium through the engine are shown here. By using insulated ducts, a cooler and thereby stronger outer shell is achieved.

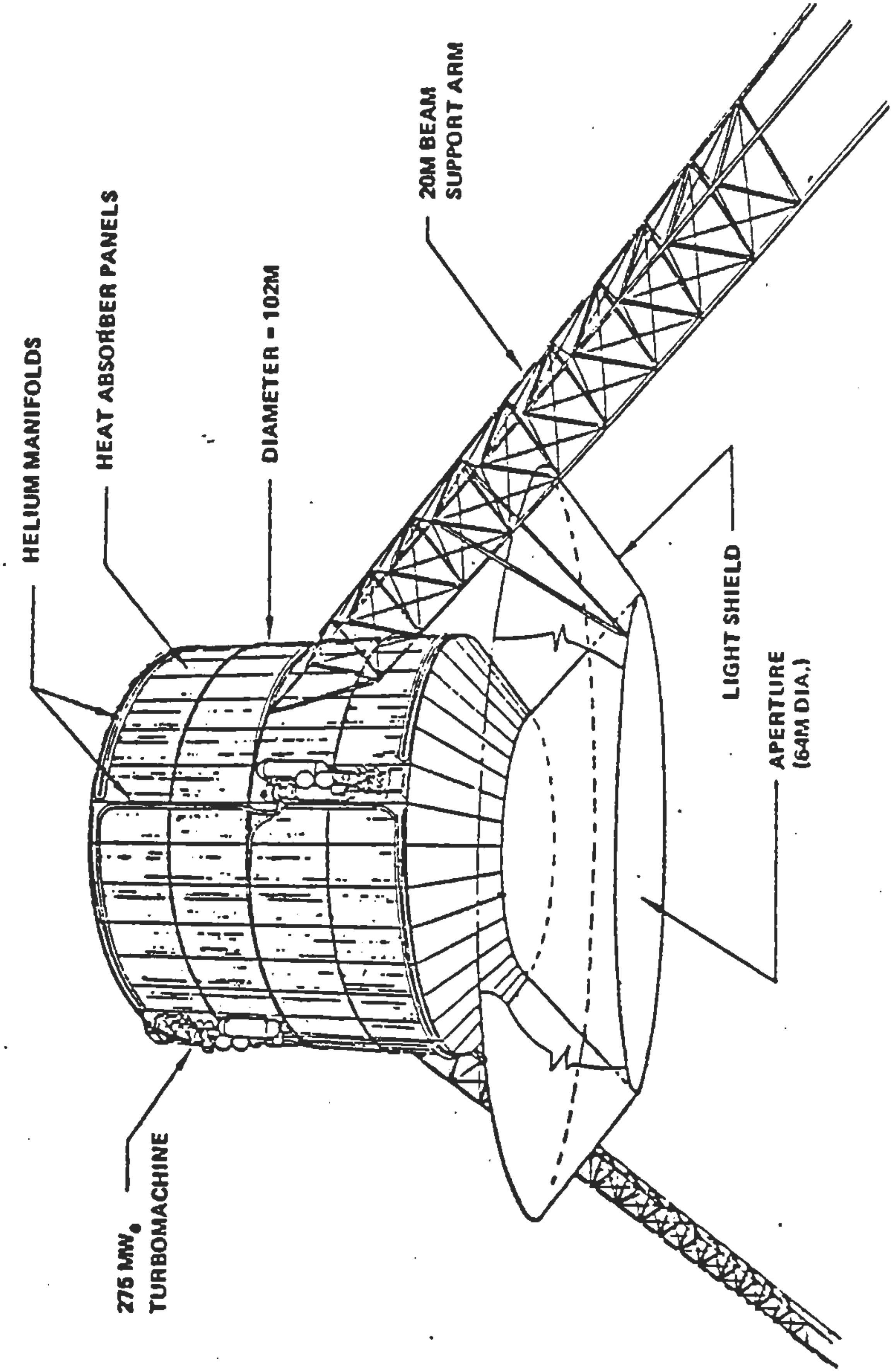

FIGURE 2.4-7, FOCAL POINT ASSEMBLY SHOWING HEAT ABSORBER PANELS

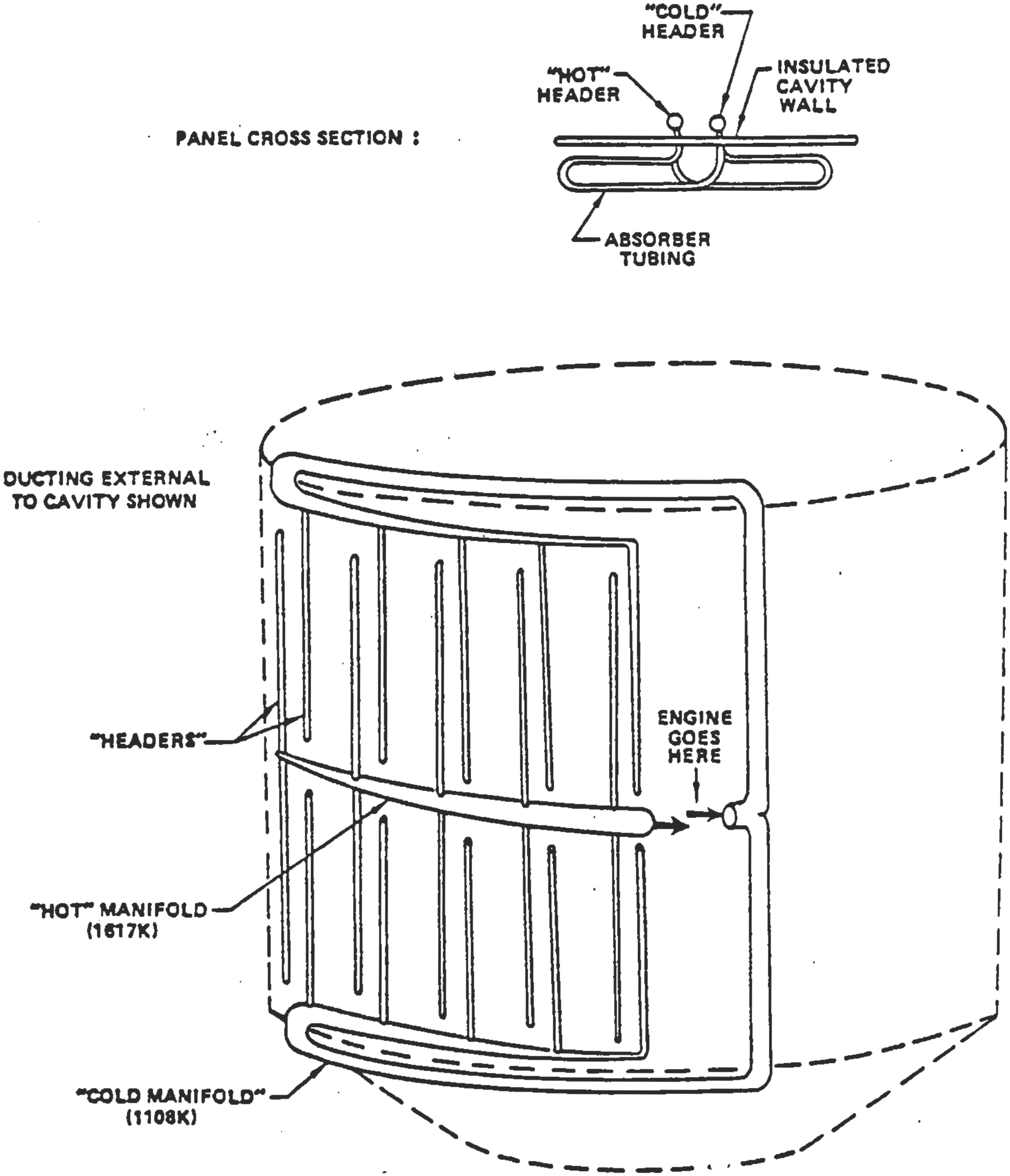

FIGURE 2.4-8, PRIMARY HEAT EXCHANGER LAYOUT (SIMPLIFIED)

The high temperatures created on the absorber panels preclude use of aluminum or even stainless steel. There are some high temperature alloys which can be made of lunar materials and which can be used for construction of the panel skins and even the pipes. Croloy (2.25% Cr - 1% Mo) can be used up to about 830 K and other heat resistant steel alloys (25% Cr - 12% Ni or 25% Cr - 20% Ni) have been exposed up to 1810 K. Nickel alloys such as Rene 41 (10% Co, 10% Mo, .005% B, plus Ni, Al, Ti, and Fe) are mostly made of lunar materials and are used in high temperature devices such as aircraft engines and re-entry vehicles.

It was assumed that 50% of the high temperature absorber and 10% of the low temperature absorber would require non-lunar materials. The absorber mass estimates are shown in Table 2.4-5.

TABLE 2.4-5 HEAT ABSORBER MASS FOR 9 GW BRAYTON CYCLE

High Temperature Brayton Cycle:
Total Absorber Mass = 4.80E+06 kg
Non-Lunar Mass = 2.38E+06 kg

Low Temperature Brayton Cycle:
Total Absorber Mass = 5.88E+06 kg
Non-Lunar Mass = 5.82E+05 kg

2.4.3.4 Turbomachinery

The high temperature of helium going through the turbine is tough to accommodate. This Brayton cycle also requires large diameter tube discs. Experiments with high temperature super alloys such as Astroloy or INCO 608 and refractory alloys such as molybdenum TZM have been conducted to overcome such problems. Refractory materials such as silicon carbide have also been suggested for making turbine blades facing high temperatures. The compressor, however, faces lower temperatures and doesn't require high temperature materials like the turbine. A good portion of the pipes also do not face the highest temperature of the cycle since the working fluid reaches the peak temperature when entering the turbine.

Overall a good portion of the turbomachinery can be made of lunar materials with good strength and resistance to higher temperatures (such as Croloy, stainless steel, or Ni alloys). It is estimated that 75% of the high temperature turbomachinery and 10% of the low temperature turbomachinery will require non-lunar materials. The turbomachinery mass estimate for each cycle is shown in Table 2.4-6.

TABLE 2.4-6 TURBOMACHINERY MASS FOR 9 GW BRAYTON CYCLE

High Temperature Brayton Cycle:
Total Turbomachinery Mass = 1.03E+06 kg
Non-Lunar Mass = 0.773E+06 kg

Low Temperature Brayton Cycle:
Total Turbomachinery Mass = 1.03E+06 kg
Non-Lunar Mass = 1.06E+05 kg

2.4.3.5 Recuperator/Cooler

The recuperator faces relatively high temperatures from the working fluid coming out of the turbine. It helps save some of the energy of the hot gas by transferring the heat to the cold gas coming out of the compressor. Therefore it needs to be made of materials with good thermal conductivity and resistant to temperatures of up to 1200 K (for the high temperature cycle). Some parts of this device, such as the outer shell and some of the pipes, can be made of lunar alloys like croloy, stainless steel, or other steels.

The high temperature cycle cooler operates at moderate temperatures. It uses NaK as the working fluid (which is assumed to be lunar) and it can be made mostly out of lunar metals such as aluminum or nickel. The low temperature cycle cooler operates at lower temperatures and is more massive. It has a smaller percentage of non-lunar materials but a higher non-lunar mass.

It is estimated that 25% of the high temperature recuperator/cooler system and 10% of the low temperature system will require non-lunar materials. The mass estimate for each cycle is shown in Table 2.4-7.

TABLE 2.4-7 RECUPERATOR/COOLER MASS FOR 9 GW BRAYTON CYCLE

High Temperature Brayton Cycle:
Total Recuperator/Cooler Mass = 2.57E+06 kg
Non-Lunar Mass = 0.646E+06 kg

Low Temperature Brayton Cycle:
Total Recuperator/Cooler Mass = 1.29E+07 kg
Non-Lunar Mass = 1.29E+06 kg

2.4.3.6 Primary Structure

The mass of the primary structure is not included in specific mass estimates for power conversion systems. This information on the Brayton SPS primary structure mass (also used in the TPV, Rankine, and Stirling) is provided for comparison with the silicon planar structure discussed in chapter 3.

The primary structure holds the modules together. It receives direct sunlight. Aluminum can be used, though passive thermal control measures like those discussed in the thermal control section of chapter 3 may be needed. The estimated mass is shown in Table 2.4-8.

TABLE 2.4-8 PRIMARY STRUCTURE MASS FOR 9 GW BRAYTON CYCLE

High Temperature Brayton Cycle:
Total Structural Mass = 3.34E+06 kg
Non-Lunar Mass = 0

Low Temperature Brayton Cycle:
Total Mass = 4.05E+06 kg
Non-Lunar Mass = 0

2.4.4 Brayton Cycle References

1. "Space Based Power Conversion and Relay Systems: Preliminary Analysis of Alternative Systems", 26 May, 1976, NASA MSFC Contract NAS8-31628, Boeing Aerospace Co., Pub., NASA-CR-150171.

2. "Brayton Cycle Technology", W.L. Stewart, W.J. Anderson, D.T. Bernatowicz, D.C. Guentert, D.R. Packe, H.E. Rohlik, Space Power Systems Advanced Technology Conf., NASA Lewis Research Center, Cleveland, OH, 1966. p. 95-145. (NASA-SP-131).

2.5 RANKINE CYCLE

2.5.1 Introduction

A simple Rankine cycle is shown in Figure 2.5-1. In this cycle, the working fluid is pumped into the heater. The fluid then undergoes constant-pressure heating in the heater. The turbine extracts the energy from the fluid and the waste heat is rejected by the radiator. The pump is run by a generator which is operated by the turbine.

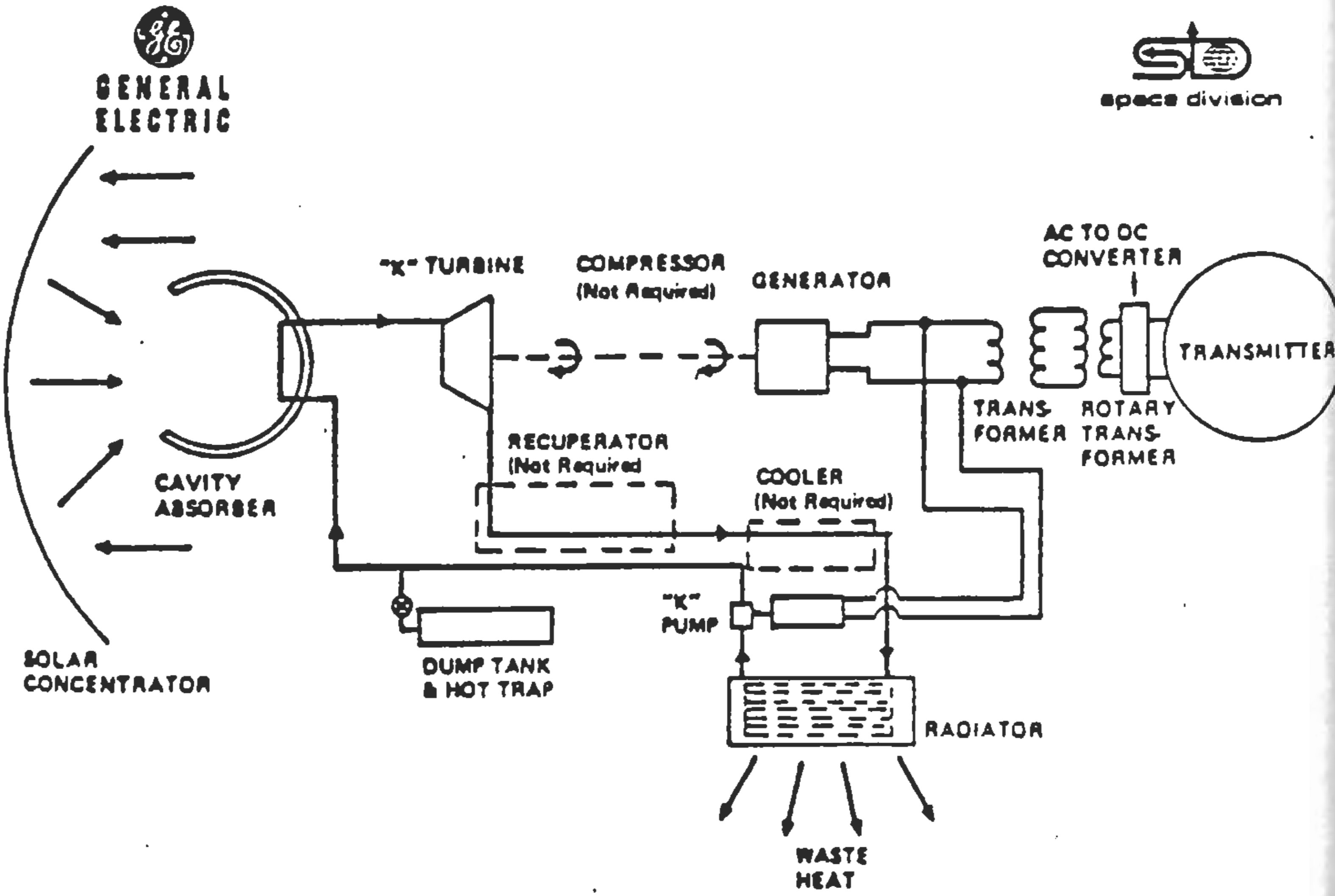

FIGURE 2.5-1, SOLAR RANKINE CYCLE

Rankine cycle is the ideal cycle for a simple steam power plant but use of such a cycle requires massive components for higher efficiencies. Pot:ssium was also considered as a working fluid for the Rankine cycle. Potassium Rankine cycle has lower efficiencies but higher power per unit mass than the steam cycle.

The efficiency of the Rankine cycle depends on the average temperature at which heat is supplied and the average temperature at which heat is rejected. This efficiency can be improved using options such as lowering the exhaust pressure, superheating, or using higher maximum pressure. However, these methods require compromises in turbine efficiency and may significantly increase blade erosion. Concepts such as reheat cycle and regeneration cycle offer efficiency improvements without the compromises on the turbines. These concepts, however, may require more massive cycles. Overall design of a suitable Rankine cycle depends on the specific requirements of the power system concerning the efficiency, mass, and availability of the materials.

The designs presented in this section are based on a NASA MSFC report.(1) This document provided the mass statements of Rankine cycles designed for an output power of 17 GW. The primary requirement of the NASA study was the efficiency of the cycle, with low total mass as the other important requirement. The data provided in the NASA report was used to estimate the component masses of the Rankine cycles presented there. Substitutions for the non-lunar materials were made where ever possible.

2.5.2 Design Description

The Rankine cycle studied by NASA uses potassium as the working fluid. Solar energy is concentrated on the cavity heat absorber, where the working fluid is heated. The radiator is a heat pipe radiator which utilizes the working fluid of the cycle, eliminating the need for a cooler. The generator converts the mechanical energy of the turbine to electrical output. The pump is run by power from the generator. The use of a recuperator in this cycle is not possible since the necessary temperature potential between the compressor output and the turbine output does not exist in this cycle.

Four Rankine cycles were investigated in this study. Two cycles use potassium, one with a turbine inlet temperature of 1600 K and the other with 1422 K. The other two cycles use steam as working fluid with turbine inlet temperatures of 1644 K and 811 K. The thermodynamic variables of these cycles were selected to give the highest cycle efficiency without unrealistic sacrifices in masses of the components. The cycle characteristics of these cycles are shown in Table 2.5-1.

TABLE 2.5-1: RANKINE CYCLE CHARACTERISTICS

Cycle	Turb. Inlet Temp. (K)	Turb. Inlet Press. (psi)	Turb. Press. Ratio	Comp. Inlet Temp. (K)	Comp. Inlet Press. (psi)	Cycle Efficiency %
Pot. Rankine	1600	410	171	867	2.4	31
Pot. Rankine	1422	201	84	867	2.4	27
Steam Rankine	1644	250	22	367	11.5	42
Steam Rankine	811	3000	261	367	11.5	34

Potassium Rankine cycles show lower efficiencies than the steam cycles. Higher efficiencies for potassium cycles would have required untractable dimensions for the low pressure stages of the turbines. Steam Rankine cycles show better efficiencies for the corresponding ranges of turbine inlet temperature, but these cycles are much more massive. The high temperature steam cycle has the lowest non-lunar mass, but the non-lunar masses of the cycles do not differ greatly. (Table 2.5-2).

2.5.3 Design Summary

The specific mass statements of the four cycles are presented in Table 2.5-2.

TABLE 2.5-2: SPECIFIC MASS STATEMENTS FOR RANKINE CYCLES

Potassium Rankine, High Temperature Cycle:

Component	Total Specific Mass (Kg/KW)	Non-Lunar Mass (Kg/KW)
Radiator System	0.354	0.002
Heat Absorber	0.59	0.29
Concentrator	0.36	0.
Turbomachinery	1.44	1.18
Total	2.74	1.47

Potassium Rankine, Low Temperature Cycle:

Component	Total Specific Mass (Kg/KW)	Non-Lunar Mass (Kg/KW)
Radiator System	0.50	0.003
Heat Absorber	0.12	0.06
Concentrator	0.41	0.
Turbo.	2.08	1.70
Total	3.11	1.76

Steam Rankine, High Temperature Cycle:

Component	Total Specific Mass (Kg/KW)	Non-Lunar Mass (Kg/KW)
Radiator System	4.93	0.03
Heat Absorber	0.18	0.09
Recuperator	0.11	0.03
Concentrator	0.26	0.
Turbo.	1.32	1.08
Total	6.80	1.23

TABLE 2.5-2 (continued):

Steam Rankine, Low Temperature Cycle:

Component	Total Specific Mass (Kg/KW)	Non-Lunar Mass (Kg/KW)
Radiator System	9.06	0.05
Heat Absorber	1.12	0.56
Concentrator	0.33	0.
Turbo.	1.59	1.30
Total	12.10	1.91

The high temperature steam cycle requires the least non-lunar mass (1.23 kg/kW) and has the lowest cost, assuming a 50:1 cost ratio favoring lunar materials. This cycle also has the highest efficiency among the Rankine cycles. A mass analysis for this cycle is shown in Table 2.5-3.

TABLE 2.5-3 MASS ANALYSIS FOR 9 GW HIGH TEMPERATURE STEAM RANKINE CYCLE

COMPONENT	TOTAL MASS (kg)	NON-LUNAR MASS (kg)
Radiator System	4.44E+07	2.70E+05
Heat Absorber	1.62E+06	8.10E+05
Recuperator	9.90E+05	2.5 E+05
Concentrator	2.38E+06	0.0
Turbomachinery	1.19E+07	9.72E+06
Total	6.13E+07	1.11E+07

In general the potassium cycles proved to be less efficient than steam cycles. The efficiency of the potassium cycle was sacrificed to avoid the low pressure stages of the turbine from growing to unrealistic dimensions, which would have also increased the amount of non-lunar mass required. The higher heat rejection temperatures of these cycles reduce the mass of the radiators greatly. But the large mass of the turbomachinery offsets the mass reduction of the radiators. Most of the non-lunar materials used in the potassium cycles are for construction of the turbomachinery. Of the potassium cycles, the high temperature cycle shows better efficiency, lower total mass and lower non-lunar mass.

The steam cycles are much more massive than the potassium cycles in spite of their higher efficiency for both high and low temperature ranges. The large mass of these cycles is mainly due to low heat rejection temperatures, which require large radiators. However, the relatively small turbomachinery of the steam cycles results in less non-lunar mass.

The potassium Rankine cycles and the low temperature steam cycle were designed with the turbine expansion process terminating in the two-phase region. Therefore, there is no temperature potential for regeneration in these cycles. A recuperator was used in the high temperature steam cycle. Coolers are not needed in any of the cycles because the cycle working fluid can be utilized as the radiator fluid.

2.5.4 Rankine Cycle Technical Discussion

The Rankine cycle SPS design (1) has the same general configuration as the Brayton cycle SPS. The input power necessary for the Rankine cycle engines depends on the efficiency of each cycle. The mass of each major component of the Rankine cycle after substitution of lunar materials is given below. The suggested lunar materials for use in these components are also listed below. These materials were chosen based on the available information about the temperature, stress, and conduction requirements.

2.5.4.1 Radiator

The radiators for the steam cycles are much more massive than those of the potassium cycles. This is due to the low rejection temperatures selected for the steam cycles to achieve good efficiencies.

The heat pipe radiator for the Rankine cycles uses the working fluid of the cycle as coolant, eliminating the need for a heat exchanger in the cycle.

The radiating sheets were assumed to be aluminum. Stainless steel was used for the heat pipe shells and the wicks. Water was used as the heat pipe working fluid.

Potassium Rankine High Temperature Cycle:
Total Radiator Mass=3.19x10E6 Kg
Non-Lunar Mass=0.0196x10E6 Kg

Potassium Rankine Low Temperature Cycle:
Total Radiator Mass=4.50x10E6 Kg
Non-Lunar Mass=0.0265x10E6 Kg

Steam Rankine High Temperature Cycle:
Total Radiator Mass=44.4x10E6 Kg
Non-Lunar Mass=0.270x10E6 Kg

Steam Rankine Low Temperature Cycle:
Total Radiator Mass=81.5x10E6 Kg
Non-Lunar Mass=0.450x10E6 Kg

2.5.4.2 Concentrator

This system is identical to that for the Brayton cycle.

Potassium Rankine High Temperature Cycle:
Total Mass=3.22x10E6 Kg
Non-Lunar Mass=0.

Potassium Rankine Low Temperature Cycle:
Total Mass=3.70x10E6 Kg
Non-Lunar Mass=0.

Steam Rankine High Temperature Cycle:
Total Mass=2.38x10E6 Kg
Non-Lunar Mass=0.

Steam Rankine Low Temperature Cycle:
Total Mass=2.93x10E6 Kg
Non-Lunar Mass=0.

2.5.4.3 Heat Absorber

It was assumed that 50% of this component would require non-lunar materials. Some possible materials for high temperature parts of this component are Croloy(2.25% Cr- 1% Mo), steel alloys such as (25% Cr - 12% Ni) or (25% Cr - 20% Ni), or nickel alloys such as Rene 41 (10% Co, 10% Mo, .005% B plus Ni, Al, Ti, Fe).

Potassium Rankine High Temperature Cycle:
Total Mass= 5.31x10E6 Kg
Non-Lunar Mass=2.61x10E6 Kg

Potassium Rankine Low Temperature Cycle:
Total Mass=1.08x10E6 Kg
Non-Lunar Mass=0.54x10E6 Kg

Steam Rankine High Temperature Cycle:
Total Mass=1.62x10E6 Kg
Non-Lunar Mass=0.81x10E6 Kg

Steam Rankine Low Temperature Cycle:
Total Mass=10.08x10E6 Kg
Non-Lunar Mass=5.04x10E6 Kg

2.5.4.4 Turbomachinery

It was assumed that 18% of this component could be made with lunar materials. Materials containing little non-lunar mass which might be suitable for colder and less stressed components are Croloy, stainless steel, and nickel alloys.

Potassium Rankine High Temperature Cycle:
Total Mass=12.96x10E6 Kg
Non-Lunar Mass=10.62x10E6 Kg

Potassium Rankine Low Temperature Cycle:
Total Mass=18.72x10E6 Kg
Non-Lunar Mass=15.3x10E6 Kg

Steam Rankine High Temperature Cycle:
Total Mass=11.88x10E6 Kg
Non-Lunar Mass=9.72x10E6 Kg

Steam Rankine Low Temperature Cycle:
Total Mass=14.31x10E6 Kg
Non-Lunar Mass=11.7x10E6 Kg

2.5.4.5 Recuperator/Cooler

As mentioned earlier, none of the cycles need coolers. The use of recuperators for all of the Rankine cycles except the high temperature steam cycle is not possible since the necessary temperature potential does not exist in these cycles. It was assumed that 25% of the high temperature steam recuperator is non-lunar material.

Steam Rankine High Temperature Cycle:
Total Mass=0.99x10E6 Kg
Non-Lunar Mass=0.25x10E6 Kg

2.5.4.6 Primary Structure

This structure is identical to that for the Brayton SPS. Aluminum was assumed for this component.

Potassium Rankine High Temperature Cycle:
Total Mass=4.5x10E6 Kg
Non-Lunar Mass=0

Potassium Rankine Low Temperature Cycle:
Total Mass=5.13x10E6 Kg
Non-Lunar Mass=0

Steam Rankine High Temperature Cycle:
Total Mass=3.33x10E6 Kg
Non-Lunar Mass=0

Steam Rankine Low Temperature Cycle:
Total Mass=4.14x10E6 Kg
Non-Lunar Mass=0

2.5.4 Rankine Cycle References

1. "Space Based Power Conversion and Relay Systems: Preliminary Analysis of Alternative Systems", 26 May, 1976, NASA MSFC Contract NAS8-31682, Boeing Aerospace Co., NASA-CR-150171.

2.6 STIRLING ENGINE

2.6.1 Introduction

A Stirling engine consists of two cylinders connected by a gas passage. One cylinder contains a displacer; the other contains a power piston. This is shown in Figure 2.6-1.

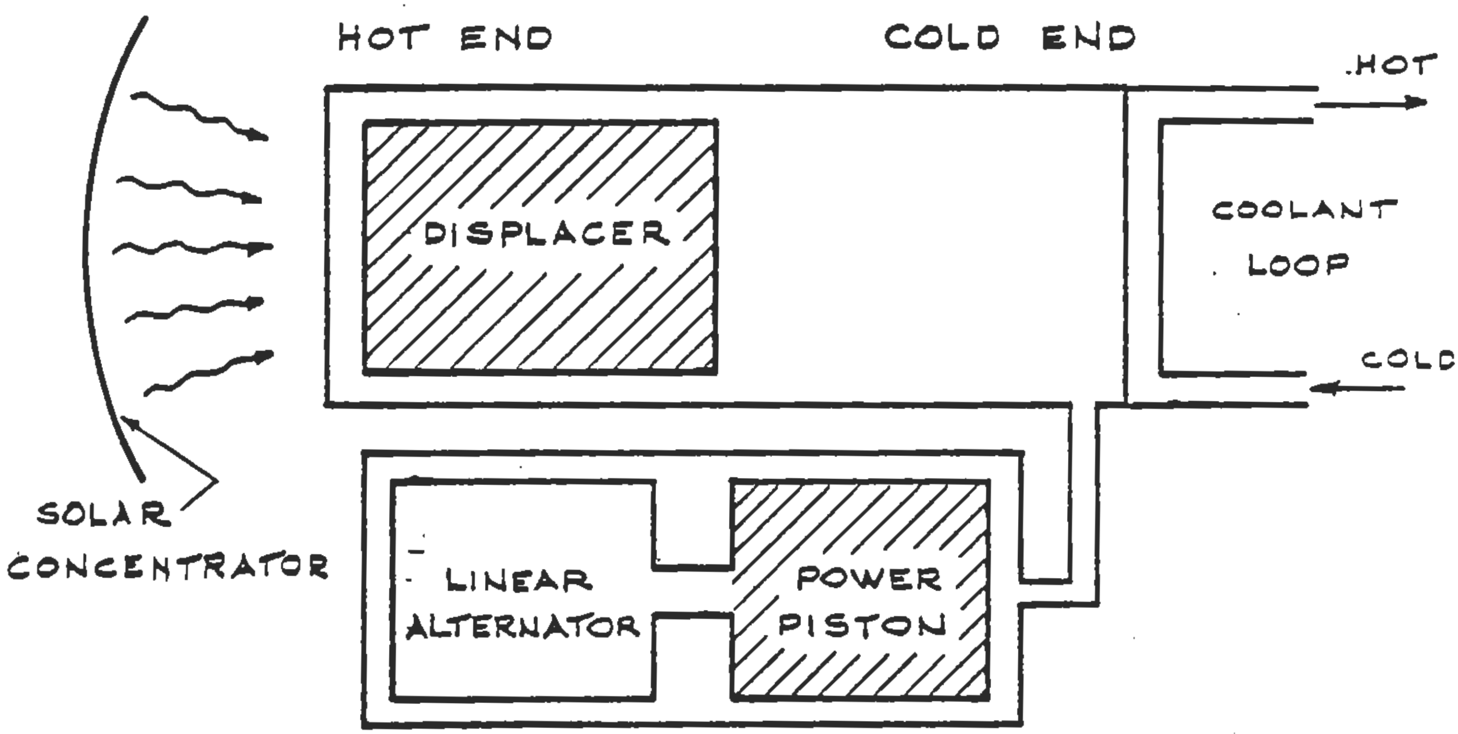

FIGURE 2.6-1, SOLAR STIRLING ENGINE DIAGRAM

The cycle starts with the displacer in the hot end of the cylinder. Most of the gas is in the cold end and in the power cylinder. The pressure is low, the gas is cold, and the power piston is near the top of its cylinder. The displacer moves to the cold end of the heated cylinder. The displaced gas flows around the displacer, or through a recuperator, to the hot end of the cylinder. The temperature of the gas increases, increasing the pressure. The increased pressure pushes the power piston downward in its cylinder. Connecting rods (or gas springs, in the case of the Free Piston Stirling) then return the displacer and the piston to the starting positions.

The Stirling engine has yet to make a significant showing in the marketplace, having failed to compete economically with either the piston engine or the standard thermal cycles. However, the space environment and the desire to avoid non-lunar materials may make the Stirling engine competitive for this application. The Stirling engine has the potential to achieve efficiencies competitive with other thermal engines at lower temperatures. This is a significant advantage as high temperature alloys can not be easily formed with lunar materials.

The Free Piston Stirling Engine (FPSE) is currently being evaluated for space applications. The Space Power Demonstration Engine by Mechanical Technology Inc. (MTI) is slated to test the major technology issues in mid-1985. The demonstration engine contract was awarded to MTI by NASA Lewis for design, fabrication, and testing of an FPSE. The FPSE is still in the early stage of development but may prove to be reliable, efficient, and suitable for manufacture with little non-lunar material. The FPSE has only two moving parts (displacer and power piston), hydrostatic gas bearings and noncontacting seals to avoid lubrication and wear of moving parts, and the potential for a hermetically sealed power module.

The planned final maximum operating temperature will be between 950 K and 1300 K. The demonstration engine goals (3,4) are:

T(hot) = 630 K
T(cold) = 315 K
25 kw output
25% efficiency (heat to electricity)
8 kg/kw
less than 0.008 cm vibration amplitude along any axis.

Because the Stirling can operate efficiently at low temperatures, it has promise as a power source on deep space probes which must generate power fron waste heat.

2.6.2 Design Description

The Stirling SPS design configuration is similar to the modular design of the Brayton or Rankine SPS. In each module, a large concentrator provides thermal energy for thousands of double cylinder engines, which reject waste heat into a single large radiator. Integrated into each engine is a linear alternator; thus no motional feedthroughs or rotary joints are required.

Each engine/generator produces 25 kW of electrical power at 25% efficiency. The hot side temperature of the engines is assumed to be 720 K, and the cold side temperature is 360 K. Helium is used as the working fluid. The engines are mostly made of steel, with part of the engine using a high temperature nickel steel.

It is assumed that absorber/concentrator modules like those of the Brayton design are used. In the Brayton and Rankine designs, each module produces up to 1.1 GW. For the Stirling design, power per module is reduced to 1.0 GW, so 40,000 Stirling engines are required in each module.

2.6.3 Design Summary

The mass statement for a Stirling system design which produces 9 GW at the SPS bus is shown in Table 2.6-1.

TABLE 2.6-1 MASS ANALYSIS OF 9 GW STIRLING ENGINE SYSTEM

	TOTAL (kg)	COMPOSITION	NON-LUNAR (kg)
Engines	7.2 E+7	High-temp Steel	3.3 E+6
Concentrators	4.0 E+6	Aluminum	0
Radiators	2.16E+8	Aluminum,Steel,Water	1.3 E+6
Absorbers	1.37E+7	High-temp Steel	1.37E+6
TOTAL	3.06E+8		5.97E+6

Non-lunar specific mass: 0.66 kg/kW
Total specific mass: 34.0 kg/kW

2.6.4 Stirling Technical Discussion

The theory and the technology of Stirling engines is advancing rapidly. Thus, predicting future performance is speculative at best.

The following is based closely on the performance objectives of the intermediate demonstration engine. The design life of this system is of the order of 100,000 hours, and the SPS goal is a lifetime of 262,000 hours (30 years); by using conservative stress levels, such as in the ASME boiler code, this may be possible. It is assumed that a large number of 25% efficient, 25 kW, double cylinder generators comprise the power generation component of an FPSE SPS.(3,5) The hot side temperature of 720 K is a conservative value.

It is difficult to scale a single Stirling engine up to high power levels, as is done with other thermal cycles. The walls of the displacer cylinder transport heat between the working fluid and the outside. As the engine gets larger, the gas volume increases more rapidly than the wall area, so heating of the gas is slowed. The need for thicker walls with larger size compounds this effect. External heat exchangers improve the rate of thermal transport but add mass and increase the dead gas volume, which decreases power.

The primary materials used in the engine are iron and Inconel 718.(6) Of the materials in the engine, only the working gas (helium) and some alloying elements in the Inconel 718 are non-lunar.

The composition of Inconel 718 (6) is as follows:

Lunar	
Nickel	50 - 55%
Chromium	17 - 21%
Iron	balance
Manganese	0.35% max
Silicon	0.35% max
Aluminum	0.2 - 0.8%
Titanium	0.65 - 1.15%
Cobalt	1.00% max
Non-Lunar	
Carbon	0.08% max
Sulfur	0.015% max
Copper	0.30% max
Columbium + Tantalum	4.75 - 5.50%
Molybdenum	2.8 -3.30%
Phosphorus	0.015% max
Boron	0.006% max

Those in the latter group total about 9.1%. Assuming no more than 50% of the mass of the engines needs to be high strength superalloys, 4.55% of the mass will be non-lunar. Perhaps the percentage of Inconel can be reduced, but this would require detailed analysis of the structures, particularly with respect to creep failure. If the high nickel or chromium content of Inconel proves to be uneconomical with lunar materials, then the Inconel might be replaced with a less exotic alloy (perhaps requiring lower operating temperatures).

Thermodynamic, mechanical, and alternator efficiencies are assumed to be 31.2%, 85.2%, and 92.4%, respectively.(3) Thus, to achieve a net 25% efficiency requires a theoretical Carnot efficiency of 50%. To achieve this Carnot efficiency with a hot side temperature of 720 K requires a maximum cold side temperature of 360 K.

The limiting failure mode of the engine will probably be from creep, i.e. the metal comprising the hot side slowly stretches. Thick walls increase resistance to creep but decrease the Stirling's thermal efficiency. An estimate of the wall thicknesses that can be used with various materials can easily be made. Based on the allowable stresses tabulated in the 1980 ASME boiler code (5) and using MTI's pressures and dimensions, the wall thicknesses (in cm) shown in Table 2.6-2 would be required.

TABLE 2.6-2 STIRLING ENGINE MINIMUM WALL THICKNESSES IN CENTIMETERS FOR VARIOUS HOT END TEMPERATURES AND MATERIALS

	1088 K	811 K	728 K	589 K
304 Stainless Steel	12.7	1.3	1.2	
316 Stainless Steel	—	.9	.8	
410 Stainless Steel	—	—	1	
SB-348 Titanium				1.9
NON BOILER CODE MATERIALS				
Inconel (6).		.11	.11	
Tungsten - 25% Rhenium (8)	.16 (6.2E+ 8 Pa yield at 1590 K)			
Mullite (9)	1.7 (low creep ceramic at these temperatures)			

Current developments are using Inconels or other nickel, cobalt based super alloys. For higher temperatures, tungsten - 25% rhenium or some other tungsten based super alloy are reasonable choices for small level production, but may exceed the productive capabilities of the Earth for the quantities required by an SPS program.

Ceramic may be useful as an outer jacket for the engine, providing high temperature strength to hold a thin metal cylinder together. This has potential to allow very high pressures and temperatures, with efficiencies approaching 50%.(1) Mullite, a ceramic of the composition $(Al_2O_3)_3 (SiO_2)_2$, may be useful for this application because it is strong at temperatures of interest and it is not into its creep range at those temperatures. Ceramic materials can be machined to fine tolerances and should be straightforward to make from lunar material. It is possible that many internal components of the engine could be fabricated from what is essentially hot pressed lunar dust.

The absorber is assumed to be similar to that used in the Brayton and Rankine systems. However, it is assumed to require only 10% non-lunar materials because of the lower maximum temperature of the Stirling engine.

2.6.5 Stirling Engine References

1. Renfroe, David, Mechanical Engineering Dept., University of Arkansas, Fayetteville, Arkansas, personal communication, June 18, 1985.

2. Beale, William T., "The Free Piston Stirling Engine: 20 Years of Development", Proceedings of the 18th IECEC, Paper No. 839111.

3. Dochat, George R., "Free-Piston Stirling Engines for Space Power", internal publication by Mechanical Technology Inc., 1984.

4. Slaby, J., "Overview of the NASA Lewis-Research Center Free-Piston Stirling Engine Activities", Proceedings of the 19th IECEC, Paper No. 849154.

5. American Society of Mechanical Engineers, "The 1980 ASME Boiler and Pressure Vessel Code, Section VIII, Div 1, Pressure Vessels". The allowable stress levels used for calculations were derived from tables UHA-23 and UNF-23.4.

6. HUNTINGTON ALLOYS HANDBOOK, page 23. INCONEL is a trademark of HUNTINGTON ALLOYS, INC., Huntington, West Virginia 25720

7. Meier, K. L.; Girrens, S. P.; Dickinson, J. M.; "Titanium Heat Pipes for Space Power Systems", Proceedings of the 15th IECEC, Paper No. 809144.

8. Moses, Alfred J., "The Practicing Scientist's Handbook", VanNostrand Reinhold, 1978, p. 861.

9. Brady, George S., "Materials Handbook 10th Ed", McGraw-Hill, 1971, p. 522.

2.7 CONCENTRATING REFLECTORS

Several of the conversion systems investigated use concentrated sunlight with a unit size of several kilowatts or more. These systems require concentrating reflectors of several square meters or larger. This section describes the design of concentrating reflectors using lunar materials. The design is not highly detailed, since the details will depend on the conversion system.

2.7.1 Design Description

Standard technology for large reflectors in space is vapor deposited aluminum on facets made of Kapton plastic film. The Kapton is supported by some open structure, typically made of a graphite epoxy composite. The facets are individually steered.

A similar design which provides the same optical performance with almost no non-lunar material was developed. This design uses vapor deposited aluminum on 25 micron aluminum foil, with an aluminum supporting structure. The reflector always points at the sun, so the structure and the reflecting surface will be in a nearly constant thermal environment. There may be transient thermal warping at the beginning and end of eclipses, but the SPS does not produce power during those periods so warping at those times will have no effect on performance.

Achievable specific mass of large reflectors is less than 0.05 kg/m^2 using Kapton facets and graphite epoxy structure.(1,pp100-102) Conversion from graphite epoxy to aluminum increases mass by a factor of 3.02, so the specific mass of the reflector technology used in this study is assumed to be 0.15 kg/m^2, with no non-lunar mass.

The facet steering devices were ignored in this mass estimate because of their small mass. Each steering device is only required to move a very lightweight facet with no atmospheric drag and insignificant gravity. Thus, each steering device should have mass of less than a kilogram. Concentration ratios used in this study would not require more than 7000 facets, so the total mass of the steering devices would not exceed 7 metric tons.

2.7.2 Concentrating Reflector Reference

1. Solar Power Satellite: System Definition Study, Parts 1&2, Vol. II, Technical Summary, D180-22876-2, Boeing Aerospace Company, Seattle, 1977.

2.8 RADIATOR SYSTEMS

The conversion systems which require large concentrating reflectors also require active cooling systems to dissipate waste heat. Two radiator concepts which were considered for this application are the heat pipe radiator (HPR) and the liquid droplet radiator (LDR). A third concept, the moving belt radiator (MBR), was found to be promising but too poorly developed to be considered.

The heat pipe radiator was found preferable for all active cooling needs in this study because it is a low technological risk and it would require less non-lunar material than the LDR.

2.8.1 Heat Pipe Radiator (HPR)

In the heat pipe radiator, coolant is pumped through a long tube. Heat pipes carry heat from the tube to radiating sheets of aluminum. A major mass component of this radiator is the coolant tube, which must be armored to prevent micrometeoroid punctures. Specific mass of the heat pipe radiator is assumed to be 8 kg/kW at a coolant temperature of 333 K.(1,3) Typically about 98% of the radiator's mass will include pipes or fins which can be manufactured from mostly lunar material.

The coolant pumped from the heat source through the ducts might be largely lunar. If coolant temperature is high, and if sodium and potassium can be extracted cost-effectively from lunar regolith, then use of NaK coolant could result in over 99% lunar material in the heat pipe radiator. Likewise, if the coolant temperature is appropriate for water, then lunar oxygen could be used for 89% of the coolant mass. Lunar oxygen could also be used to make heat pipe working fluid, such as water or methanol.

There are potential problems with HPRs. Aluminum cannot be used for the radiating sheets if the coolant temperature is near the melting point of aluminum, 934 K. Corrosion and oxidation may limit the useful lifetime of heat pipes to less than the assumed 30 year life of the SPS.

2.8.2 Liquid Droplet Radiator (LDR)

In the LDR concept, coolant is sprayed through space as a thin sheet of small droplets which radiate heat to space. The droplets are captured for re-use. Micrometeoroids pose no danger to the droplet sheet, so extensive armor is not needed. Specific mass of the LDR has been estimated to be potentially one third that of HPR at high temperatures where liquid metals would be used as the working fluid, and one sixth that of HPR at low temperatures where silicone oils would be used.(2,3) About 30% of the mass of the LDR would be coolant, which would probably not be lunar. It is possible that lunar NaK could be used if the temperature range were suitable and this material could be extracted from the lunar crust. Coolant would have to be resupplied at regular intervals due to evaporation losses. A complete LDR system has not been demonstrated, but subsystems have been tested.

2.8.3 Moving Belt Radiator

In the moving belt radiator (MBR) concept, coolant is pumped through a rotating drum. Heat is conducted through the wall of the drum to a moving belt, which radiates the heat to space. The MBR offers specific mass comparable with that of the LDR(3,4) and may require less non-lunar material than the HPR. It is not well enough developed to use in this study. Nonetheless, this radiator concept is promising and deserves further study.

2.8.4 Radiator Systems References

1. Advanced Propulsion System Concepts for Orbital Transfer Study, Final Report, Vol. 2, Study Technical Results, Boeing Aerospace Company, Seattle, Wash., D180-26680-2, 1981.

2. Mattick, A. T., and Hertzberg, A., "The Liquid Droplet Radiator - An Ultralight Heat Rejection System for Efficient Energy Conversion in Space", 23rd Congress of International Astronautical Federation, 1981.

3. Feig, Lt. J., "Radiator Concepts for High Power Systems in Space", AIAA 22nd Aerospace Sciences Meeting, Jan. 9-12, 1984, Reno, Nevada, AIAA 84-0055.

4. Prenger and Sullivan, "Conceptual Designs for 100 MW Space Radiators", NAS Symposium on Advanced Compact Radiators, Nov. 15-17, 1982, Washington, D.C., LA-UR-82-3279.

5. Sercel, Joel, Jet Propulsion Laboratory, Pasadena, California, personal communication, May, 1985.

3. STRUCTURE

3.1 INTRODUCTION

The structure of an SPS maintains the separation and orientation of the components with respect to each other. It provides the rigidity through which the attitude control system points the power conversion system at the sun and the microwave transmitter at the earth.

Each SPS concept has unique structural requirements. This section uses the silicon planar structure as an illustrative example. Silicon planar was chosen as an example because it has already been well studied, because silicon planar is a very promising concept for lunar SPS design, and because it has a relatively simple structure which makes a good example.

The most important requirement for the design is to minimize use of non-lunar materials for construction of the satellite. This requirement minimizes the importance of a low total mass.

Assembly in space is another important consideration in design of this space structure. This problem has been studied before (1-9), since launch limitations do not allow launching massive or bulky satellites into space.

In addition to these requirements, the structure must have a sufficiently high natural frequency to satisfy pointing accuracy needs and to avoid attitude control interactions.

3.2 DESIGN ALTERNATIVES

The configuration of the satellite has to account for stiffness, stability, and power distribution needs. A rectangular shape is one of the common shapes chosen for a silicon array satellite (ref 1 and 10). In this configuration, the structure bears a compression load caused by tension in the solar cell array. The aspect ratio (length/width) of a rectangular structure must be chosen to provide a natural frequency high enough to satisfy the stability requirements.

An uncommon SPS option is spin-stabilization, which is often used in smaller satellites. Some spin-stabilized configurations would have advantages. For example, arrays of solar cells could extend radially outward from a spinning central hub. Centripetal force would maintain the rigidity and orientation of the arrays, thus eliminating most structural mass.

One drawback of spinning the solar array is that an additional slipjoint is required between the spinning array and the microwave antenna. This drawback is minor: the slipjoint is better than an order of magnitude less massive than the SPS primary structure.

Another drawback is that a flexible rotating structure is relatively difficult to maneuver. This may be a minor factor because the primary maneuvering of the solar array is simple: a steady force to offset radiation pressure (both solar and microwave) and a sun-following rotation with a period of one year.

Because standard structural design was judged to be sufficient, no design or analysis of spin-stabilized or spin-stiffened solar arrays was done in this study. However, this option merits further consideration if conventional structures prove to be uneconomical.

The selected structural material should be lunar and structurally sound. Abundant lunar materials suitable for structural applications are glass, aluminum, iron, titanium, and magnesium. Foamed glass was selected by General Dynamics because of its low coefficient of thermal expansion (CTE). It was rejected for primary structure in the current study because its properties are only hypothetically known and because of the general inadvisability of building with brittle materials. Other options based on glass were dropped from consideration for the same reasons or because they require a large fraction of non-lunar material. However, foamed glass was assumed for the microwave transmitter structure because of the stringent flatness requirements of the antenna.

Magnesium was unlikely to be needed in non-structural parts of the SPS, so it was eliminated to avoid the need for additional refining processes. Titanium was not considered because, as with magnesium, there is only limited experience using it in large structures. Titanium should be reconsidered when more experience with it has accumulated or if aluminum proves to be unsuitable.

Aluminum was found to be the most acceptable structural material because it has less mass for equivalent strength than iron and because it requires less non-lunar alloying agents. Due to its relatively high CTE, some parts of the aluminum structure may require thermal coatings, shielding, insulation, or active length adjustment, as discussed in section 3.5. In high temperature applications, steel may have to be used because of its higher melting point.

An aluminum-glass composite might be developed which would combine the low CTE of glass with the superior qualities of aluminum. If developed, such a material would probably be an ideal structural material for SPS applications.

3.3 SELECTED DESIGN

The structure selected has a rectangular shape. This structure, with dimensions of 10710 m by 5348 m (aspect ratio 2), weighs about 5.98X10E6 kg. It is made of 4 modules each containing 32 bays as shown in Figure 3.3-1.

This rectangular structure was chosen mainly because it has been studied extensively in ref. 1 for a 10 GW solar power satellite. The design parameters of this 10 GW sattelite are used to approximate the configuration of a 5 GW satellite. Due to the limited scope of the project, detailed investigation of other possibilities for the structure was not undertaken.

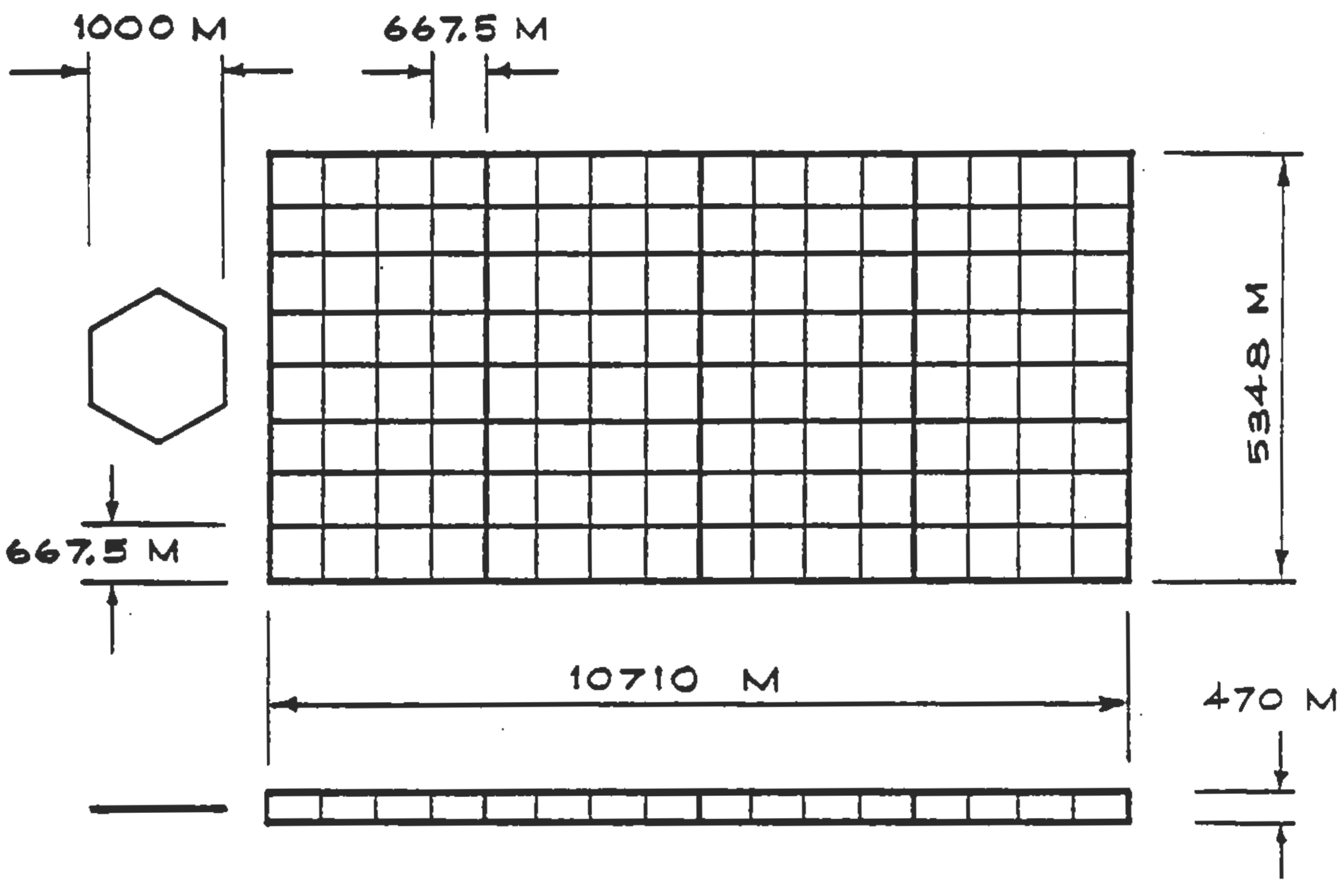

FIGURE 3.3-1, SPS OVERALL STRUCTURAL CONFIGURATION

The material selected for the beams is aluminum. A premilinary study on the feasibility of using aluminum alloys for a silicon planar structure of the same configuration with Aspect Ratio (AR) of 4 and output power of 10 GW has been done by Grumman under the Boeing contract.(1)

The structure is designed to be assembled in space. Assembly of the structure in space is a technology which is not fully developed, but preliminary studies have been done. Structural assembly can be done by robots (2,3,4,7) or by manned work stations located in orbit (5,6,8). The studies done on these problems show that the technology for such an operation is possible in the near future.

The Boeing study,(1) which uses a composite material for the structure, uses low earth orbit (LEO) as the construction site for the modules. The modules are then transferred from LEO to geosynchronous orbit (GEO) by thrusters attached to the four corners of each module. This transfer requires higher load capacities for the beams than construction of the satellite in GEO. The higher load requirements result in about 35% more massive composite structure, according to the Boeing study.

The aluminum structure described in this report uses the mass statements of a structure designed for handling the higher load requirements, although construction in GEO is assumed here. The Grumman study does not include a mass estimate for an aluminum structure constructed at GEO. It is certain that the lower loads will require less stiffness and therefore a lighter structure. The lower stiffness, however, will result in a lower natural frequency which will tend to destabilize the structure. Calculations of the natural frequency of the structure with lower stiffness were not within the scope of this study. At any rate, the results of such calculations would not significantly affect the non-lunar mass of the design.

3.4 DESIGN SUMMARY

The rectangular structure has dimensions of 10710 m by 5348 m by 470 m. The configuration of a corner bay is shown in Figure 3.4-1. The bays are made of triangular frames. The frames are designed to be assembled in space from aluminum beams. Preloaded cross cables are used to provide shear stiffness in the frames.(1)

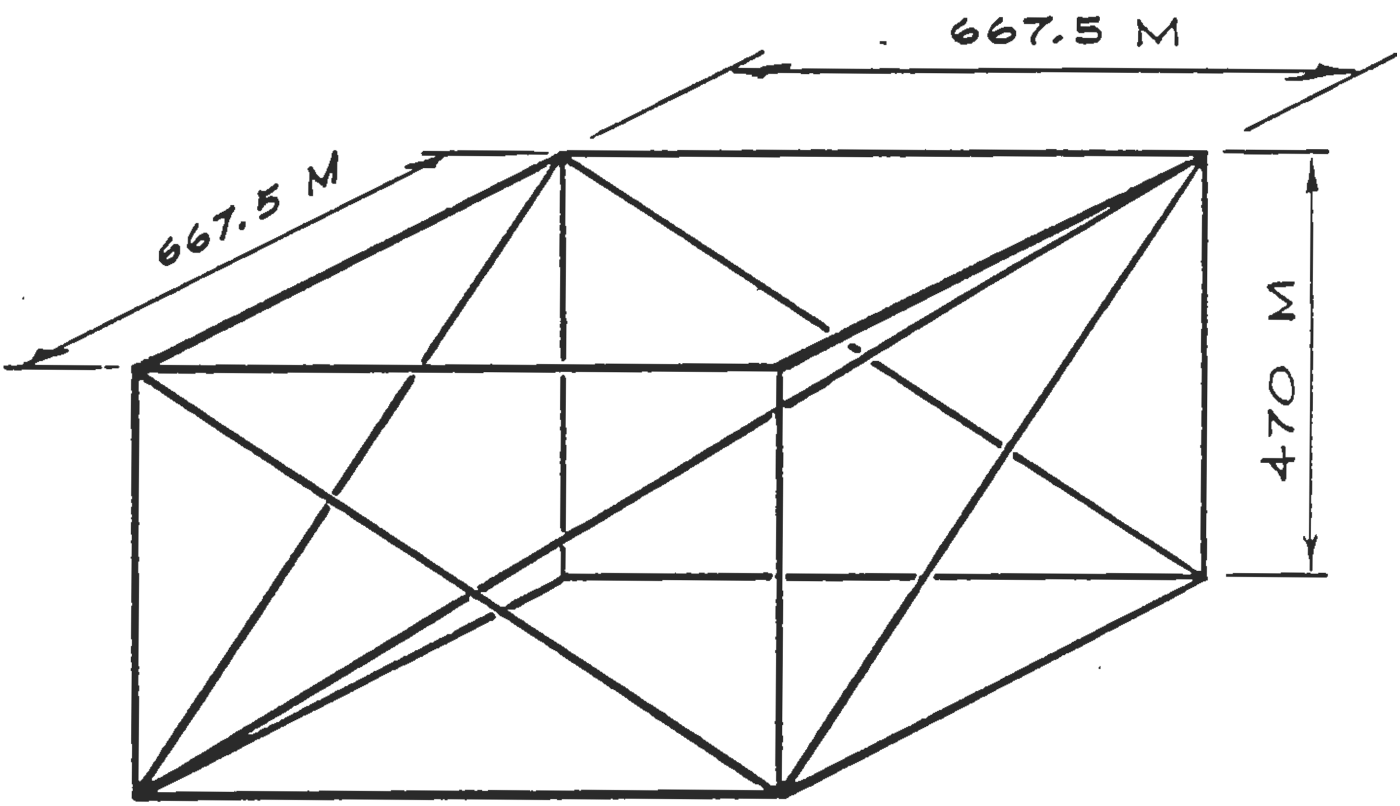

FIGURE 3.4-1, STRUCTURAL CORNER BAY

To reduce thermal gradient on the members, flanged lighting holes can be spaced to reduce shadowing as much as possible.(1) Thermal coatings can also be used to maintain temperatures and gradients within acceptable limits.(10)

The structure weighs about 5.98X10E6 kg. This mass can be reduced by replacing some of the frames with lighter and less stiff members or by eliminating some frames if the lower load requirements of construction in GEO are considered.

Based on the Grumman report, the natural frequency of the aluminum structure is estimated to be 1.3X10E-3 Hz, which is more than 100 times the orbital frequency. Such natural frequency, according to ref. 1, is adequate for control system stability. This estimate is obtained from the data provided in ref. 1 which is used to find the natural frequency of a structure with AR=4. This is probably a conservative estimate since the design described here has AR=2.

One of the major concerns with the use of aluminum in construction of the SPS structure is the deflection of the structure due to the large thermal expansion of aluminum and the temperature differential between the front and back members of the structure. This problem was considered in a short study by Grumman(1), which considered a larger structure than the one investigated here. The Grumman study shows a deflection of less than 60 m at each end of a 10.7 km long satellite. The temperature differential used for this calculation was 55 C. The angular deflection of the structure is not significant according to the Grumman report. Doubts remain, however, as to the behavior of an aluminum structure in the geo-synchronous thermal environment. In response to this concern the following section provides a qualitative assessment of the problem.

3.5 PASSIVE THERMAL CONTROL

Thermal control is a necessary concern in an aluminum SPS structure because of the large temperature change experienced when the SPS is eclipsed and because of the high coefficient of thermal expansion for aluminum. The problem can be broken down into two general areas, total linear expansion and differential expansion.

Total linear expansion is a measure of the overall change in size that the SPS experiences. This occurs when the SPS enters or leaves an eclipse. Leaving an eclipse, the SPS structure rises from the minimum to the maximum operating temperature in a relatively short period of time. This implies that the expansion also occurs on a short time scale. As a result, even though the total expansion may be small, some components may experience a high acceleration which increases the potential for damage to the system.

For example, a typical 5 GW SPS design is 10,000 m long with 5E+07 kg mass distributed uniformly along its length. The CTE of aluminum is 2.23E-05.K at room temperature. The SPS passes from the Earth's shadow into full sunlight in as little as 2 minutes. Assuming a temperature change of 100 K in the same time, the rate of temperature change of the aluminum structure must rise at 0.04 K/sec/sec or greater. The corresponding compression load at the center of the SPS is found to be 5.58E+04 newtons, or about 6.3 tons. Actual loads for the structure used here would be lower, since the structure's temperature would not rise 100 K in two minutes. The force is proportional to the mass per unit length and to the square of the structure's length, so the stress would be four times larger for a typical 10 GW SPS.

Differential expansion refers to different changes in size of connected structures. This can result from using materials with different coefficients of thermal expansion or from connecting structures that have different operating temperatures. In the SPS most of the structure is shielded from the sun by the solar cell panels. The face structure, however, which supports the solar cell panels receives direct sunlight and has an operating temperature which is estimated to be 100 degrees higher than the rest of the structure. As a result of this, when the SPS is eclipsed, the face contracts more than the back, causing the SPS to warp. Even though the differential contraction is small it is significant because it causes changes that are not axial with respect to the structural members. As the SPS is eclipsed this effect could cause permanent warpage or even failure in some structural components.

There are two potentially useful passive devices for dealing with this problem, high emissivity coatings and sun shields. The back surface of the exposed beams could be coated with aluminum oxide to achieve a high emissivity. As the emissivity of aluminum oxide ranges from about 0.1 to 0.8 depending on the thickness and anodization process, an emissivity may be chosen such that the shielded and unshielded structure will have approximately the same operating temperature. This effectively minimizes the differential contraction (assuming a uniform cooling rate) during an eclipse. The maximum linear contraction, however, would not be significantly reduced. If it proves too high then the additional step of adding sun shields to the exposed beams may be warranted. This would reduce the maximum temperature of these beams considerably. As the beams shielded by the solar cells would now be the 'hot' portion of the structure, it would be necessary to anodize them to return the system to a near uniform (but now lower) operating temperature. It is believed that by adding heat shields to all of the major structure and further adjusting the high emissivity coatings, an operating temperature near the minimum (or eclipsed) temperature may be achieved, at which point all size changes would be very small.

Nearly uniform temperatures will only exist during static operating conditions. Immediately after entering and exiting the eclipse this will not be the case and significant distortions will still occur. This is added incentive to decrease the maximum operating temperature as the coefficient of thermal expansion for aluminum decreases significantly as the absolute temperature decreases.

All of these estimates were supported by computer models developed by SRA to simulate the thermal behavior of objects in space. These calculations were based on formulas for radiative heat transfer between bodies of various geometries.(11,p242-254) Though complex, these models provided only a low level approximation of an SPS. As such, they served to support the qualitative conclusions presented above but could not provide precise numerical results. Any attempt to adequately determine the behavior of an SPS during eclipse (and the thermal control required thereby) would necessitate the development of a much more sophisticated model. Such an effort was beyond the scope of this project.

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4. POWER DISTRIBUTION SYSTEM

The Power Distribution System (PDS) controls, conditions, and transmits power from the conversion system to the MPTS. It also provides energy storage and handles fault detection and isolation.

The major mass components of the PDS are: the feeders which carry power from the individual power conversion units, such as a single generator, to the main bus; the main bus, which carries power from the conversion section of the SPS to the rotary joint; the electrical rotary joint, which carries power across the joint between the sun-facing and the earth-facing parts of the SPS; the voltage conversion units which transform the bus potential to the various potentials required by electrical subsystems; switchgear which allows individual power sources and sinks to be separated from the rest of the PDS; and energy storage facilities for powering essential subsystems during occultations and shutdowns.

The details of the PDS are dependent on the final SPS configuration; however, some concepts for the major subsystems are discussed below.

4.1 MAIN BUS

Four options were considered for the main power buses. These were to use sheet aluminum conductors, refrigerated conductors, superconducting buses, or microwave transmission. The refrigerated conductor option utilizes the decreased electrical resistance of most solid conductors at low temperature, resulting in less lost power and a less massive power conversion system. The superconducting option utilizes the fact that electrical resistance vanishes in a few materials at temperatures of 20 K or less, so no power is lost to resistance heating. The microwave transmission option replaces massive conductors with massless beams of radiant energy, but inserts an additional conversion process with its losses and mass.

The sheet aluminum option requires essentially no non-lunar material, and resistance losses can be made arbitrarily small by increasing the width or thickness of the bus. Both of the cooled systems require a great deal of Earth material as coolant and, in the superconducting case, as conductor. They also add to the system complexity and absorb some parasitic power in the cooling system. The microwave system is limited by poor efficiency and requires some non-lunar material as coolant. Thus, the sheet aluminum option was selected because it requires the least non-lunar material and is the least complex.

4.2 SWITCHGEAR

The reference SPS(2) uses liquid metal plasma valve switchgear. Two other options have been identified. The first is to use mechanical switches with solid conductors. The other is to use semiconductor switches.

Mechanical switches require little non-lunar material, but they are relatively massive because of the need for large contact areas and cannot respond very quickly. Semiconductor switches (silicon controlled rectifiers or power transistors) offer good switching speeds and could be made with little non-lunar material, but may be difficult to manufacture in space, will require radiators, and create small losses (about 99% efficient). Little information about vapor-phase switchgear was available to the study team, but the fact that it was selected in earlier SPS studies suggests that its lifetime and efficiency are high and its mass is low.

Experience with lunar material substitutions in other subsystems suggests that at least 90% of the mass in a vapor-phase switch could be replaced with lunar material, e.g. aluminum conductors and glass envelopes. However, a vapor-phase switch design has not been examined to verify this assumption.

Because of its acceptance in earlier designs and the belief that it can be built with little non-lunar material, vapor-phase switchgear was selected. However, further study is needed to select an optimum switchgear system.

4.3 VOLTAGE CONVERSION

Previous studies (1,2,3) have used an oscillator-transformer-rectifier system for DC to DC conversion at the microwave antenna. The major mass components of this system are the transformer, the inductors, and the capacitors. The transformer requires active cooling. The only non-lunar material required in any of the major items is pumps and coolant for the transformer. Non-lunar material would also be needed for the oscillator and the rectifier.

Transformers are only about 98% efficient, making the overall conversion process about 96% efficient. Modern power electronics provide DC-DC conversion at over 98% efficiency without using transformers.(4) This increased efficiency reduces non-lunar mass needed for cooling and reduces the mass of the power conversion system. The semiconductor voltage converters were assumed for this study.

4.4 ENERGY STORAGE

Stored energy is required during eclipses to maintain essential systems and to keep RF amplifier cathode heaters hot. For a 5 GW SPS, the required energy storage is 6 MW-hrs.(2, vol. 4, p. 159) Storage alternatives considered in this study are compared in Table 4-1.

TABLE 4-1 ENERGY STORAGE TECHNOLOGIES AND MASSES FOR 5 GW SPS

Technology	Density	Mass	Non-lunar Mass
Nickel-Hydrogen	40 Whrs/kg	150 T	150 T
Sodium Chloride	200 Whrs/kg	30 T	30 T
S-glass Flywheel	54 Whrs/kg	111 T	< 8 T

The sodium chloride energy density is a theoretical value, not a practical one.(3, vol. 7, p. 159)

The energy density of the S-glass flywheel was estimated from the current best performance of a steel flywheel, which is 3.125 Whrs/kg.(5) The storage capacity of a flywheel is proportional to the working stress of the material divided by its density. Steel has a working stress of 124 MPa and a density of 7.8; S-glass has a working stress of 1379 MPa and a density of 2.4, so its capacity should be better by a factor of about 36. Not all of the system's mass is in the flywheel, however, so improvement by a factor of 18 was assumed. The estimated non-lunar composition in the flywheel is probably quite conservative.

4.5 ELECTRICAL ROTARY JOINT

All SPS studies to date have used a slipring-and-brush assembly as the electrical rotary joint. In the reference design (2, vol 2, p. 76), a slipring carrying half the load of a 10 GW SPS consisted of 11,810 kg. of coin silver.(90% silver, 10% copper) The brush assemblies (aluminum, spring steel, and brushes) total 1970 kg.; the brushes are 85% silver, 12% molybdenum disulfide, and 3% graphite.

The slipring's cross-section is a hollow hexagon; each of the two contact surfaces is a 1.0 cm x 50 cm bar of coin silver.(2, vol. 3, Fig. 4.6-9) There are 832 brushes and three concentric sliprings in the reference design.

The current density in this design is about 10 A/cm^2. The power loss for a 5 GW SPS with 40 kV power distribution is 40 kW. (6, p. 104)

The use of silver molybdenum sulfide brushes on silver was chosen by Boeing to minimize abrasion, drag, and electrical losses. In that study, copper and aluminum were considered as slipring materials - both were less massive and less expensive - but were rejected due to limited space experience with them.

Selection of new slipring/brush material combinations is not within the scope of this study, so the same materials were used at all sliding contacts. New materials, such as copper or aluminum sliprings, should be reconsidered when more space experience with them has accumulated.

Three options were considered to decrease the use of Earth material in the electrical slipring. The first was to use a thin plating of coin silver on an aluminum slipring, as in the Rockwell design. (3, vol. 7, Table 3.1-13) The second was to eliminate the sliding contact entirely by using conductive vapor in a sealed enclosure. This option was rejected because of insufficient data on rotary seals in a high-voltage space environment. This option should be reviewed when more space experience with rotary seals has accumulated. The third option was to replace the sliding contact with beams of electromagnetic radiation, such as microwaves or light. This option was rejected because its low efficiency (< 90%) would significantly increase the mass of the power collection and conversion systems, and because it would require considerable non-lunar mass to build and cool the electronics.

The option of using a thin coin silver cladding on aluminum was selected. The design uses a 1 mm layer of coin silver on a 1.6 cm thick bar of aluminum, which provides the same losses as the 1.0 cm bar of coin silver. This reduces the requirement for a 5 GW SPS with the reference configuration from 11.8 metric tons of coin silver to 1.2 metric tons. It also reduces the total slipring mass from 11.8 tons to 4.0 tons.

4.6 POWER DISTRIBUTION REFERENCES

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5. MICROWAVE SUBSYSTEMS

The microwave power transmission system (MPTS) converts electrical power to microwave radiation and transmits this radiation as a coherent beam which must be kept focused on the receiving antenna (rectenna).

Electrical power is converted to microwaves by RF amplifiers. Solid state amplifiers were rejected because their low efficiencies would greatly increase the size of the SPS power conversion and distribution systems and because considerable non-lunar material would be required to cool the amplifiers. The gyrocon, klystron, and magnetron amplifier concepts were selected for study. They are discussed in sections 5.1, 5.2, and 5.3. The gyrocon was eventually dropped from consideration due to its low state of development.

The masses and projected efficiencies of the magnetron and klystron are listed in Table 5-1. The magnetron design uses more non-lunar material but is more efficient, reducing the size of the power conversion system. Thus the magnetron is preferred if the power conversion system has high non-lunar mass, and the klystron is preferred if the power conversion system contains little non-lunar mass. The critical non-lunar specific mass for the power conversion system is 0.074 kg/kW; below this value, the klystron gives lower total non-lunar mass than the magnetron. This critical value was estimated using the projected efficiencies shown below.

TABLE 5-1 COMPARISON OF KLYSTRON AND MAGNETRON DESIGNS

	Klystron	Magnetron
Demonstrated efficiency	74%	84%
Projected efficiency	85%	90%
Output power/unit	112 kW	3.2 kW
Total mass/unit	72 kg	2.32 kg
Non-lunar mass fraction	0.65%	1.25%
Non-lunar mass/power	4.21 g/kW	9.09 g/kW

Configuration of microwave antennae was outside the scope of this study, so the configuration used in the Boeing report(6) was adopted. In that design, microwave power is radiated from slotted waveguides arranged in a planar array. Two waveguide designs were considered. Both are compatible with the Boeing design, and both can be constructed entirely of lunar materials. One design, selected by General Dynamics, uses a conductive coating of aluminum in a foamed glass waveguide. The other design uses a pure aluminum waveguide. Because the waveguides do not bear a structural load, the foamed glass may be acceptable for this application. The aluminum waveguide will require stable temperatures in order to stay within the length tolerance. Providing thermal stability under varying solar illumination would greatly complicate the antenna design, so the General Dynamics waveguide is used.

Significant obstacles to acceptance of SPS have been the large unpopulated land area (about 40 km^2) needed for a receiving antenna (rectenna) and the large quantities of power provided by each rectenna (usually 5 GW). Locations suitable for selling 5 GW of power typically have high land prices. Remote locations with inexpensive land are far from population centers and require construction of power transmission lines. This dilemma could be resolved if smaller rectennae providing less power were available. (Smaller rectennae providing the same power would exceed the peak intensity limit of 300 W/m^2.)

For fixed wavelength and transmission distance, reduction of the rectenna diameter requires increasing the effective diameter of the transmitting antenna by the same factor. Increasing actual antenna size would be costly and would increase resistance losses in conductors, with an effective limit of about 2 GW of power at the ground. The effective antenna size might be increased at lower cost by placing a large lens in front of the antenna. An appropriate lens design is discussed in section 5.4. This concept is promising and deserves further study.

5.1 GYROCON

The gyrocon is a recently developed type of microwave amplifier. While the demonstrated efficiency is only 23% (13), the theoretical efficiency is competitive with other microwave amplifiers, the design has a low non-lunar material content, and it is relatively simple in construction. The construction is basically stainless steel 8 mm thick, with .05 mm coatings of copper and titanium on the inside surfaces to increase electronic efficiency and suppress electrical discharges, respectively. The design point is a 200 kW RF output at 2.45 GHz, 80% efficiency.

Table 5.1-1 gives the derived mass statement for the gyrocon.

TABLE 5.1-1 DERIVED MASS STATEMENT FOR GYROCON

Item	Mass(kg)	Composition	Non-Lunar Materials
1 Electron Gun			
1.1 Insulator Shell	1.86	Aluminum Oxide	-
1.2 Heater, Anode	0.04	Tungsten	.04
1.3 Cathode	0.007	Barium & Strontium Oxides in Nickel matrix	.007
2 Focus Coil			
2.1 Windings	2.7	Aluminum	-
2.2 Insulation	0.2	Silicon Dioxide	-
3 Output Coils			
3.1 Conductor Windings	11.0	Aluminum	-
3.2 Insulation	0.85	Silicon Dioxide	-
4 Bender Solenoid			
4.1 Windings	3.9	Stainless Steel*	.012 (carbon)
4.2 Insulation	0.3	Silicon Dioxide	-
4.3 Coating	0.005	Titanium	-
4.4 Nose	0.003	Copper	.003
5 Cavity Bodies			
5.1 Structure	116.4	Stainless Steel*	.35 (carbon)
5.2 Conductive coating	0.84	Copper	.84
5.3 Anti-discharge coat	0.42	Titanium	-
Totals	138.5 kg		1.25 kg

(*) Stainless steel composition : 1% Mn, 1% Si, 5% Cr, .3% C, 92.7% Fe

5.2 KLYSTRON

The klystron amplifier was used in the reference SPS design developed by Boeing in 1980. The best demonstrated efficiency of a klystron is 74%; the best attainable efficiency is probably about 85%.(6)

The design reference used here is Ref #10, p 175ff. In order to increase the lunar material content of the klystron, aluminum has been substituted for copper in the electromagnet. Aluminum has 64.9% of the conductivity of copper, hence requires 1.541 times the cross sectional area to carry the same current. The ratio of densities of aluminum to copper is 2.70/8.89 or copper. To be conservative, a factor of 0.50 was used when making mass calculations.

The particular klystron design analysed is as follows:

RF output:	112	kw
Voltage:	52	kV
Current:	2.8	A
Mass:	72	kg

TABLE 5.2-1 KLYSTRON MASS STATEMENT

Item	Mass(kg)	Composition	Non-Lunar Materials
1 Electron Tube			
1.1 Pole pieces	0.542	94% Fe, 5% Co, 1% V	.005 V
1.2 Cathode	0.011	90% W, 6% Ba, 4% other	.011
1.3 Insulating Shell	1.5	Aluminum Oxide	-
1.4 Leads & heater filament	0.04	Tungsten	.04
1.5 Cavities	11.0	Stainless Steel*	.033 C
1.6 Cavity lining	0.068	Copper	.068
1.7 Cavity coating	0.034	Titanium	-
2 Collector			
2.1 Body	9.0	Stainless Steel*	.027 C
2.2 Lining	0.2	Copper	.2
2.3 Coating	0.1	Titanium	-
3 Solenoid			
3.1 Conductor windings	10.0	Aluminum	-
3.2 Insulation	1.0	Silicon Dioxide	-
4 Radiators			
4.1 Surface	34.0	Aluminum	-
4.2 Coating	0.75	Titanium	-
4.3 Emissive coating	0.05	Carbon	.05
4.4 Heat pipes	4.0	Stainless Steel*	.012 C
4.5 Heat pipe fluid	0.05	Methanol (CH3OH)	.025 C
Totals	72.345 kg		.471 kg

* Stainless Steel: 92.7% Fe, 5% Cr, 1% Mn, 1% Si, .3% C

5.3 MAGNETRON

The magnetron is a well-developed technology, and is used in microwave ovens. The best demonstrated efficiency for a magnetron is 84%; the best attainable efficiency is probably about 90%.(16)

The starting point for the magnetron analysis is the design presented on pages 7-3 and 7-27 in Reference 16. This magnetron has an output of 3.2 kW, and a mass of 1.018 kg. The mass elements are listed in Table 5.3-1.

TABLE 5.3-1 PRELIMINARY MASS STATEMENT FOR MAGNETRON

Item	Mass (grams)	Composition
Power Generation		
Antenna Probe	11	Cu
Vanes	44	Cu
Shell	45	Cu
Ceramic Insulators	30	Aluminum Oxide
Filament	8	Carbon, Th, W
Magnetic Circuit		
Shell	163	Steel
Permanent Magnets	103	SmCo
Control		
Inductive Tuner	64	Cu, SmCo
Buck-Boost Coil	200	Cu, insulation
Radiator	350	Pyrolytic Graphite

Several changes were made to the composition of the magnetron to decrease the mass of non-lunar materials. The changes are detailed below.

Change #1: Replace pyrolytic graphite in the radiator with iron oxide coated aluminum. Mass increases to 1510 grams for this component, since aluminum has a lower thermal conductivity.

Change #2: Replace samarium-cobalt permanent magnets with Alnico 5 permanent magnets. Alnico-5 has a composition of 51% Fe, 24% Co, 14% Ni, 8% Al, 3% Cu. The main field magnets weigh 328 grams instead of 103. The voice coil magnet weighs 80 grams instead of 25.

Change #3: Change buck-boost coil composition to aluminum. Mass decreases from 200 grams to 100 grams. This is comprised of 90 grams of aluminum and 10 grams of fiberglass insulation.

Change #4: Replace copper parts with copper coated aluminum parts. It was decided to use copper coated aluminum rather than pure aluminum because of the severe thermal environment inside the magnetron. The parts changed are as follows:

Antenna Probe	11 grams Cu	to 5.5 grams Al	+ 2.2 grams Cu
Vanes	44 "	to 22 "	+ 0.55 "
Shell	45 "	to 22.5 "	+ 3.5 "
Inductive Tuner	39 "	to 19.5 "	+ 2.5 "

These four parts can collectively be called the RF Cavity.

Change #5: Increase size of shell to accomodate larger magnets and coils. The shell increases from 163 grams to 188 grams.

The final design is shown in Table 5.3-2.

TABLE 5.3-2 FINAL MAGNETRON MASS STATEMENT

Item	Mass (g)	Composition	NLM (g)
RF Cavity	78.25	Copper coated Aluminum	8.75 Cu
Ceramic Insulation	30	Aluminum oxide	-
Filament	8	Thorium and Tungsten on Carbon	8 CThW
Shell	188	Type 1006 Steel	.11 C
Permanent Magnets	408	Alnico-5	12.24 Cu
Buck-Boost Coil	100	Aluminum and Fiberglas	-
Radiator	1510	Iron Oxide coated Aluminum (10g FeO)	-
Total	2322.25		29.1 (1.25%)

5.4 MICROWAVE LENS

To make a Solar Power Satellite (SPS) more commercially feasible, a smaller unit size than 2.5 GW is needed. This would provide a better interface to the utility grid, reduce investment for the first unit, and allow more sites for the rectenna.

A microwave lens of large aperture can be placed in front of the transmitter to achieve good coupling between the microwave transmitter and the rectenna. The product of rectenna and lens apertures is a constant, and the rectenna area is proportional to the power level; thus as satellite power level and rectenna aperture decrease, the microwave lens grows in size as the inverse of the power level. At 1 GW the lens constitutes 4% of the satellite mass; at 200 MW the lens is 50% of the satellite. Finding the economic limit on how small the SPS can be before the increasing cost of the lens offsets the advantages of low power was outside the scope of this study.

An example design is detailed below for 1 GW size. Note that above 2 GW the lens is not needed since the size of the microwave antenna is sufficient.

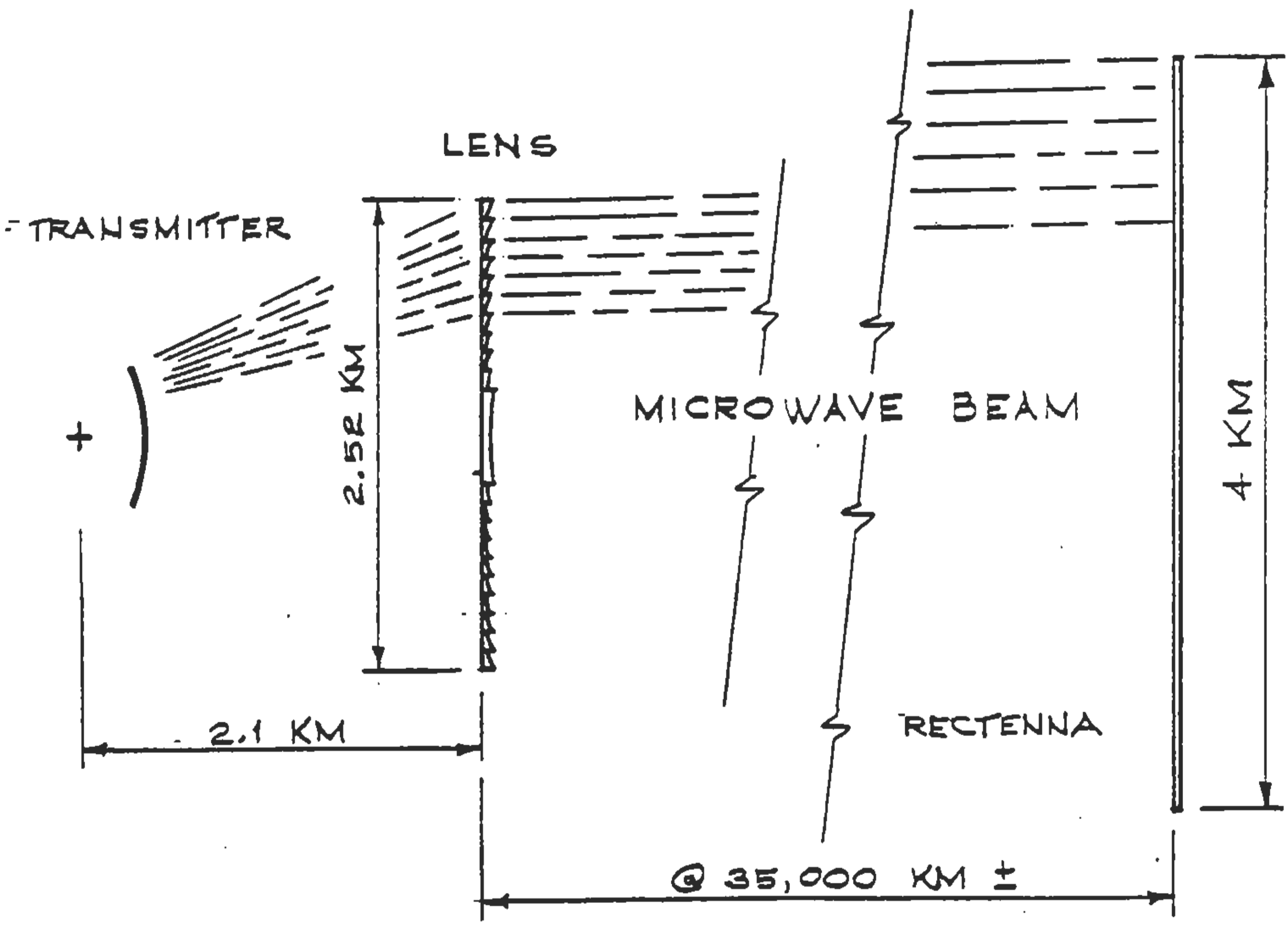

FIGURE 5.4-1, MICROWAVE LENS DIAGRAM

Assuming that 1 GW is to be delivered to the utility grid, that the transmitter frequency is 2.45 GHz, and that the peak power at the receiver is 300 w/m^2, then the nomograph in the SPS System Definition Study (6, vol. 2, p. 175) determines:

Rectenna Diameter = 4 km   Space Antenna = 2.5 km

The lens is a Fresnel lens made of layers of wire mesh. It is flat on the side toward the SPS and ridged on the side toward Earth. The index of refraction at the first and second surfaces of the lens is selected to minimize reflections. The index of refraction is varied smoothly from front to back and from center to edge within the lens by using different wire diameters. The smallest wire diameter used is 0.125 mm. The near focus of the lens is 2.1 km from the lens center.

Figure 5.4-1 shows the general arrangement of the lens. The transmission pattern at the lens was assumed to be a 10dB gaussian taper. The transmitting pattern required at the focus to achieve this has not been determined. The simplifying assumption that the waves will travel in straight lines in the interior of the lens was made.

The lens is 99.95% composed of lunar materials, as shown in Table 5.4-1.

Figure 5.4-2 shows how several lens parameters vary with distance from the center of the lens. Using data from Jasik (7, p 14-34), the theoretical transmission efficiency of the lens was found to be 97.12%. Some attendees at the 1985 Princeton Conference on Space Manufacturing questioned this result, claiming that recent experiments show much lower transmission efficiencies for microwave lenses. However, Richard Dickinson of JPL reports (17) that the experiments cited all deal with flat (e.g. Fresnel zone plate) lenses, many operating at multiple frequencies. These lenses use diffraction to redirect the microwave beam; the lens proposed here uses refraction. Dickinson believes that a single-frequency lens with 3-dimensional structure would indeed have better transmission than flat multi-frequency lenses, though he is unconvinced that efficiencies as high as 97% are attainable.

TABLE 5.4-1 MASS STATEMENT FOR 1 GW SPS MICROWAVE LENS

Item	Composition	Mass	Non-lunar Mass
Wire mesh	Aluminum	224 T	0
Spacers between mesh layers	SiO2	81	0
Lens frame	Aluminum	33	0
Support fibers	Glass fiber	30	0
Positioning spider	Aluminum	10.5	0
Thermal expansion control jackscrews	Complex	0.2	0.2
Total		378.7 T	0.2 T

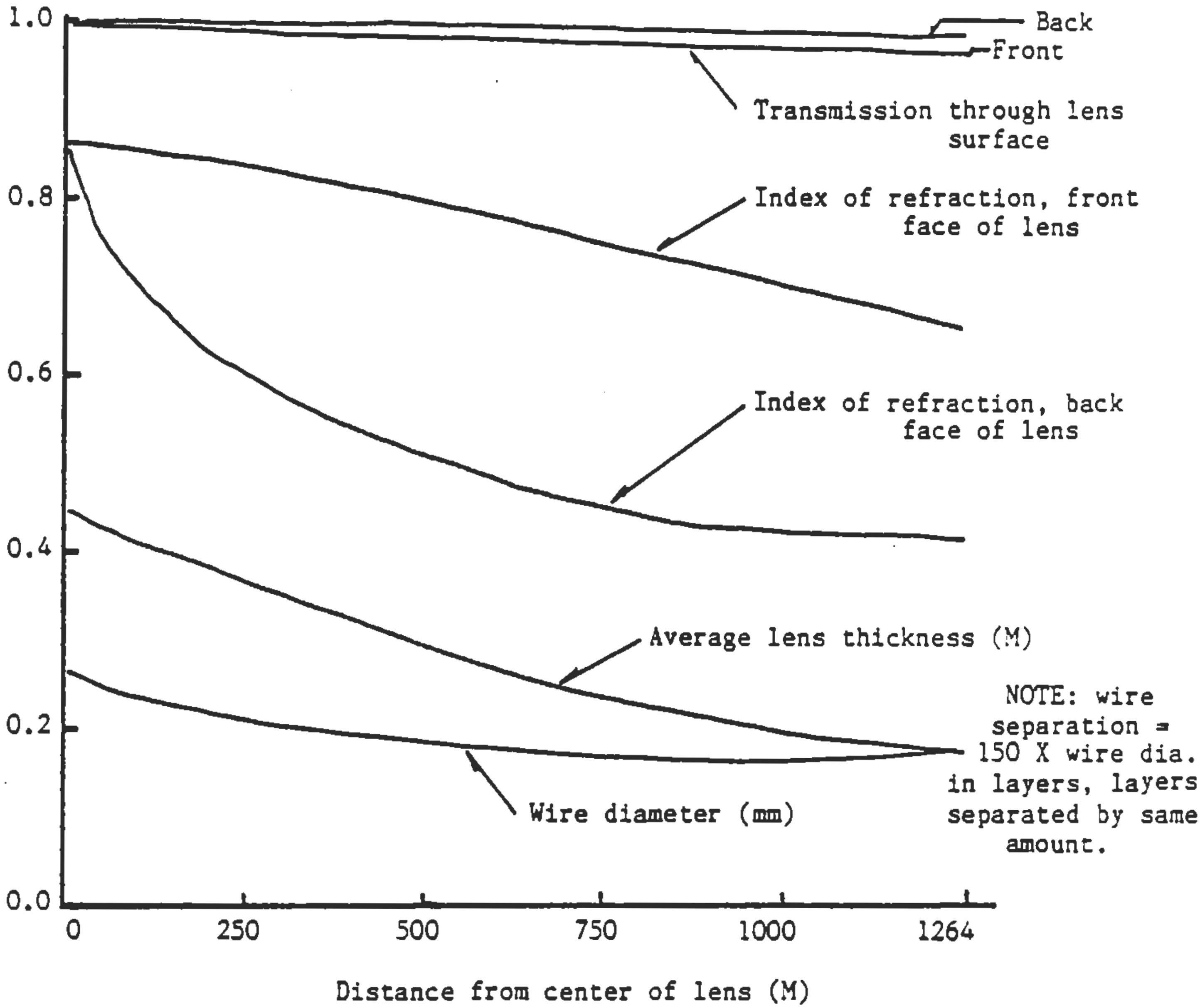

FIGURE 5.4-2, VARYING LENS PARAMETERS

Joel Sercel of JPL has pointed out (18) that lightweight microwave reflector designs exist. These relectors could replace the lens design should it prove to be invalid, so the idea of using large effective apertures to reduce SPS size should remain valid.

One drawback of a large lens or reflector is that it could block part of the sunlight falling on the SPS. A solution to this problem is to separate the microwave transmitter from the power con-ersion system by a distance at least as large as the radius of the lens or reflector. This would increase the mass of the structure and the power distribution system. An alternative is to accept the once-daily power decrease. With a lens, the decrease would occur near local midnight. The decrease would be minor because the lens blocks only about 7% of the light. With a reflector, the decrease would occur near local noon, so it would be less acceptable. The optical transmission of microwave reflectors depends on the design, so the decrease might be relatively small.

5.5 MICROWAVE TRANSMISSION REFERENCES

1. Forward, Dr. Robert L. "Starwisp", Hughes Research Laboratories Report #555, June 1983. 3011 Malibu Canyon Road, Malibu, CA 90265

2. "The Spacewatch Report", T. Gehrels and M.S. Matthews, eds., Space Sciences Building, Univ. of Arizona, 85721. #1-3: 12Nov82,1Jul83,20Apr84.

3. Rice University proposal for this study. Freeman, Dr. J. W. et al, January 1984.

4. L5 Proposal for this study, DuBose, et al. Jan. 1984

5. Request for proposal for this study, O'Neill, G. K. et al. Dec. 1983.

6. SPS System Definition Study, Volume II, Phase I Final Report, Boeing Aerospace Company for Johnson Space Center under Contract NAS9-15636, March 1979. POB 3999, Seattle WA 98124.

7. Jasik, Henry "Antenna Engineering Handbook". New York, McGraw Hill, 1961.

8. Brown, J. "Microwave Lenses", London, Methuen & Co,Ltd. 1953.

9. Lengyel, Bela A. "Reflection and Transmission at the Surface of Metal Plate Media", Journal of Applied Physics, Vol 22#3, pp265-278.

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14. Thomson-CSF, Electron Tube Division, Microwave Tube catalog, July 1983. 301 Route 17 North, Rutherford, NJ 07070.

15. NASA Technical Memorandum #58231 "Solar Power Satellite Cost Estimate" January 1981.

16. William C. Brown, NASA Contractor Report #3383, SPS Magnetron Tube Assessment Study, Contract NAS8-33157, Raytheon Company, February 1981.

17. Richard Dickinson, Jet Propulsion Laboratory, Pasadena, CA, personal communication, June 18, 1985.

18. Joel Sercel, Jet Propulsion Laboratory, Pasadena, CA, personal communication, May, 1985.

6. ESTIMATED MASS STATEMENT FOR 5 GW SPS

This section gives an estimate of the non-lunar mass required in an SPS. It includes estimates of the lunar fractions of all subsystems of the 5 GW silicon planar SPS described in reference 1, listed by Work Breakdown Structure (WBS) number. For each subsystem, the composition was estimated and lunar materials were substituted where possible. In many cases, subsystems have been re-designed for better utilization of lunar materials; these changes are described in section 6.2. Design alternatives are also described as appropriate. Many of the lunar substitutions required no design change. These substitutions were not made in the General Dynamics study because that study did not attempt substitutions in systems which comprised less than 1.2% of the SPS mass.

Table 6-1 lists the estimated mass and lunar content of each subsystem studied. Some WBS numbers are not used in the silicon planar configuration.

6.1 GENERAL ASSUMPTIONS

Most structural members are assumed to be made of aluminum alloy, with no non-lunar material. An exception is the microwave transmitter structure, which was assumed to be made of foamed glass because of the stringent flatness requirements and the demanding thermal environment of the transmitter.

Any wires or cables which traverse distances of at least a meter are assumed to use aluminum conductor and some form of glass fabric insulation, so these cables require no non-lunar material.

Radiators are assumed to be heat pipe radiators. The worst case encountered for these is 80% aluminum fins, 18% stainless steel heat pipes, and 2% working fluid. We estimate 98% lunar fraction for radiators in general; 99.6% lunar if water can be used as coolant and heat pipe fluid.

6.2 EXPLANATIONS OF SUBSYSTEM MASS STATEMENTS

WBS 1.1.1.1: Primary Array Structure.
The actual mass of the primary structure will vary greatly with the final design, but the aluminum composition is unlikely to change. If it does change, it will probably become titanium or a glass-aluminum composite, which should require no non-lunar material.

WBS 1.1.1.3: Solar Array
The silicon planar array described in section 2.1 is assumed. The only non-lunar material required in this design is less than one kilogram of dopant in the silicon cells.

The most likely alternative to silicon planar is the GaAs concentrator system, which would require less than 100 tons of non-lunar material in the solar array.

WBS 1.1.1.4: Power Distribution
This WBS number does not include energy storage or the electrical slipjoint. Acquisition buses are assumed to be 100% lunar aluminum, as are the main buses.

TABLE 6-1 ESTIMATED MASS STATEMENT FOR 5 GW SILICON PLANAR SPS

WBS #	SYSTEM	%LUNAR	MASS, M.T.	LUNAR	NON-LUNAR
1.1.1	Energy Conv.				
1.1.1.1	Structure	100	5,980	5,980	0
1.1.1.3	Solar Array	100	20,000	20,000	0
1.1.1.4	Power Dist.				
1.1.1.4.1	Main Buses	100	1,090	1,090	0
1.1.1.4.2	Acquisition Buses	100	40	40	0
1.1.1.4.3	Switchgear	90	116	104	12
1.1.1.6	Maintenance	90	621	559	62
1.1.2	MPTS				
1.1.2.1	Structure	100	467	467	0
1.1.2.2.1	Klystrons	99.3	7,007	6,958	49
1.1.2.2.2	Distrib'n Waveguide	100	625	625	0
1.1.2.2.3	Radiating Waveguide	100	2,740	2,740	0
1.1.2.2.5	Wiring Harness	61.1	91	56	35
1.1.2.2.6	Control Circuits	90	380	342	38
1.1.2.2.7	Structure	100	827	827	0
1.1.2.3	Power Dist. & Cond.				
1.1.2.3.1	Conductors	100	356	356	0
1.1.2.3.2	Switchgear	95	222	211	11
1.1.2.3.3	DC-DC Converters	98	1,112	1,090	22
1.1.2.3.4	Thermal Control	98	268	263	5
1.1.2.3.5	Energy Storage	93	112	104	8
1.1.2.5	Phase Dist.	0	12	0	12
1.1.2.6	Maintenance				
1.1.2.6.1	Gantries	95	91	86	5
1.1.2.6.*	Other	0	85	0	85
1.1.2.6.7	Tracks	100	54	54	0
1.1.2.7	Antenna Pointing				
1.1.2.7.1	CMG's	90	128	115	13
1.1.2.7.2	Star Scanners	0	1	0	1
1.1.2.7.3	Installation	0	6	0	6
1.1.3	Info Management			0	0
1.1.3.2,4	Cables	100	91	91	0
1.1.3.*	Computers, etc.	0	5	0	5
1.1.4	Att. Control			0	0
1.1.4.1	Sensors	0	1	0	1
1.1.4.2	Electric Propulsion				
1.1.4.2.2	Annual Propellant	100	50	50	0
1.1.4.2.5	Power Processors	95	112	106	6
1.1.4.2.*	Engines, Tanks, etc.	0	17	0	17
1.1.4.3	Chemical Propulsion	66	26	17	9
1.1.4.4	Structure	100	18	18	0
1.1.4.5	Maintenance	0	1	0	1
1.1.5	Communications	0	1	0	1
1.1.6	Interface Joint				
1.1.6.1	Structure	100	529	529	0
1.1.6.2	Turntable & Drive	99.5	90	90	0
1.1.6.3	Electric Slip Ring	85	13	11	2
1.1.7	Growth & Contingency	99.07	11,584	11,475	108
	TOTAL MASS, M.T.		54,967	54,454	513
	TOTAL % LUNAR	99.07			

WBS 1.1.1.6: Maintenance
The maintenance system is mostly annealing systems and the gantries and tracks to support them. Tracks are assumed to be structural aluminum, gantries are largely structure but include motors, and at least half of the annealer mass is assumed to be radiators, solar arrays, and power processors. Minor components of the maintenance system include cargo handling and docking systems, which were assumed to be non-lunar due to complexity. Thus, 90% lunar is probably a conservative estimate of the composition of this system.

WBS 1.1.2.1: Microwave Power Transmission System (MPTS) Structure
As mentioned above, the transmitting antenna must remain flat to high precision. Because of variations in the antenna's thermal environment, this flatness can only be maintained by using structural materials with low CTE, or by using active structural adjustment. A passive structure made of foamed glass is assumed.

WBS 1.1.2.2.1: RF Amplifiers
Klystrons are assumed to be more than 99% lunar, as described in section 5.2. With any power conversion system but silicon planar, magnetrons would replace klystrons because the magnetrons' higher efficiency would reduce non-lunar mass in the power conversion system.

WBS 1.1.2.2.6: Control Circuits
The mass of control circuit components is difficult to estimate. The most massive component of each control circuit is probably the radiator for the klystron circuit breaker, a semiconductor switch. The radiator is assumed to be 99.6% lunar and to constitute 90% of the control circuit's mass. All other components are assumed to be made on Earth due to their electronic complexity.

WBS 1.1.2.3.3 DC-DC Converters
As described in section 4.3, an electronic voltage converter was assumed to replace the transformer used in previous designs. The primary components are inductors (iron, silicon, aluminum), capacitors (aluminum and silicon dioxide), and a non-lunar semiconductor switch. There must also be a control component, which is assumed to be non-lunar. Elimination of the transformer would reduce total mass, but replacement of non-lunar electrolytic capacitors with solid-state lunar capacitors would slightly increase mass. To be conservative, it is assumed that total mass is unchanged.

WBS 1.1.2.3.4 Thermal Control
The mass of the radiators which cool the DC-DC converters is assumed to decrease by 50% from its original value due to the increased efficiency of the electronic power converter, as compared to transformers.

WBS 1.1.2.3.5 Energy storage
In the reference SPS, nickel-hydrogen batteries provided energy storage. Flywheel storage technology (section 4.4) was assumed here to reduce non-lunar mass.

WBS 1.1.2.5 Phase Distribution
This component includes mostly electronics and fiber optics, which would be

difficult to manufacture in space. The total mass of this component is so low that no attempt was made to substitute lunar materials.

WBS 1.1.2.6 MPTS Maintenance
The reference MPTS maintenance system is heavily oriented to manned operation. Technology being developed for the manned space station suggests that many of these operations will be automated by the time an SPS could be functional. The impact of this change has not been estimated for the maintenance system, but it seems likely that the reduced need for life support systems and structures would result in a simpler and less massive system. Because of the uncertainties, the estimated mass of this subsystem was left unchanged.

WBS 1.1.2.6.1 Gantries
This component is mostly aluminum structure. It also includes motors and related control systems, which are assumed to be non-lunar.

WBS 1.1.2.6.7 Tracks
These aluminum tracks support the gantry.

WBS 1.1.2.7 Antenna Pointing
The most massive item in this system is the Control Moment Gyros (CMG), which are essentially spinning blocks of aluminum or iron. It is assumed that at least 90% of the CMG mass can be easily replaced with lunar material. The star scanner has low mass and high complexity, so no attempt was made to use lunar material in it.

WBS 1.1.3 Info Management
The major mass component is cabling. As described in the introduction to this section, cables are assumed to be 100% lunar. The reference design includes five tons of computers, but this estimate probably includes much more than computers.

WBS 1.1.4.2 Electric Propulsion
The most massive component of the electric propulsion system is power processing and associated thermal control. These components are assumed to be similar to those on the MPTS. The rest of the electric propulsion system includes electric thrusters and fuel systems which are assumed to be made on Earth.

The reference design uses argon propellant in ion thrusters. The resupply of 50 tons of argon annually would more than triple the non-lunar requirements of the SPS over a 30-year lifetime. It is assumed here that some lunar material, perhaps aluminum, calcium(3), or oxygen, will be used as propellant. The effect of this propellant change on the rest of the system is beyond the scope of this study.

The propulsion system and fuel requirements used here are for the silicon planar configuration, which flies perpendicular to the orbital plane. All other power conversion systems must fly perpendicular to the ecliptic plane, which would significantly increase the fuel requirements.

WBS 1.1.4.3 Chemical Propulsion
The chemical propulsion system includes 20 tons of maneuvering reserve fuel

for the hydrogen-oxygen motors. This fuel must be annually re-supplied. It is assumed that the 17.1 tons of oxygen will be lunar.

WBS 1.1.6 Interface Joint
This component is mostly aluminum structure. The turntable is also mostly aluminum structure, and many of the roller parts are steel. The electric slipring design is described in section 4.5.

WBS 1.1.7 Growth and Contingency
Estimating the growth allowance was outside the scope of this study. The value from the Boeing study, 26.7%, was assumed for comparison purposes. It is assumed that most growth would be in the largely lunar items, so the lunar fraction of the growth margin was assumed to be the same as for the rest of the satellite, 99.07

6.3 REFERENCES FOR ESTIMATED SPS MASS STATEMENT

1. Harron, R. J., and Wadle, R. C., Solar Power Satellite Cost Estimate, NASA Technical Memorandum 58231, Johnson Space Center, Jan, 1981.

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7. CONCLUSIONS & RECOMMENDATIONS

7.1 CONCLUSIONS

It is possible to design an SPS which contains less than 1% as much non-lunar material as the earth baseline design, with an increase in total mass of less than 8%. This implies a cost reduction of over 97%, assuming a 50:1 cost ratio favoring lunar materials. The design uses mostly existing technology and is suitable for automated manufacture in space.

Photovoltaic SPS concepts are clearly preferable to turbine engines for lunar resource utilization. Besides the great advantage in non-lunar mass required, photovoltaic systems are much better suited to automated manufacture in space. Photovoltaic designs, with the partial exception of TPV, contain large numbers of identical simple components. Turbine engines have a great variety of parts, many of them complex, most of which are used in limited numbers. Stirling engines occupy the middle ground in both complexity and volume of components.

Whether or not repeated annealing of solar cells heals radiation damage is an important but not critical question. Without annealing, the total mass of the silicon planar SPS would increase by 50%, but the mass of the GaAs SPS design would increase less than 1%. Neither would require a significant increase in non-lunar mass, so thermal systems would still have no advantage over photovoltaics and silicon planar would still be preferred over GaAs concentrator designs.

This study determined that significant mass savings can be realized in the silicon planar design by canting the panels at 12 degrees and rotating the SPS at each equinox, i.e., twice a year. This reduces the solstice cosine loss from 8.29% to 2.19%, without requiring the satellite's attitude to be continuously adjusted to face the sun squarely.

Heat pipe radiators are preferable to liquid droplet radiators. A possible exception to this conclusion would be the temperature range where sodium-potassium could be used as droplet fluid. This technology is speculative at present, and the recoverability of lunar potassium is questionable. Moving belt radiators are very promising, but remain to be demonstrated in space.

Aluminum with passive thermal control is suitable for most structural applications in the SPS. An exception is the microwave antenna structure, which must remain very flat despite large temperature excursions. A lunar-based material, foamed glass, meets this requirement, but has not been demonstrated.

Flywheels composed of lunar glass appear to be the most promising SPS energy storage technology. Glass flywheel technology is still uncertain, however.

The klystron is the preferred microwave amplifier for photovoltaic SPSs. Although less efficient than the magnetron, the klystron's low non-lunar mass is a clear advantage with either silicon or GaAs conversion systems.

A new type of microwave lens proposed in this study, or a microwave reflector, might allow economic construction of SPSs as small as 200 MW.

7.2 RECOMMENDATIONS

The recommendations which are judged by the study team to be most important are presented first.

There should be a strong effort to find and characterize a structural material which has very low coefficient of thermal expansion and which can be made from lunar material. Such a material is vital to construction of the SPS microwave antenna, and would be useful in other structural applications. Recent work on lunar concrete is promising, and should be encouraged. A glass-metal composite would probably be ideal.

The microwave lens concept reported in this study should be verified and investigated. If valid, this concept could greatly improve SPS economics.

The GaAs concentrator design should be thoroughly optimized. It may be possible to reduce both the total mass and the non-lunar mass by more than half. This would make the GaAs design more economical than silicon planar, if silicon cannot be repeatedly annealed or if manufacture of silicon cells in space proves to be very costly.

Moving belt radiator technology should be followed closely. If an MBR can be built from lunar material, it has potential to greatly reduce the mass of all the power conversion systems which require active cooling.

TPV and Stirling technologies should be followed. Major improvements are possible in either of these technologies, and could make TPV or Stirling strong candidates for SPS power conversion. This is especially true if a lightweight lunar radiator such as the MBR can be developed.

LIST OF ACRONYMS & ABBREVIATIONS

AR	Aspect Ratio
CMG	Control Moment Gyro
CPC	Compound Parabolic Concentrator
CTE	Coefficient of Thermal Expansion
GaAs	Gallium Arsenide
HPR	Heat Pipe Radiator
LDR	Liquid Droplet Radiator
LMR	Liquid Metal Radiator
MHD	MagnetoHydroDynamic
MPTS	Microwave Power Transmission System
MTI	Mechanical Technology, Inc.
NLM	Non-Lunar Mass
PDS	Power Distribution System
Si	Silicon
SPS	Solar Power Satellite
SRA	Space Research Associates
SSI	Space Studies Institute
TPV	ThermoPhotoVoltaic