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SOLAR ELECTRIC GENERATING SYSTEM
RESOURCE REQUIREMENTS AND THE
FEASIBILITY OF ORBITING
SOLAR REFLECTORS

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SOLAR ELECTRIC GENERATING SYSTEM RESOURCE REQUIREMENTS AND THE FEASIBILITY OF ORBITING SOLAR REFLECTORS

THESIS

Presented to the Faculty of the School of Engineering

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by

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Preface

Solar energy is looked upon by many as the certain solution to the energy crisis. Tax advantages are proposed for installation of solar heating and cooling facilities in homes, and even law suits are being filed relative to Sun rights. Thus, it is not surprising that in the past few years several proposals have been made to use solar energy to generate electricity on a commercial scale.

In this study, I have attempted to assess fairly the consumption of resources of four proposed solar electric generating systems, and to objectively evaluate the feasibility of orbiting solar reflectors. I have approached this study from the point of view that each system should pay for its own developmental cost, and that since all of the systems are somewhat speculative, it should not be assumed that more than one copy of any of the systems will ever be built.

Although most of the data for this study was gathered from the work of others, I have tried to repeat calculations in order to verify to myself that all factors have been considered. In those cases where verification was impossible, I have indicated that fact.

The entire field of solar electric power generation is so new that consistent information is very difficult to find. Depending on the optimism of the author, data on resource consumption varies widely.

In gathering information for this study, I have tried to utilize the more conservative estimates.

Throughout this study I have been blessed with the help of many kind people. A list of all their names would be prohibitively long. However, I would like to take this opportunity to again thank the many people who sent me materials. These materials proved to be of invaluable assistance to me in this research effort. In addition, I am grateful to my typist, Mrs. Joyce Clark, for her professional work and helpful suggestions.

I especially would like to thank Lieutenant Colonel Hugo Weichel, deputy head of the Air Force Institute of Technology department of physics, my advisor, for his expert guidance and constructive criticism during the course of this research project.

Finally, I would like to extend a special word of thanks to my wife who lovingly supported me throughout the long months of preparation of this document.

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List of Symbols and Abbreviations

a	acceleration
A_c'	reflecting area of the secondary mirror
AFIT	Air Force Institute of Technology
AOSR	AFIT orbiting solar reflector
A_p	reflecting area of the primary mirror
AU	Astronomical unit
$COTV_G$	cargo orbital transfer vehicle for construction in geosynchronous orbit
$COTV_L$	cargo orbital transfer vehicle for low Earth orbital construction
d	distance between two objects
d_G	distance of the AOSR from the Earth's surface
D_s	diameter of the Sun
DDT&E	Design, development, testing, and evaluation
E_v	illuminance of the Sun
ERDA	Energy Research and Development Administration
F	force
f	focal length of an optical device
G_e	universal gravitational constant
GEO	geosynchronous orbit
H_o	solar constant
H_λ	contribution to the solar constant by a specific wavelength

$H_{\Delta\lambda}$	contribution to the solar constant by an interval of wavelengths
HLLV	heavy lift launch vehicle
I_e	irradiance measured on the Earth
I_m	illuminance of a full moon directly overhead on a clear night
IA	area of the Sun's image as focused on the Earth by an orbiting solar reflector
JPL	Jet Propulsion Laboratory, California Institute of Technology
K	luminous efficacy
K(r)	kinetic energy
LEO	low Earth orbit
LH_2	liquid hydrogen
LO_2	liquid oxygen
m	mass
m_e	mass of the Earth
m_s	mass of a satellite
M_I	Magnification of a one-mirror optical system
M_{II}	Magnification of a two-mirror optical system
MM	mirror matrix
n	index of refraction
NASA	National Aeronautics and Space Administration
O	center of Fresnel Mirror
OSR	orbiting solar reflector electrical generating system
P	pressure

P_i — solar power intercepted by the primary mirror

P_v — a portion of the solar constant in photometric units

PLV — personnel and priority cargo launch vehicle

POTV — personnel orbital transfer vehicle

q — area of solar cells

r — distance from the center of the Earth to a satellite

R — radius of curvature of a mirror

r_{oo} — radius of the Earth

S_i — distance between an optical device and the image it produces

S_o — distance between an object and an optical device

SI — diameter of the Sun's image

SSP — Satellite solar power electrical generating system

SSTO — single stage-to-orbit vehicle

STPS — solar thermal power system

T — transfer matrix

tan — tangent

TBD — to be determined

TP — terrestrial photovoltaic electrical generating system

TST — terrestrial solar thermal electrical generating system

U(r) — potential energy

WPAFB — Wright-Patterson Air Force Base

x — distance from the center of the mirror, measured along the optical axis

y — distance from the optical axis

y_λ	luminous efficiency
Z	system matrix
α, β	angles
$\mathcal{E}$	difference in energy between two orbits
γ	angular diameter of the Sun as seen from the primary mirror
θ_v	luminous flux intercepted by the primary mirror
θ_d	half angle of cone of light leaving the AOSR tertiary mirror

List of Units

Btu	British thermal unit
C	Centigrade
cm	centimeter
F	Fahrenheit
ft	foot or feet
g	gram
gal	gallon
GW_e	gigawatt of electrical generating capacity
hr	hour
kg	kilogram
km	kilometer
kwhr	kilowatt-hour
lb	pound
m	meter

mil	0.001 inch
mill	0.1 cent
mm	millimeter
mW	milliwatt
MW	megawatt
MWe	megawatt of electrical generating capacity
N	newton
nm	nanometers
psi	pounds per square inch
sec	second
π	3.1415926

Subscripts

i	image
o	object
x	general variable
y	general variable
1	first item of a system
2	second item of a system

Abstract

The potential consumption of natural resources by four types of solar electric generating systems was evaluated. The four systems included a terrestrial solar thermal system, a terrestrial photovoltaic system, an orbiting solar reflector system, and a satellite solar power system. Each system, assumed to be operational by the year 2000, was evaluated on its projected consumption of materials, land, water, manpower, energy, and money.

The evaluation demonstrated that, per megawatt of electrical generating capacity, terrestrial systems would consume less material, manpower, energy, and money. This resulted primarily because they would not require massive space transportation and construction systems and expensive developmental programs. It was also shown that construction of terrestrial systems would require fewer technological advancements and would pose less of a threat to the environment.

A feasibility study of orbiting solar reflectors demonstrated that single-mirror systems may be useful for power generation in space but that multiple-mirror systems have little applicability.

I. Introduction

Background

During the past few years, the diminishing nature of the world-wide supply of fossil fuels has become increasingly apparent. In an address before the nation, President Carter stated that ". . . we must start now to develop the new, unconventional sources of energy we will rely on in the next century [Ref 9]."

One of the ways proposed to solve the "energy crisis" would be to use the Sun as an energy source. In particular, it is proposed that solar energy could be used to generate electricity on a commercial scale. This would be desirable because the Sun provides energy without the need for mining and refining of fuel supplies, because it is essentially inexhaustible, and because solar generated electricity is practically pollution free.

However, the Sun is not necessarily a perfect energy source. When compared to conventional methods of generating electricity, generation of electricity from solar energy is more expensive, requires larger generating plants (Ref 8:1-2), and is hindered by the constant motion between the Sun and the Earth.

Problem

Several proposals have been made relative to the design of a commercial solar electric generating system. The primary purpose of this thesis was to evaluate the potential consumption of natural resources by four of these proposed systems: a terrestrial solar thermal (TST) system, a terrestrial photovoltaic (TP) system, an orbiting solar reflector (OSR) system, and a satellite solar power (SSP) system.

A secondary purpose of this thesis was to evaluate the feasibility of orbiting solar reflectors as Sun imaging systems.

Scope

Each of the four systems listed above was evaluated on its consumption of the following natural resources: materials, land, water, manpower, energy, and money. The usefulness of the results of this evaluation is limited by the fact that the four systems are at different stages of development. Therefore, Chapter V is included to point out some of the major technological advancements that will have to be made, and some of the questions that will have to be answered before these systems can become operational.

In addition, the usefulness of this resource evaluation is limited by the uncertainty of some of the resource consumption data. For example, much of the source information for this analysis is considered, by its authors, to be the result of preliminary estimates. In the future, as more accurate information becomes available, it will be necessary to reconsider the conclusions of this study in light of the new information.

Assumptions

It is assumed in this analysis that all of the systems either proposed or evaluated could be operational by the year 2000. It is furthermore assumed that the necessary mining, refining, and manufacturing facilities could be built in time to produce the tremendous quantities of materials needed for construction and operation of the systems. In addition, the analysis is based on the assumption that there will not be large scale social intervention, such as the environmentalist fight over the Alaskan oil pipeline, that would force system designers to adopt significantly more costly construction and maintenance procedures.

General Approach

The major task of this research effort was to collect data relative to the four systems evaluated. Every effort was made to compare the systems using the same assumptions. Ground rules were established for this purpose. Where necessary, modifications were made to existing designs, but only in those instances where it was believed to be in the best interest of the system being altered.

A space transportation system was developed, based on work by NASA, and applied as fairly as possible to the OSR and SSP systems, the two systems using space components. In those cases where detailed system information was not available, independent calculations were made based on available information. As a final step in the resource evaluation, all data and calculations were assembled, and the four systems were compared on the basis of their resource consumption per megawatt of electrical generating capacity.

The feasibility of orbiting reflectors was determined by considering the physics of space optics and by considering the advantages and disadvantages of several large space reflectors.

Literature Search

A literature search showed that work has been done on orbiting solar reflectors by at least six companies or individuals in the past ten years. These include: the Goodyear Aerospace Company, the Westinghouse Defense and Space Center, the Space Division of Rockwell International, the Boeing Company, and NASA, as well as independent work by Krafft Ehricke of Rockwell International. Information relative to this work can be found in references (6, 7, 13, 14, 15, 20, 21, 34, 39).

Sequence of Presentation

Chapter II concerns the physics of space optics and the impact of this physics on the feasibility of orbiting solar reflectors. Chapter III applies the results of Chapter II to a particular orbiting solar reflector proposal. The applicability of orbiting solar reflectors is then discussed.

Chapter IV explains in detail the procedure and results of the resource evaluation. This is followed, in Chapter V, by a discussion of some of the major technological advancements that must be made, some of the environmental concerns that must be resolved, and some of the general questions that must be answered before the four systems evaluated in Chapter IV could be realized. Some concluding comments about the significance of the results of this research are found in Chapter VI.

II. Solar Optics

> The symmetrical properties of a thin lens
>
> can easily be shown by an experiment in which a lens is used to focus the parallel rays of the sun to a point on a piece of paper or cardboard [Ref 36:228].

The above statement must be read carefully. As the statement claims, the rays from the Sun which are parallel can be approximately focused to a point. However, the statement must not be interpreted to mean that, in all cases, all of the Sun's rays can be focused to a point or that all of the Sun's rays are mutually parallel. This is because "the diameter of the Sun's disk as seen from the Earth is approximately 0.5 degrees, or about 0.01 radians (Ref 31:9)." Thus, the Sun is so large that even for optical systems on Earth, the Sun appears not as a point but as an object of finite size. When imaged, the Sun's image, although frequently small, is nevertheless of some finite size. This chapter will point out the implications of this fact on efforts to produce an image of the Sun on the Earth's surface using orbiting mirrors. Since parallel light is commonly referred to as collimated, these two words will be used interchangeably in this chapter.

The fact that sunlight is not collimated can be illustrated by using one of the basic laws of geometrical optics. This law states that a converging (concave) mirror which is perfect, (that is, a mirror free of all aberrations) will reflect collimated light to the focal point of the mirror (Ref 18:947). Thus, to determine if sunlight is collimated, it is simply necessary to see if a perfect mirror, with the proper focal length, could focus the sunlight to a diffraction limited point on the Earth's surface. The following calculation, for a mirror in geosynchronous orbit, shows that sunlight cannot be focused to a point.

The governing equation for a perfect mirror is known as the Gaussian formula. Symbolically, it is as follows

$$\frac{1}{f} = \frac{1}{S_i} + \frac{1}{S_o} \tag{1}$$

where S_o is the distance between the object and the mirror, S_i is the distance between the image and the mirror, and f is the focal length of the mirror (Ref 24:108, 126). In the case of a mirror in geosynchronous orbit that reflects the Sun, $S_o = 1.49 \times 10^{11}$ m (Ref 43:F-117). Recall that collimated light will converge to the mirror focal point. Therefore, since in Eq (1), $S_i \neq f$ unless S_o = infinity, it follows that sunlight is not collimated and cannot be focused to a point.

A portion of this thesis was concerned with aspects of orbiting solar reflectors. An important question relative to these reflectors is whether it is possible to consider sunlight as being approximately collimated. If sunlight is approximately collimated, then it would still be possible to focus the Sun to a fairly small diffraction limited spot.

The above question can be answered by using the equation for the diameter of the Sun's image

$$SI = (M_I)(D_s) \tag{2}$$

where D_s is the diameter of the Sun and M_I is the magnification of the reflector:

$$M_I = -\frac{S_i}{S_o} \tag{3}$$

(The minus sign may be neglected since it refers to the orientation of the image which is of no significance when dealing with a symmetrical object) (Ref 24:112, 126). The subscript, I, in Eq (3) indicates that the magnification is for a one-mirror system.

For a reflector in geosynchronous orbit, $S_i = 3.59 \times 10^7$ m, $S_o = 1.49 \times 10^{11}$ m, and $D_s = 1.39 \times 10^9$ m (Ref 43:F-117). Using these numbers in Eq (2), the diameter of the Sun's image as focused on the Earth would be $SI = 3.34 \times 10^5$ m = (207 miles).

The conclusion to be drawn from the result that SI = 207 miles, is that, at least in some applications, it is not even safe to consider sunlight as approximately collimated. Although the Sun is very far from the Earth, it is also very large. Thus, optical designs for Sun imaging systems should not be based upon collimated incident radiation but must rather consider the Sun as an object to be imaged.

Reduction of the Sun's Image

It is possible to reduce the size of the Sun's image. However, as this section of the thesis will demonstrate, such a reduction in image size can be accomplished only at the expense of larger and more complex optical configurations.

A typical two-mirror reflection system is shown in Fig. 1. In this discussion, the two mirrors will be referred to as the primary and secondary mirrors. The primary mirror is defined as the mirror which would first intercept the sunlight. The sunlight would then travel from the primary to the secondary mirror, where it would be reflected again and redirected to the Earth. Although the analysis presented below is only for two-mirror configurations, it could be extended to configurations of three or more mirrors.

The magnification of a two-mirror system is given by

$$M_{II} = \left(\frac{S_{i2}}{S_{o2}}\right)\left(\frac{S_{i1}}{S_{o1}}\right) \qquad (4)$$

where the subscripts 1 and 2 refer to the first and second mirrors, respectively, the subscripts i and o refer to the image and object distances, respectively, and the subscript II indicates that the magnification is for a two-mirror system (Ref 24:115). Analogously to Eq (2), the diameter of the Sun's image produced by a two-mirror system would be

$$SI = (M_{II})(D_s) \qquad (5)$$

Thus, for geosynchronous solar reflectors, where $S_{i2} = 3.59 \times 10^7$ m, $S_{o1} = 1.49 \times 10^{11}$ m, and $D_s = 1.39 \times 10^9$ m, the ratio of the size of

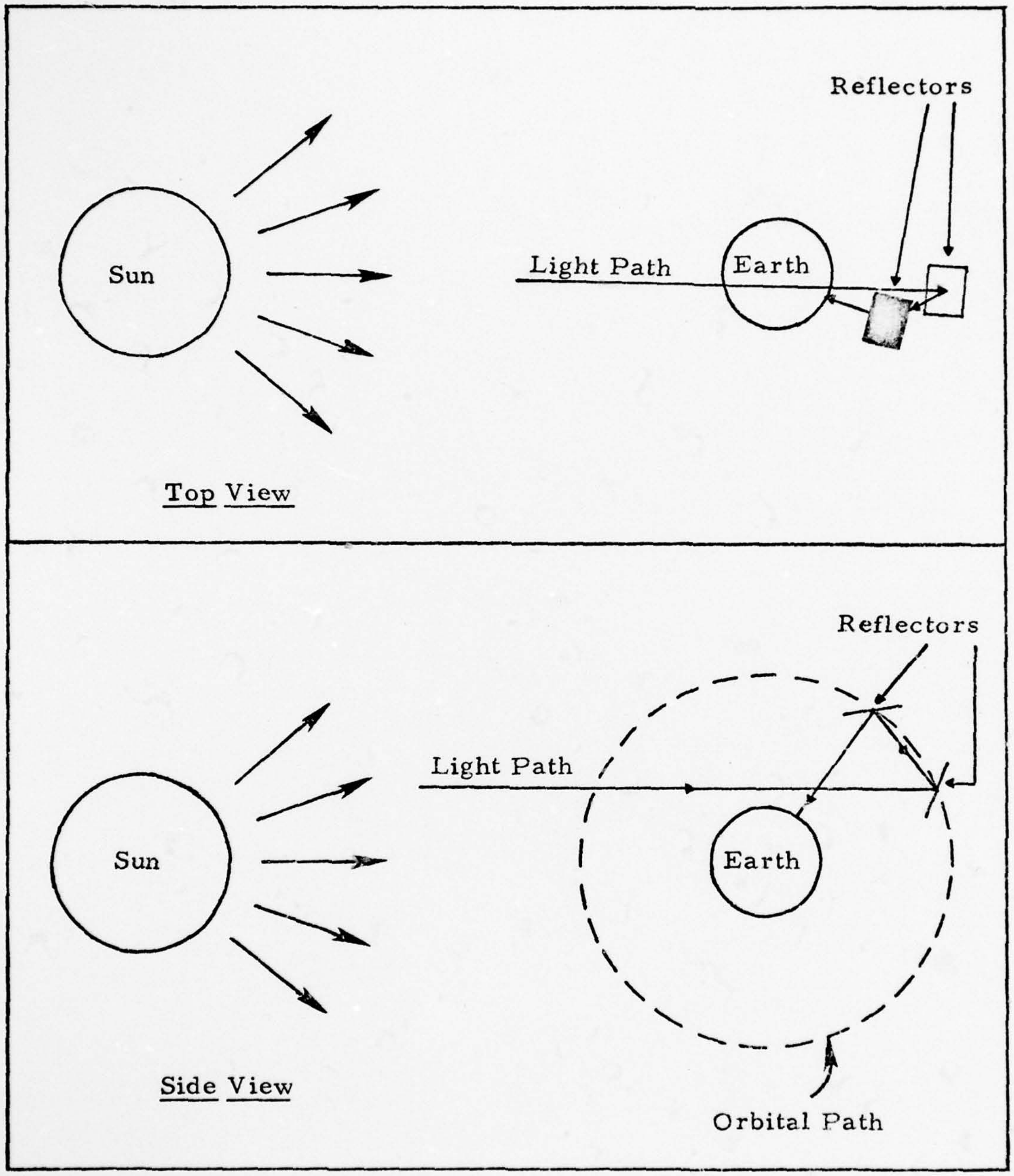

Fig. 1. Two-mirror Reflection System

the Sun's image between a two-mirror and a one-mirror system is

$$\frac{(M_{II})(D_s)}{(M_I)(D_s)} = \frac{S_{i1}}{S_{o2}} \tag{6}$$

as obtained by dividing Eq (5) by Eq (2).

The conclusion to be drawn from Eq (6) is that it would be possible to decrease the size of the Sun's image as focused on the Earth. All that would have to be done would be to position the two mirrors such that S_{i1} was less than S_{o2}, a relatively easy thing to do. However, it would probably not be an advantageous thing to do, for the reasons discussed below.

Figure 2 illustrates geometrically why two-mirror reflection would probably not be advantageous. Shown in Fig. 2 are two primary mirrors, a concave and a convex mirror. Each would reflect sunlight and each could be coupled with a secondary mirror to reduce the size of the Sun's image. Also shown in Fig. 2 is the cone of reflected light that would be produced by each mirror. The marginal rays of these cones of light are drawn with dashed lines in the region of space where S_{i1} would be greater than S_{o2}, and with solid lines in the region where S_{i1} would be less than S_{o2}. Therefore, in order for a two-mirror system to reduce the image size, the secondary mirror would have to intercept the cone of light from the primary mirror in the region of the cone drawn with solid lines.

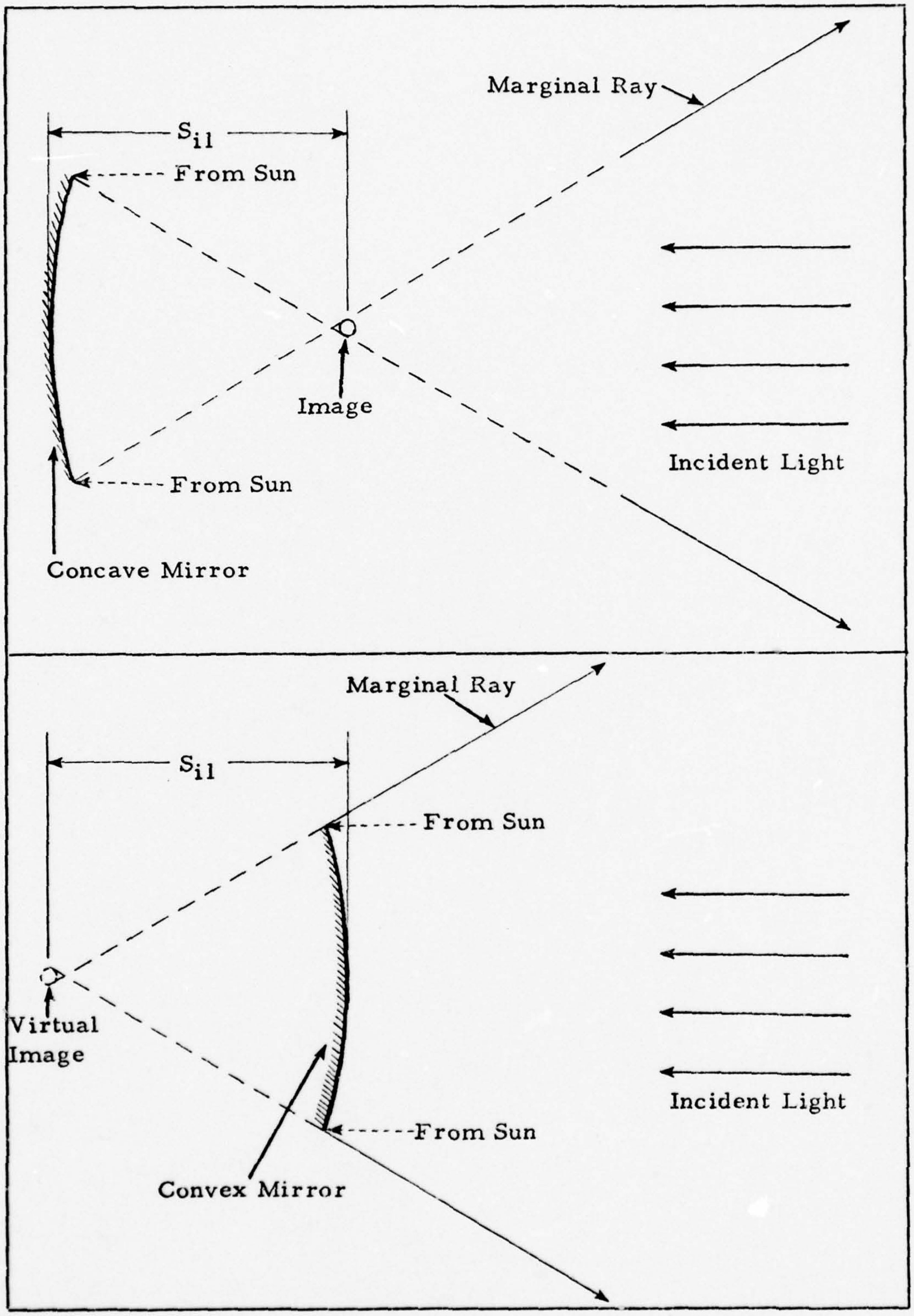

Fig. 2. Marginal Ray Geometry of Two-mirror System

The point that Fig. 2 illustrates is that by the time the cone of light from the primary mirror has traveled far enough that S_{i1} is less than S_{o2}, that cone of light would be as large or larger than the primary mirror itself. Consequently, the secondary mirror would have to be as large if not larger than the primary mirror, in order for the secondary mirror to intercept the entire cone of light coming from the primary mirror. If the entire cone of light from the primary mirror was not intercepted, some of the sunlight would miss the secondary mirror and the total amount of light directed to the Earth would be decreased. Such a reduction in the total amount of sunlight directed to the Earth could negate the advantages to be gained from a reduction in the size of the image.

In particular, the purpose of reducing the size of the Sun's image would be to increase the intensity of that image. But as it turns out, and, as Fig. 2 illustrates, it would not be possible to use a two-mirror system to increase the intensity of the Sun's image without greatly increasing the total reflective area of the mirror system. The following analysis points out the size of this problem.

Image Intensity from a Two-mirror Reflection System

Krafft Ehricke, in an analysis of two-mirror orbiting solar reflectors, investigated the relationship between the intensity of the Sun's image focused on the Earth and the total reflection area required to produce that intensity level. In his analysis, he used a symmetrical two-mirror system in which the reflecting area of the primary and secondary mirrors was identical.

Symmetrical two-mirror systems would be advantageous because, in such systems, the primary and secondary mirrors could easily switch roles after midnight. Such a switch would be required because of the change in the mirror orientations with respect to the Sun that would occur at midnight. This situation is illustrated in Fig. 3. The mirrors would switch roles so that the one furthest from the Sun would always be the primary mirror.

In general, Ehricke found the following relationship to be true of two-mirror systems:

$$A_p A_c \approx 10^{-6} \frac{E_v/I_m}{F(d, S_{i2})} \tag{7}$$

where A_p is the reflecting area of the primary mirror, A_c is the reflecting area of the secondary mirror, E_v is the illuminance on the Earth, I_m is the illuminance of a full moon directly overhead on a clear night = 0.107 lumen/m^2 and $F(d, S_{i2})$ is given by the following equation

$$F(d, S_{i2}) = \frac{\sin^2\left(\frac{180}{2\pi}\frac{d^*}{S_{i2}^*}\right)\sqrt{\dfrac{1 + \sin\left(\frac{180}{2\pi}\frac{d^*}{S_{i2}^*}\right)}{2}}}{\left(\frac{\pi}{4}\gamma^2 r^2_{oo}\right)^2 (d^*)^2 \left[(S_{i2}^* - 1)^2 + \frac{1}{4}(d^*)^2\right]} \tag{8}$$

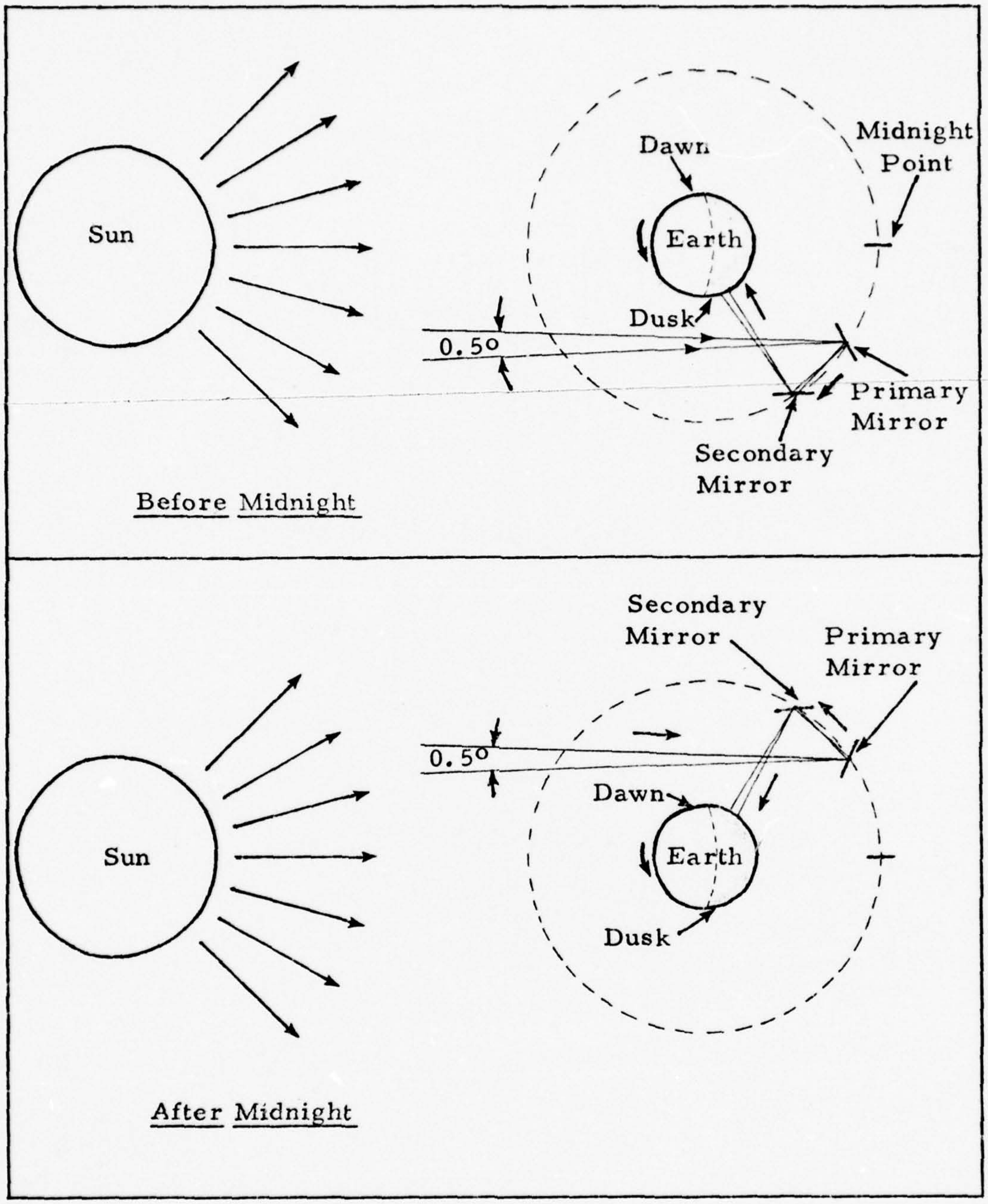

Fig. 3. Two-mirror Reflection Before and After Midnight

where S_{i2} is the distance from the secondary mirror to the Earth's surface, r_{oo} is the radius of the Earth, γ is the angular diameter of the Sun as seen from the primary mirror (in radians), d is the distance between mirrors, and the asterisk means that the variable is expressed in units of the Earth's radius.

To evaluate Eq (7), Ehricke completed a calculation based on a clear night with a symmetrical two-mirror system such that $A_p = A_c$. He found that it would be possible to use a two-mirror system to produce an image intensity of $(10)(I_m)$. However, he concluded:

> . . . a reduction in image area is bought at the expense of a very large increase in overall reflecting area . . . In fact, compared to single reflection, the overall area is larger by a factor of the order of 700 . . .
>
> Thus, while it is possible to reduce the image size by optical means . . . it becomes practical only after larger reflector units and low transportation costs are state of the art [Ref 14:30-32].

Thus, at least at this time, it does not appear advantageous to use a two-mirror system. Although a two-mirror system could be used to reduce the size of the Sun's image on the Earth, the total reflecting area would have to greatly increase.

The Effect of Lower Orbits

Although two-mirror systems do not seem advantageous, the size of the Sun's image on the Earth can be reduced, using a single reflector, by reducing the orbital altitude of the reflector. This technique is advantageous because it would result in an increase in the intensity of the image without requiring an increase in the total reflecting area. Table I lists the area of the Sun's image as focused by a single reflector for various Earth orbits. The orbits are specified by the length of one period of revolution.

Table I
Sun Image Size for Several Earth Orbits

Orbital Period	Area of Image (IA)
2 hrs	192 km^2
3 hrs	1,188 km^2
4 hrs	2,793 km^2
6 hrs	7,337 km^2
8 hrs	13,190 km^2
12 hrs	27,800 km^2
24 hrs	88,000 km^2

(From Ref 13:Fig.12)

The image areas listed in Table I can be easily verified by solving the standard equation for the area of a circle

$$IA = \left(\frac{SI}{2}\right)^2 (\pi) \tag{9}$$

where SI is given by Eq (2) using S_i equal to the orbital altitude of the reflector.

As might be expected, lower orbits have their disadvantages as well. Probably the most obvious disadvantage is that non-geosynchronous orbits would not permit the reflector to remain stationary over one part of the Earth's surface. As a result, the reflector would have to constantly readjust for its changing position with respect to a given ground location. Furthermore, continuous illumination of a particular point on Earth could require the use of several reflectors. For example, in a typical 3 hr orbit, a given reflector would be capable of illuminating a given point on Earth for only about 0.4 hr per orbital revolution (Ref 13:Table 1). Thus, if all of the reflectors were in identical 3 hr orbits, a total of 8 properly spaced reflectors would be needed to provide continuous illumination of a given ground location.

Another disadvantage of lower orbits would be their orientation with respect to the Earth-Sun plane. In geosynchronous orbit it is possible to align the orbit such that the reflector would not pass into the Earth's shadow. This would be done by aligning the orbital plane vertically, or near-vertically, with respect to the Earth-Sun line (Ref 13:28). Such an orbit is known as a sun-synchronous orbit.

In some lower orbits, however, it is not possible to prevent the reflector from periodically passing into the Earth's shadow. The reason this is a disadvantage is that reflectors which periodically pass into the Earth's shadow must be capable of coping with the stresses associated with the effects of alternation between Sun exposure and shadow.

Fortunately, this last disadvantage can be ignored for some low Earth orbits. For example, as will be pointed out in future chapters of this thesis, it is possible to place reflectors in sun-synchronous 3 hr Earth orbits.

III. The Feasibility of Orbiting Solar Reflectors

In 1973, the Air Force Institute of Technology (AFIT) at Wright-Patterson Air Force Base (WPAFB), Ohio, discontinued its space physics graduate program. One of the last research projects conducted by students in the AFIT space physics program was a design proposal for an orbiting solar reflector. The expressed purpose for the AFIT orbiting solar reflector (AOSR) was to "extend the operation of the solar farm concept to include nighttime energy collection." The solar farm was defined as an Earth-based complex "which collects energy from the Sun and converts it to electrical energy [Ref 22:1]."

The students involved in the original study considered the benefits of a solar reflector and the potential materials which could be used to construct the reflector system. However, because of time limitations, they were unable to complete a detailed optical analysis of the AOSR.

Because the students who worked on the AOSR concept were members of AFIT's last space physics class, no further work was done on the AOSR concept. However, considering the current emphasis being placed on alternative energy sources, it was decided that one of the objectives of this thesis would be to complete a detailed optical feasibility study of the AOSR.

The major emphasis of this chapter will be the subject of orbiting solar reflectors, and in particular the AOSR. Following a brief description of the AOSR, the results of the feasibility study will be presented in detail. As a final portion of this chapter, the potential usefulness of orbiting solar reflectors will be discussed.

The AOSR

Figure 4 is a sketch of the AOSR. Designed to be launched in one space shuttle mission, it would consist of three mirrors held at fixed positions with respect to each other. Incident sunlight would be collected by a 330 m diameter concave mirror with a nominal reflectivity of 85%. This mirror, designated the primary mirror, would be made of aluminum coated mylar and would reflect the sunlight to a secondary mirror. The secondary mirror, a 36 m diameter convex mirror, would be positioned such that it and the primary mirror formed a confocal optical system. With such a configuration, it was expected that the sunlight reflected from the secondary mirror would be collimated. This collimated light would then be redirected to the desired location on Earth by a tertiary mirror 44 m in diameter. Both the secondary and tertiary mirrors would be made of highly polished Aluminum honeycomb coated with a 7 micron thick film of Aluminum Oxide (Ref 22).

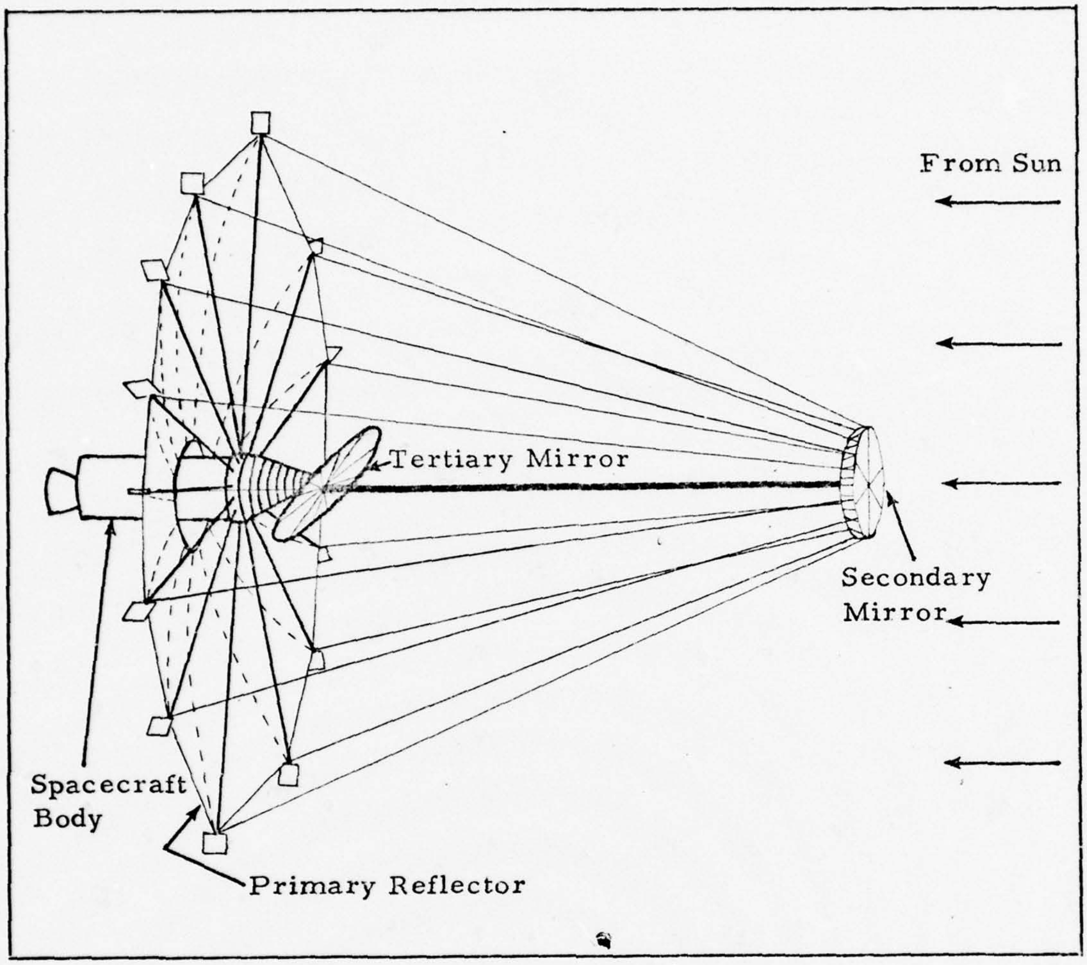

Fig. 4. The AOSR (based on Ref 22:4)

Feasibility of AOSR

The initial AOSR design study was directed primarily at space physics problems. Thus, as a first approximation, the Sun was assumed to be a very distant source and incident solar radiation was assumed collimated. Figure 5a shows how collimated light, incident on an arbitrary point of the primary mirror, would be imaged by the AOSR.

The current AOSR feasibility study emphasized the optical problem. The purpose of this recent study was to determine the impact on AOSR performance if the Sun was treated as an object to be imaged by the AOSR optical system rather than as a collimated light source.

As pointed out in Chapter II, the diameter of the Sun's disk, as seen from the Earth, is approximately 0.5 degrees. Thus, the cone of light from the Sun, incident on each point of the AOSR primary mirror, would have a full angle of about 0.5 degrees. Figure 5b illustrates the way this incident cone of sunlight, incident on an arbitrary point of the primary mirror, would be imaged by the AOSR.

It should be noted in Fig. 5b that the cone of sunlight leaving a given point of the tertiary mirror would have a greater full angle than the incident cone of light from the Sun. It should also be noted that the path taken by the cone of light in Fig. 5b includes the path of the collimated light shown in Fig. 5a. The path of the collimated ray has been superimposed on Fig. 5b using a dashed line.

Figures 6a and 6b are included to show the effect of the AOSR system when the entire system is illuminated. Again, the path followed by the collimated light in Fig. 6a is contained within the cone of light of Fig. 6b, as indicated by dashed lines.

Figure 6b illustrates the approximate path which actual sunlight would follow through the AOSR system. However, Fig. 6b is just a rough sketch and does not show what happens to the rays which miss

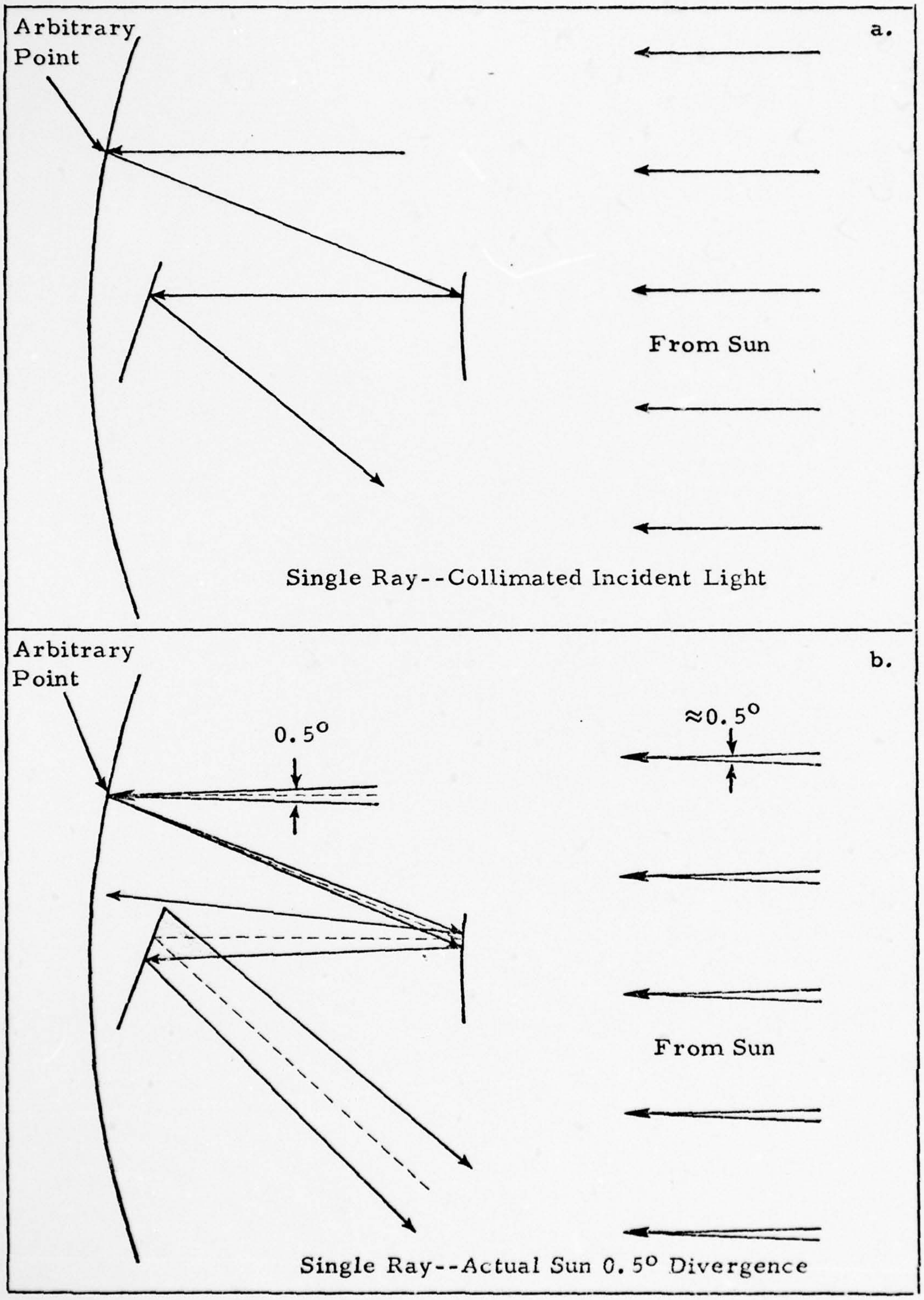

Fig. 5. AOSR Performance with a Single Ray of Both Collimated and Uncollimated Incident Sunlight

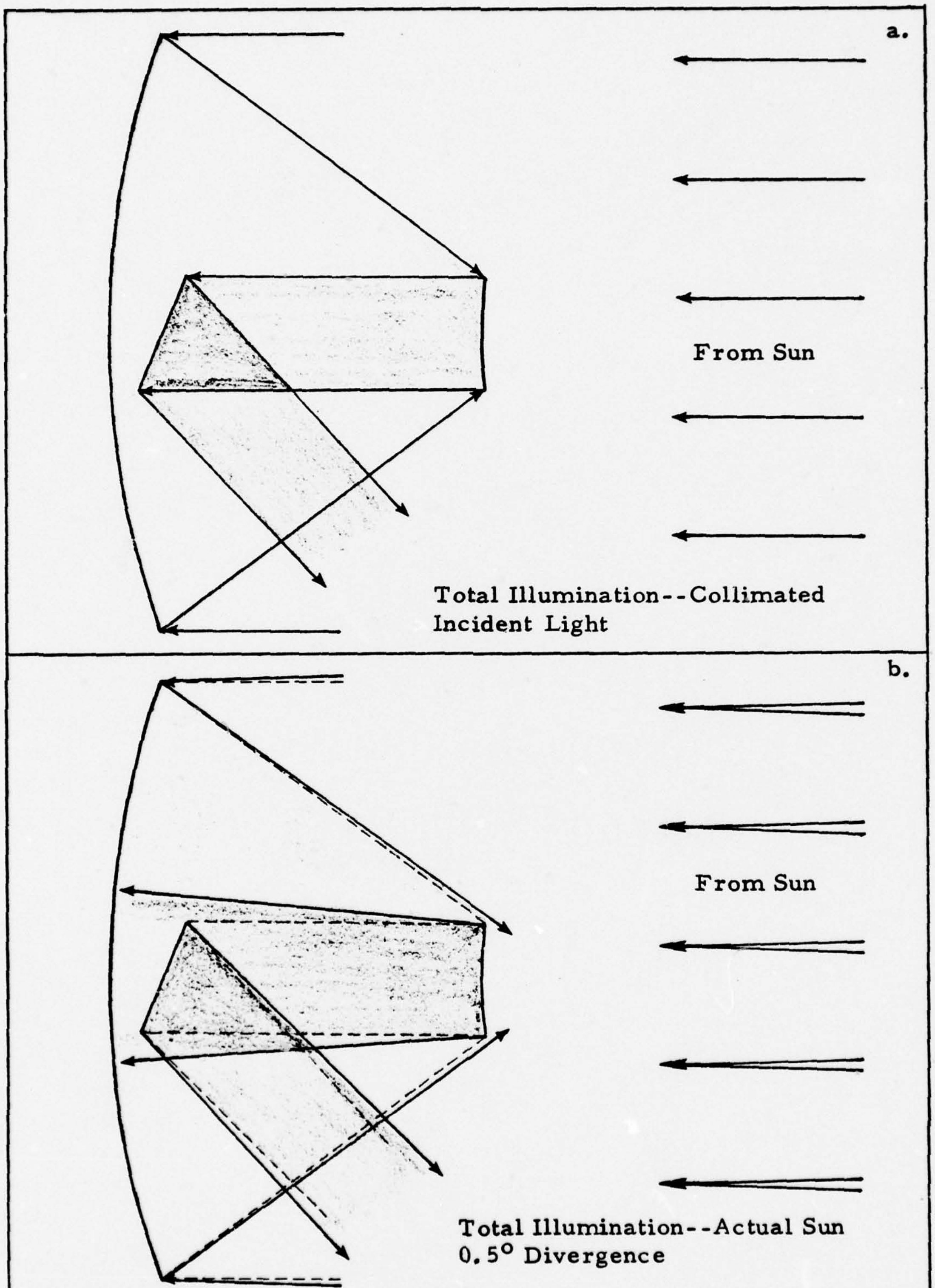

Fig. 6. AOSR Performance for Total Illumination by Both Collimated and Uncollimated Incident Sunlight

the tertiary mirror. The marginal rays, which form the extreme outer edges of the cone of sunlight in Fig. 6b and actually miss the tertiary mirror, were traced more accurately through the AOSR by using matrix techniques. In matrix representation, a typical ray would be written as follows:

$$\begin{bmatrix} n\,\alpha \\ y \end{bmatrix} \tag{10}$$

where n is the index of refraction of the material through which the ray would travel, α is the small angle approximation for tan α where tan α is the slope of the ray as measured in the direction of the ray's travel, and y is the distance between a point on the ray and the optical axis of the lens system (Ref 24:172). This geometry is illustrated in Fig. 7. In applying matrix methods to the AOSR optical system, n was assumed to be 1.0 because the space environment is essentially a hard vacuum. Details of the calculation are contained in Appendix A.

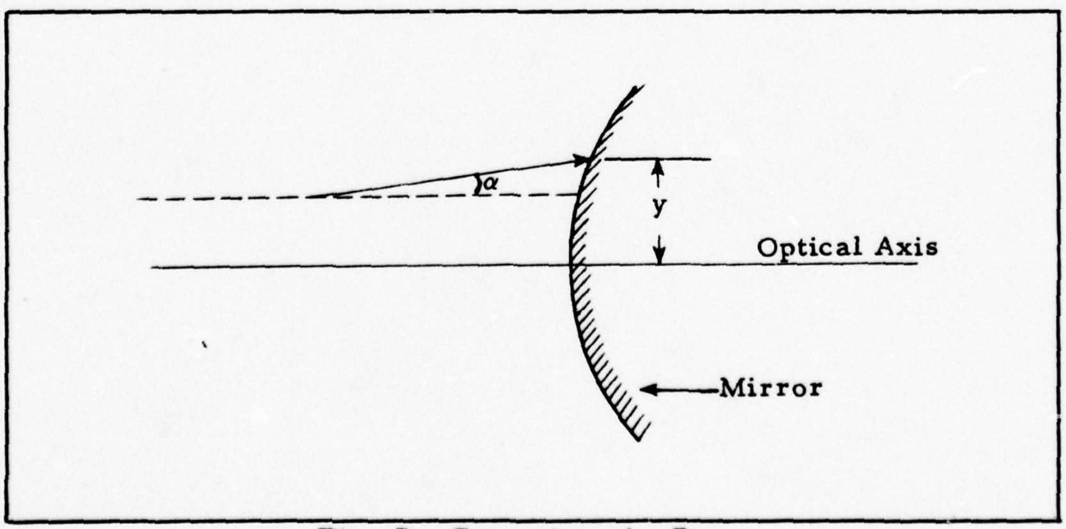

Fig. 7. Geometry of a Ray

The results of the ray trace described in Appendix A are shown in Fig. 8. As shown in both Figs. 6b and 8, the tertiary mirror, while redirecting the sunlight, would also act as an aperture. Only a portion of the sunlight reflected by the secondary mirror would be reflected by the tertiary mirror. The rest would miss the tertiary mirror and be lost.

The light reflected to the Earth by the AOSR would continue to diverge. The minimum diameter of the Sun's image on the Earth is given by

$$SI = (d_G)(2 \tan \theta_d) \tag{11}$$

where d_G is the distance of the AOSR from the Earth's surface (approximately 3.59×10^7 m for geosynchronous orbit), θ_d is the half angle of the cone of light leaving a given point of the tertiary mirror, and the comparatively small diameter of the tertiary mirror has been neglected. The geometry of this situation is shown in Fig. 9.

The results of the ray trace described in Appendix A indicate that

$$\theta_d = 2.44^o \tag{12}$$

Substituting for d_G and θ_d in Eq (11) gives the diameter of the Sun's image on the Earth, produced by the AOSR system, as

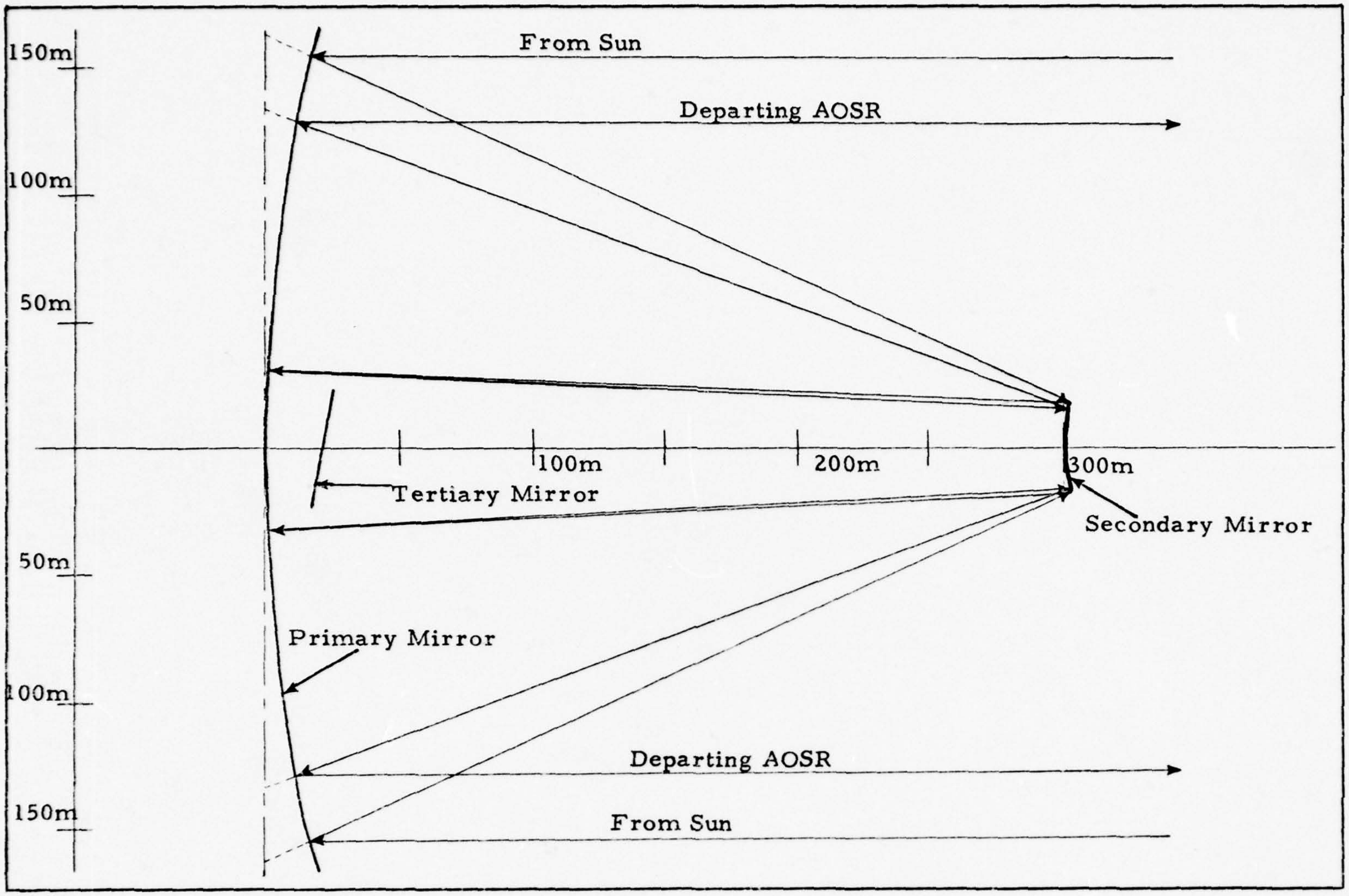

Fig. 8. Ray Trace for the AOSR

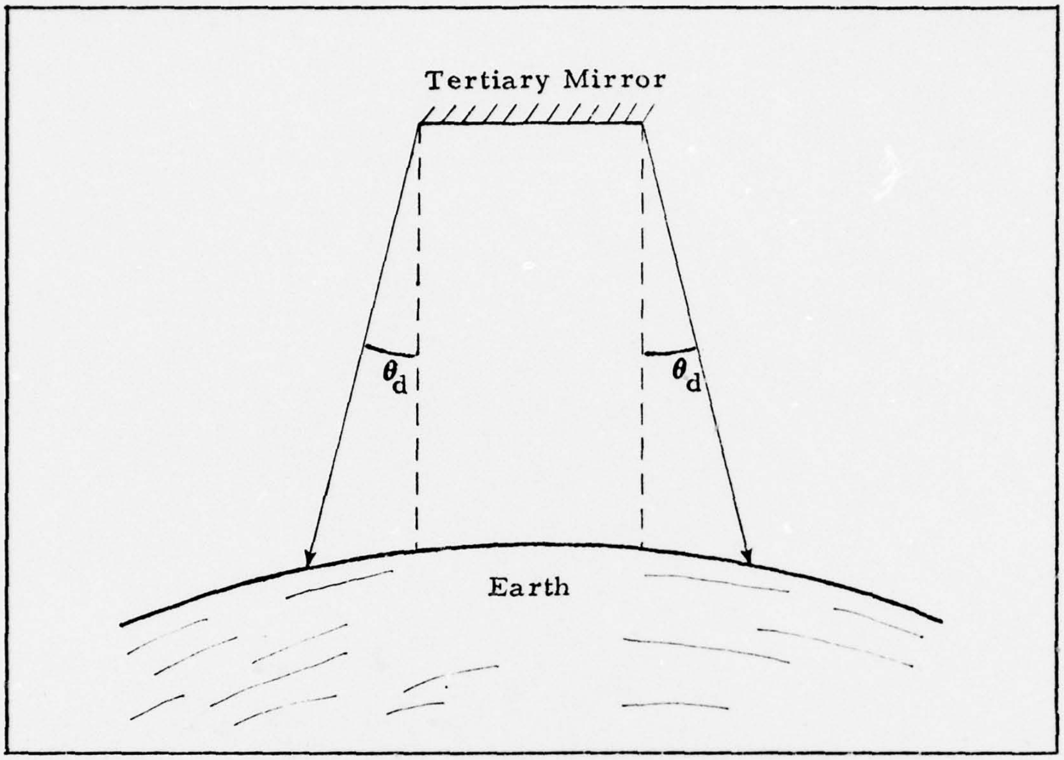

Fig. 9. Geometry Used to Calculate Size of Sun's Image on the Earth

$$SI = 3.06 \times 10^6 \text{ m} \tag{13}$$

$$\approx 1900 \text{ miles} \tag{14}$$

If all losses are neglected including those from diffraction, atmospheric scattering, and impaired mirror reflectivity, then it is possible to determine a first order approximation for the intensity of the sunlight beam received on Earth from the AOSR. The incident solar energy at one astronomical unit from the Sun (AU) is given by the solar constant

$$H_o = 1390 \text{ W/m}^2 \tag{15}$$

(Ref 42:16-3). Ignoring shadowing effects from the secondary and tertiary mirrors, the total solar power intercepted by the primary mirror would be

$$P_i = (1390 \text{ W/m}^2)(165 \text{ m})^2(\pi) \tag{16}$$

$$= 1.19 \times 10^8 \text{ Watts} \tag{17}$$

of reflected sunlight. It then follows that the irradiance of the reflected sunlight as measured on the Earth's surface would be

$$I_e = \frac{(1.19 \times 10^8 \text{ watts})}{(1.53 \times 10^6 \text{ m})^2(\pi)} \tag{18}$$

$$= 1.62 \times 10^{-5} \text{ watts/m}^2 \tag{19}$$

where it has been assumed that there would be no losses and that the 1.19×10^8 watts of reflected sunlight would be equally distributed across the sunlight spot on the Earth. Although an intensity of 1.62×10^{-5} watts/m^2 is weak, the actual sunlight intensity would be even weaker because of losses which would occur. These losses, from the following causes, could exceed 80%:

1. Diffraction losses
2. Reflectivity of the primary mirror at 85% (Ref 45)
3. Reflectivity of the secondary and tertiary mirrors at 93% (Ref 4:18).

4. Losses due, on the average, to atmospheric scattering from cloud surfaces, clear atmosphere, Earth/air interface, and particles such as dust and ice crystals suspended in the atmosphere. These losses could be as large as 40% (Ref 42:3-1).

5. Losses as a result of light missing the secondary mirror could exceed 13%.

6. Losses as a result of light missing the tertiary mirror could exceed 50%.

7. Losses as a result of shadowing by the secondary and tertiary mirrors could exceed 1.5%.

When this 80% loss is included as a factor in Eq (18), the calculated irradiance on the Earth becomes

$$I_e = 3.24 \times 10^{-6} \text{ watts/m}^2 \tag{20}$$

The same calculations as above could be done with illuminance as well as with irradiance. The illuminance of the Sun outside the Earth's atmosphere (at 1 AU) is

$$E_v = 1.37 \times 10^5 \text{ lumens/m}^2 \tag{21}$$

Details of the derivation of Eq (21) are contained in Appendix B. The luminous flux intercepted by the primary mirror would be

$$\phi_v = (1.37 \times 10^5 \text{ lumens/m}^2)(165 \text{ m})^2(\pi) \tag{22}$$

$$= 1.17 \times 10^{10} \text{ lumens} \tag{23}$$

Based on a calculation similar to Eq (18), the illuminance of the reflected sunlight as measured on a clear night on the Earth's surface would be

$$E_v = 3.98 \times 10^{-4} \text{ lumens/m}^2 \tag{24}$$

In determining Eq (24), a 75% loss was assumed rather than 80% in order to account for the clear night. On a clear night, the atmospheric losses would amount to roughly 21% (Ref 13:10). For purposes of comparison, the illuminance of a full moon on a clear night on the Earth's surface, is 0.1076 lumens/m^2 (Ref 13:13). Thus, the intensity on the Earth's surface of the sunlight reflected by the AOSR would be roughly 0.37% of the intensity of a full moon.

In summary, the feasibility study of the AOSR demonstrated that because light incident from the Sun is not collimated, the AOSR concept is not feasible. Unless modifications were made in accordance with the discussion of Chapter II of this thesis, beneficial sunlight intensities could not be obtained.

Applicability of Orbiting Solar Reflectors

The multiple-mirror AOSR was not feasible for concentration of sunlight on the Earth's surface because of the large Earth area illuminated by AOSR reflected sunlight. To determine if single-mirror systems have potential applicability, Chapter IV contrasts the use of single-mirror orbiting solar reflectors with other methods of using solar energy to generate electricity for terrestrial use and this section evaluates the use of reflectors for in space power generation.

In the discussion in Chapter II on solar optics, it was pointed out that although two reflectors could be used to reduce the size of the Sun's image on the Earth's surface, this method would require very large reflectors. However, a single mirror could be used to produce a small spot in space, as long as the image distance could be kept small. For example, the primary mirror proposed for the AOSR would have an image spot size of roughly 3 m. However, the image would be formed at a distance of just over 336 m from the primary mirror. Furthermore, since image size is determined by the equation

$$M_I = -\frac{S_i}{S_o} \tag{3}$$

it follows that images formed even closer to the primary mirror would be even smaller. Depending upon the image size required, it would be possible to alter the primary mirror focal length appropriately to produce that particular spot size. Unfortunately, small spot sizes also require short focal lengths because when $S_o \gg S_i$

$$\frac{1}{f} = \frac{1}{S_i} + \frac{1}{S_o} \approx \frac{1}{S_i} \tag{1}$$

A short focal length becomes a problem because parabolas with short focal lengths have smaller cross sectional areas than parabolas with long focal lengths. Smaller cross sectional area means that less sunlight would be intercepted by the mirror, and, therefore, less sunlight reflected. Thus, it would be necessary to weigh the advantages of a small spot size against the disadvantages of a reduction in the amount of sunlight reflected.

One example of the use of an orbiting solar reflector is work currently being done by the Space Division of Rockwell International on a solar thermal concept. A large space reflector would be used to concentrate sunlight onto a heat absorbing material. Under such conditions, heat would be transferred by flowing liquid helium/xenon or liquid sodium/potassium, and power would be generated by either a Brayton or a Rankine generating system (Ref 39:10-12). In this thesis, this system will be known as a Solar Thermal Power System (STPS).

It is conceivable that at some time in the future, there may be a use for systems like the STPS. Today, most space systems are powered by banks of photovoltaic (solar) cells. However, photovoltaic cells offer maximum theoretical efficiencies of approximately 25%, while thermal power cycles can reach actual efficiencies over 75% (Ref 39:10). Of course, these high efficiencies also require large pieces of hardware and, therefore, large expenditures of energy and money for orbital insertion. Yet, because of their high efficiencies, it is possible that they may prove to be advantageous when large quantities of power are required. In particular, they may prove useful in serving as central power stations for large orbiting space stations. As an example, this might be the case if and when large scale manufacturing in space becomes a reality.

In a developed space industrial complex, an STPS could be centrally located around several industrial and space station complexes. Large quantities of electricity could be generated by the STPS and either transmitted via conventional power lines, or via microwaves to user complexes surrounding it.

It is beyond the scope of this thesis to consider, in detail, the design characteristics and potential of an STPS as a reliable and economical source of space power. However, Appendix C contains several possible mirror designs which may be applicable to an STPS should further work be done in this area.

IV. A Resource Evaluation of Four Solar Electric Generating Systems

One way of evaluating alternate systems for the production of electricity from solar energy is to consider the resources that would be required by each system. This chapter of the thesis will be devoted to an outline of the resources required by a terrestrial solar thermal (TST) system, a terrestrial photovoltaic (TP) system, an orbiting solar reflector (OSR) system, and a satellite solar power (SSP) system.

This chapter is organized as follows. Included first is a brief discussion of the ground rules and limitations of this analysis. Then the major components of each system are briefly described. Following this description, each of the four competing systems is evaluated relative to their ability to produce electricity and their consumption of the following resources:

1. The mass and type of materials required for construction, operation, and maintenance.
2. The land area required for both the generating plant and the transmission lines from the plant to major user areas.
3. The volume of water required during construction, operation, and maintenance.
4. The manpower required for construction, operation, and maintenance.
5. The energy required to mine and refine the materials needed in item one above.

6. The cost of development, design, testing, evaluation, construction, operation, and maintenance.

As a final part of this chapter, Table XIV is included as a summary of the results of this analysis.

Ground Rules

Ground rules were established in order to simplify the analysis. It was assumed that each system would be operational in the year 2000, and that following construction, would operate for 30 years. In addition, it was assumed that no part of a plant would be operational while the rest of the plant was still being constructed. These assumptions removed the possibility that some plant parts would have to survive for over 30 years.

Of course, since, in reality, construction time would be long and partial plant operation would be likely, some parts of the generating system would actually operate for more than 30 years. However, the additional operational and maintenance expenses of operation for more than 30 years may be offset by the revenue obtained from partial operation. To determine this relationship precisely would require an analysis of detailed construction plans which are not available. Therefore, partial operation and extended lifetime were not considered.

Limitations

The reason why detailed construction plans are not available is that none of the systems evaluated has ever been constructed. Of the four systems considered, the two terrestrial proposals can most nearly be termed "state-of-the-art" proposals. In fact, a prototype solar thermal plant is currently under construction in Barstow, California (Ref 25:94). The information presented in this section of the thesis should be understood as a best estimate as of this stage in the development of each system. As development continues, changes are sure to be made. For example, in the aerospace industry, a system typically becomes more massive during development. This is termed weight growth. According to a Johnson Space Center report,

> For all aerospace vehicles, the usual range of weight growth is between 5 and 50 percent, represented by low-risk design aircraft and complex, advanced spacecraft, respectively [Ref 34:IV-A-5-1].

The goal of this analysis was to make a fair comparison between the various systems. Ideally, this means that each system should be evaluated using identical parameters. This is especially true when competing systems use identical subsystems. This was done whenever possible. However, in order to do this, it was sometimes necessary to alter the design of some of the proposed systems.

Unfortunately, alteration was not always in the best interest of a given system. In those cases where it was felt that alteration would significantly affect the overall system's performance, no alteration was made. To avoid confusion in this thesis, alterations made are identified. In addition, in those cases where alteration was not considered in the best interest of the particular system, differences between systems are identified.

An example where alteration was both possible and impossible is the area of solar cells. Both the TP and SSP systems use concentration equipment while the OSR system, as currently proposed, does not. In this regard, a recent study indicates that the mass of silicon solar cells required to produce a given amount of electricity decreases as the concentration ratio increases, up to a limiting concentration ratio of 2.4:1 (Ref 1:35). This concentration ratio specifies the ratio of the total sunlight eventually reaching the solar cells to the sunlight received directly by the solar cells. Although this finding pertained to solar cells when used in Earth orbit, it does point out the benefit of concentration. In light of the high cost of silicon solar cells, it was decided to evaluate the OSR system under the assumption that it had a concentration ratio identical to that of the TP and SSP systems.

On the other hand, solar cell alteration was impossible with respect to cell thickness. As explained below, the SSP solar cell thickness would be different from the thickness of the TP or OSR system solar cell because of cost and weight.

There are currently several approaches being investigated to drastically reduce the cost of manufacturing silicon solar cells. The current technique uses the following five step process:

1. Quartzite pebbles are reduced to metallurgical grade silicon.

2. Metallurgical grade silicon is refined to form semiconductor grade material.

3. The semiconductor grade material is processed into single crystal ingots from which silicon wafers are cut.

4. Silicon wafers are processed into silicon cells.

5. Silicon cells are interconnected [Ref 28:7].

The major disadvantage of today's technique is that it is very wasteful. As much as 92% (by volume) of the silicon prepared in step three for cutting will be lost as a result of sawing, lapping, polishing, dicing, and breakage (Ref 34:IV.B.1.a.27). However, improvements are being made and it is conceivable that the 1985 Energy Research Development Administration (ERDA) cost goal, to reduce solar array costs from $15,000 to $500 per peak kilowatt[1], will be achieved by improving this ingot cutting technique (Ref 17:1). If this is the case, it is also likely that the thinnest silicon cell obtainable using cutting techniques will be 10 mils (Ref 8:4-36).

A 10 mil thick silicon cell is probably adequate for terrestrial uses, especially if the ERDA goals are achieved. However, even a 10 mil thickness would place a large weight burden on the SSP system. Proponents of the SSP concept contend that a 4 mil thickness would be required in order for space operations to be competitive.

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[1]Peak kilowatt is the maximum electrical power output of a solar cell, measured at the Earth's surface, for normal solar incidence.

Ideally, resource analyses would be most straightforward if one cell thickness was used in all systems. However it was decided that a common ground could not be achieved. A 4 mil cell might not have the strength to withstand the environmental hazards of earth such as hard rains and hail. Additionally, development of 4 mil cells could conceivably be very expensive since it probably would require development of some form of growth technique (Ref 8:4-36). Yet, 10 mil cells are unacceptable to the SSP. Therefore, it was decided that terrestrial cells would be 10 mils thick while the SSP solar cells would be 4 mils thick.

Terrestrial Solar Thermal System

Information used in the analysis of the TST system was extracted primarily from a study by the Jet Propulsion Laboratory, California Institute of Technology (JPL) (Ref 8). The type of TST generating plant selected for this analysis was one similar to a TST plant currently being constructed at Barstow, California (Ref 8:4-10). As designed, it would be a central receiver type of plant with six hour thermal storage. Electricity produced from thermal storage would be only 70% of rated capacity because of conversion and storage inefficiencies (Ref 8:6-13).

A central receiver plant uses Sun-tracking mirrors to concentrate the Sun's energy on receivers located on top of a central tower. The heat of the sunlight is used to heat a liquid contained in the receiver. The steam produced by this process is then used in a conventional steam Rankine generating plant (Ref 8:4-22). Figure 10 is a sketch of a TST plant.

Although other solar thermal designs have been proposed, the central receiver design was chosen as the representative of solar thermal generating systems because National Science Foundation supported studies have found that the central receiver design is 20% to 50% cheaper than its nearest competitor, the parabolic trough or dish (Ref 8:4-11).

The mirrors, known as heliostats, are the major components of a solar thermal plant. The heliostats considered in this analysis were based on an early preliminary design by the Honeywell Corporation. Excluding concrete in the foundation, their mass would be 51.25 kg/m^2. The major components of the heliostats would be glass and metal. Two additional designs have been proposed but because they are considered more speculative, were not used in the analysis. The most speculative design would use an aluminized mylar reflector in a clear tedlar dome and would have a mass of 19.5 kg/m^2. Use of this design would reduce the material resources required to build the central receiver plant by approximately 60% (Ref 8:6-15).

For this resource analysis it was assumed that roughly 30% of the land required for a solar thermal plant would be covered with heliostats and that the average plant efficiency would be 17% (Ref 8:6-13). Every five weeks the heliostat surfaces would be cleaned, using 0.75 gal/m^2 of water per cleaning (Ref 8:6-22). However, to

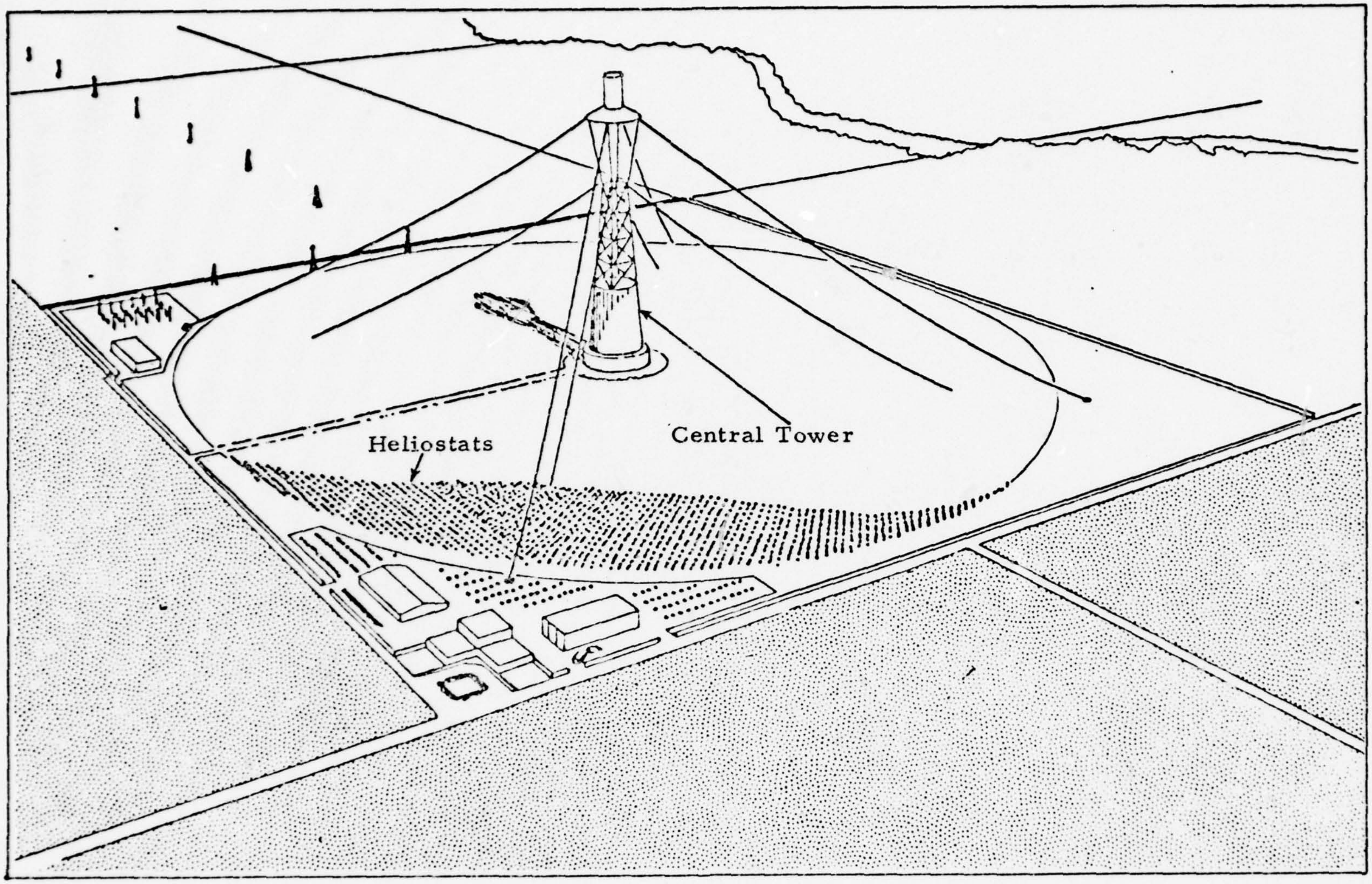

Fig. 10. Terrestrial Solar Thermal (TST) System (Ref 30:Fig. 1a)

conserve on water, dry cooling towers would be used (Ref 8:6-13). In dry cooling towers, fan-forced air cools the liquid from the turbine generators as in an automobile radiator. Although dry cooling towers cause a reduction in the plant efficiency, especially in hot weather, they would probably be necessary because efficient solar thermal plants would require large amounts of sunshine in arid climatic regions where water is scarce.

The need for large amounts of sunshine poses another problem. Ideal sunlight conditions are typically found in the southwestern portion of the United States, while the population is concentrated in the eastern states. Therefore, implementation of the solar thermal system would require long transmission distances.

Terrestrial Photovoltaic System

The terrestrial photovoltaic system selected for evaluation in this thesis is a system using silicon solar cells supplemented with battery storage (Ref 8:6-13). Primary source for data relative to this system was a study by JPL (Ref 8). The silicon cells used in the system selected for this analysis would be 10 mils thick and would have an efficiency of 13% at air mass 1[2] and at a cell temperature of 28°C. The glass cover plates protecting the silicon cells would be 3 mils thick (Ref 8:4-36).

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[2]Air mass 1 is the mass of air in a vertical path through the atmosphere, above a sea level point on the Earth's surface.

The silicon cells would be secured to tilted surfaces which would be rotated twice a year. Tilted surfaces would be advantageous in order to compensate for the change in the inclination of the Sun throughout the year. Since solar cell efficiency is greatly affected by the angle of incidence of the sunlight, the net efficiency of a generating plant can be enhanced by periodically reorienting the solar cells. However, the advantages to be gained by this reorientation would be partially offset by the cost of performing the reorientation. Thus, for this analysis, reorientation twice a year was selected (Ref 8:1-3).

The TP system would use a concentration ratio of 2:1. This would be obtained by using non-tracking asymmetric v-trough concentrators (Ref 8:4-22). Further details of the design used for this analysis are unavailable because they are contained in an unpublished JPL internal document. However, Fig. 11 contains a sketch of a symmetrical v-trough concentrator. Theoretically when the concentration ratio is 2:1, the area enclosed by points a, b, c, and d of Fig. 11, is twice the area of the silicon solar cells (Ref 37:107). In practice, however, because the mirrors would not be perfect reflectors, the geometric ratio of the area abcd, to the area of the silicon cells, would have to be greater than 2:1 in order to achieve a concentration ratio of 2:1. For example, if the reflectivity of the mirrors is 85%, then the geometric ratio would need to be roughly 2.15 (Ref 34:IV-B-1b-8). A mirror reflectivity of 85% was assumed for this analysis. Loss of reflectivity was assumed to be due to

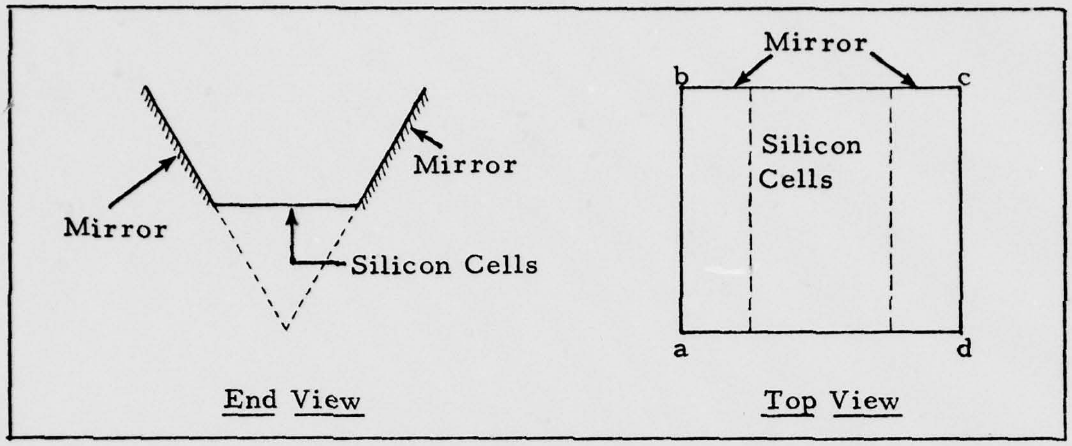

Fig. 11. Symmetrical V-trough Concentrator

construction imperfection and the cumulative effects of dust accumulation between washings. Figure 12 is a sketch of a TP plant.

Orbiting Solar Reflector System

The orbiting solar reflector system selected for analysis is the system currently proposed by Ehricke. This analysis is based primarily upon information supplied by Ehricke (References 13, 14, and 15).

The system, as proposed, would have both a ground and a space component. The ground component would be similar to the TP system with the exception that it would not have the battery storage facilities. In place of battery storage, the system would operate continuously by supplementing daytime sunlight with sunlight reflected from orbiting reflectors. Although it might be necessary to use some battery storage in order to levelize the electrical output of the plant, this factor has not been considered in the analysis. A sketch of the OSR system

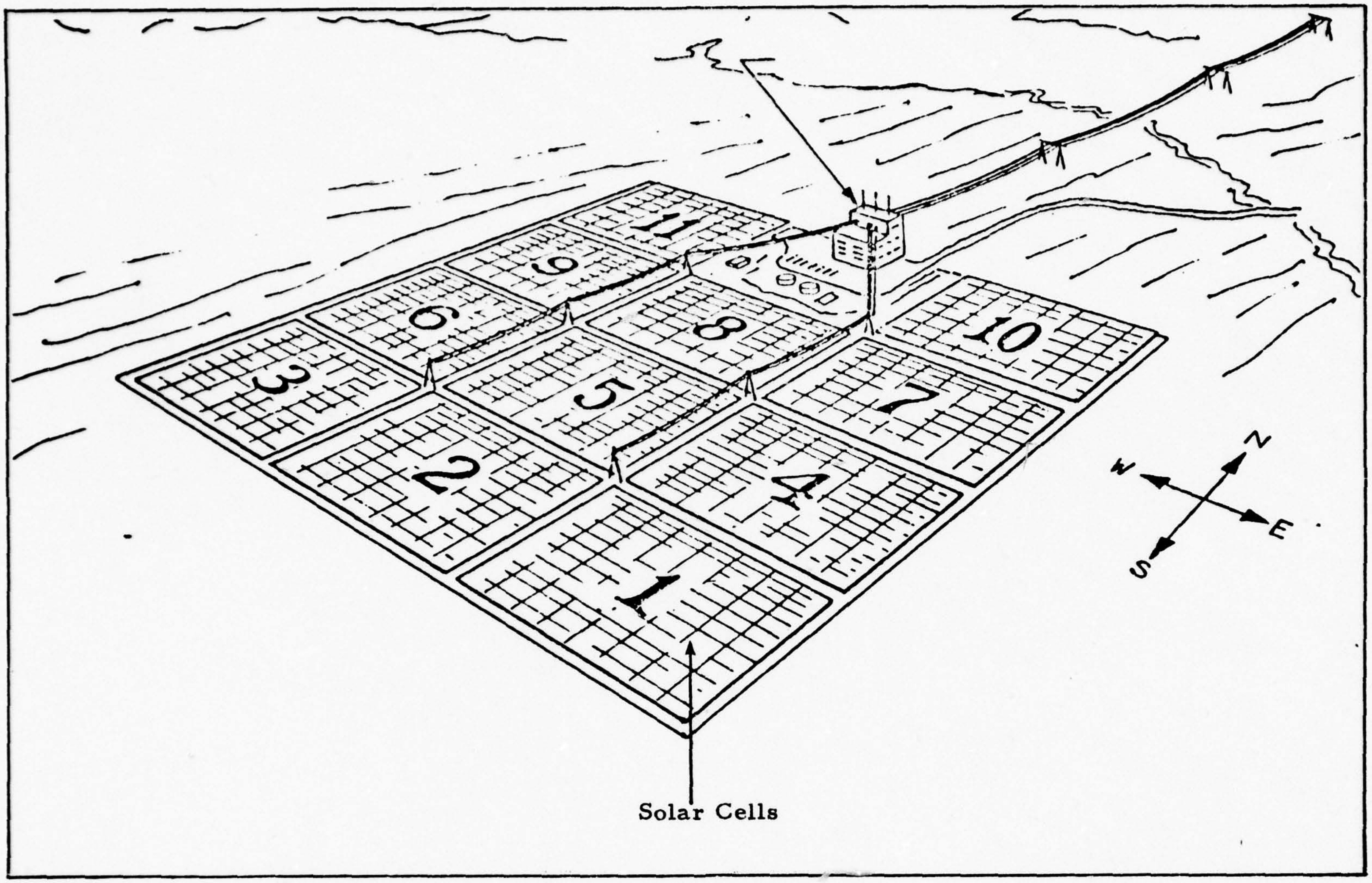

Fig. 12. Terrestrial Photovoltaic (TP) System (Ref 30:Fig. 1b)

is contained in Fig. 13.

The system evaluated, known as Soletta II, would consist of 1320 individual reflectors placed in three-hour Sun-synchronous orbits. Each reflector would have an area of 8.73 km^2, a variable focal length, and a mass of 50 to 150 tons/km^2 (Ref 13:26, 40, 43). For this analysis, a mass of 100 tons/km^2 was used. When focused on the Earth's surface, the image of the Sun produced by each reflector would have a diameter of 38.9 km (see Table I). The orbital altitude of the reflectors would be 4184 km.

As currently proposed, the OSR system would have a rated generating capacity of 74.2 GWe. This projection is based on the assumption that 776.6 km^2 of silicon solar cells, with a before-concentration efficiency of 13% and a concentration ratio of 2:1, would receive and process sunlight at an annual rate of 5 x 10^9 kwhr/km^2-year. Of this incident energy, 2.15 x 10^9 kwhr/km^2-year would be provided directly from the Sun and 2.85 x 10^9 kwhr/km^2-year would be provided by sunlight reflected from the orbiting reflectors (Ref 13:51).

A total of five different orbits (Ref 13:43) would be used in order to permit reflectors to simultaneously illuminate the terrestrial solar cells. Because each reflector would only be in position to illuminate the ground system for approximately 0.4 hours per orbital period (Ref 13:23), shadows from a given reflector would move quickly across the ground complex. Thus, multiple orbits would be used to provide a more even illumination of the ground facilities. Multiple orbits would

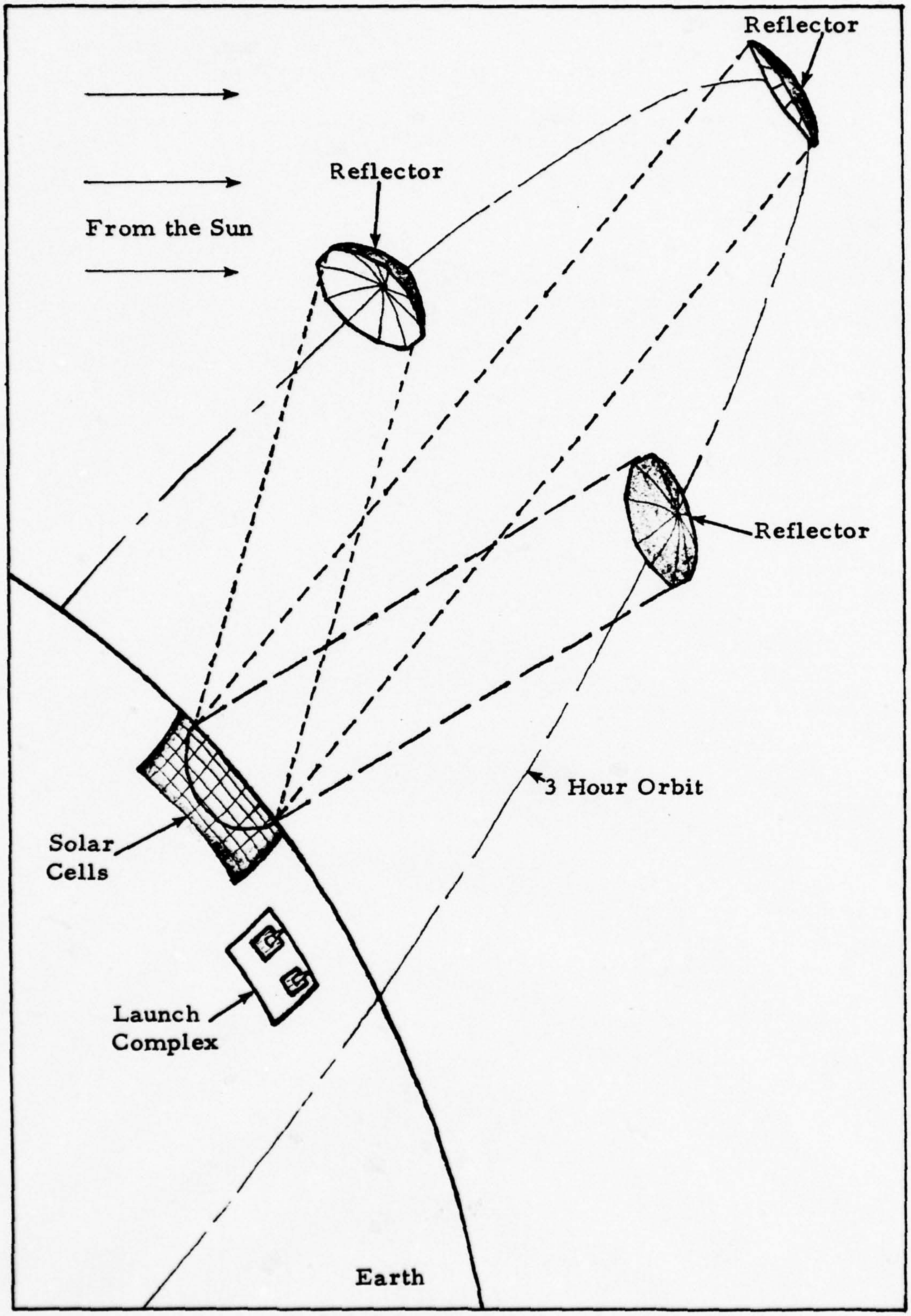

Fig. 13. Orbiting Solar Reflector (OSR) System

also make it possible to position reflectors such that one third of those illuminating the ground facility would be ready to move out of position, one third would be in an ideal position, and one third would be coming into position (Ref 13:16-19). This would help to insure a constant production level of electricity.

Each reflector would be made of several smaller reflector elements. The smaller reflector elements would be made of a sodium-coated kapton film supported by a truss-like structure made of aluminum-coated graphite epoxy. The reflective surface of each element would be 1.3×10^{-2} mm thick (Ref 13:25) and cover an area of 0.2 km^2 (Ref 13:45). Sodium was selected because it would be light, less costly, and have a reflectivity higher than aluminum. However, this sodium coating would have to be applied in space in order to prevent the reflectivity from being destroyed by oxidation (Ref 14:35).

Satellite Solar Power System

First proposed in 1968 by Peter Glaser, the SSP system would also use silicon solar cells to generate electricity. Large blankets of silicon solar cells would be placed in geosynchronous orbit. Sunlight, intercepted by the solar cells, would be converted into electricity and then into microwave radiation. A beam of microwaves would then be transmitted to an Earth receiving station located on low-value land or offshore. As a final step in the process, the microwaves would be reconverted to electricity for commercial distribution at the ground station. The ground station is commonly referred to as the rectenna. Fig. 14 contains a sketch of the SSP system.

Most of the information relative to the SSP system design was obtained from preliminary work done by the NASA, Johnson and Marshall Space Centers. Data relative to the ground portion of the SSP was taken from a study by JPL.

There is one significant difference between the data from NASA and that from other sources. NASA has found that the most efficient and safest way to generate 10 GWe with the SSP system would be to build two space components, each capable of supplying enough power to the ground system to produce 5 GWe (Ref 32:5-4). However, instead of building two ground systems, the two orbiting systems would direct their microwave beams at the same ground system. This NASA concept was adopted for this analysis.

Glaser expects to use silicon solar cells with an efficiency of 18% (Ref 19:574). However, studies by JPL, the Johnson Space Center, and the Marshall Space Center all have used lower efficiencies in their evaluations of the SSP system (Ref 34, 32, 8). In fact, even though their theoretical efficiency is 22% (Ref 1:8), there is some question as to whether 18% efficiency will ever be obtained from silicon solar cells on a mass production scale. Conventional silicon solar cells in production volumes currently have an efficiency between 10% and 12% (Ref 34:IV.B.1.a.4,9).

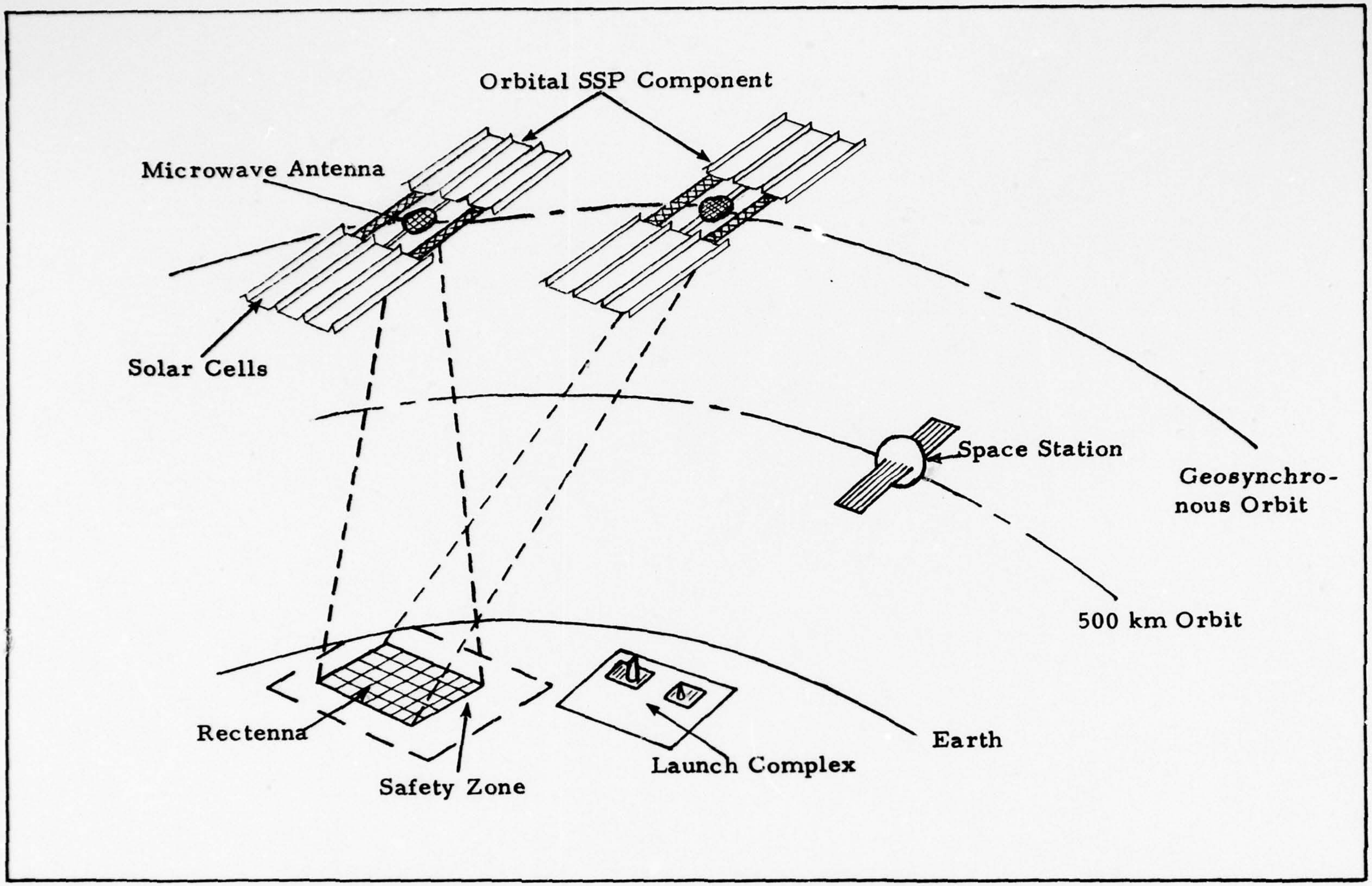

Fig. 14. Satellite Solar Power (SSP) System (Based on Ref 8:4-34)

In order to be consistent with the other designs evaluated, the SSP solar cell efficiency is taken to be 13% at air mass zero and 30°C. The solar cells would be used in conjunction with v-trough type concentrators at a concentration ratio of 2:1, would be 4 mils thick, and have a plastic cover 1 mil thick for radiation and micrometeroid protection. The mirrors would be made of 0.25 mil thick kapton, coated with aluminum, 0.1 mil thick. Since the mirrors would have a reflectivity of approximately 85%, the geometric ratio would be roughly 2.15:1. It is anticipated that the operating temperature of the silicon solar cells would be 100°C. Since a silicon cell of 16% efficiency at 30°C would have an efficiency of 10.3% at 100°C (Ref 34:IV.B.1b.6,7,8), it was determined by analogy, that a 13% efficient solar cell at 30°C would have an efficiency of 8.37% at 100°C. Assuming that 8.37% is the operating efficiency of the silicon solar cells that would be used in the SSP system, the over-all system efficiency would be 4.8% (Ref 34:IV. A.1.1). However, this low efficiency would be offset by the fact that the amount of solar energy available in synchronous orbit is 6 times greater than that available at the best location on Earth and approximately 15 times greater when compared to a United States location with average weather conditions (Ref 32:1-2).

Resource Evaluation Introduction

The remainder of this chapter will be devoted to an evaluation of each of the four competing systems relative to their ability to produce electricity and their consumption of the resources listed at the beginning of this chapter. Each system will be assumed operational in the year 2000. The OSR system evaluation is based on a rated capacity of 74.2 GWe which represents the smallest scale on which it could be built. The three other systems would have rated capacities of 10 GWe, although the terrestrial systems could be built on still smaller scales.

Most of the information presented in tables in this chapter is, for ease of comparison, given per megawatt of electrical generating capacity. The results of the resource analysis are summarized in Table XIV. In addition, Table XIV includes the results of a similar resource analysis completed by JPL for a gasified coal generating plant (Ref 8:6-12). These results were included for purposes of comparison because such a plant is likely to be the conventional plant of the year 2000.

Material Requirements

The first step in determining the total required materials was to determine the major components of each system. These components, many of which are discussed briefly in the previous portion of this chapter, are listed in Table II.

Using Table II as an outline, it was possible to assemble a list of specific materials required by each system. Table III is a summary of the material requirements for a TST generating system. Data for

Table II
System Components

System	Components
TST	Heliostats (mirrors) Central receiver tower Energy conversion system
TP	Photovoltaic collection system (includes 2:1 concentration)
OSR	Photovoltaic collection system (includes 2:1 concentration) Orbiting solar reflectors and associated propulsion system Space transportation system Space construction and maintenance facilities
SSP	Rectenna Photovoltaic collection and associated propulsion system (includes 2:1 concentration) Microwave power transmission system Space transportation system Space construction and maintenance facilities

Table III
Terrestrial Solar Thermal System
Material Requirement

Material	Mass Required (Metric Ton/MWe)
Steel	827[1]
Concrete	3666[1]
Silver	0.006[1]
Glass	133[1]
Aluminum	45.5[1]
Rock[2]	1495[1]
Heat Transfer Oil[2]	202[1]
Coal[3]	21,005[3]

[1]Ref 8:6-14 plus 30% for maintenance.

[2]Material required for the six hour storage at 70% rated power using caloria rock.

[3]Coal required to provide 20% backup power supply using a gasified coal generating system (Ref 8:6-12).

this table was obtained from a resource evaluation conducted by JPL to which 30% was added to cover maintenance requirements. Unfortunately, it was impossible to verify these figures because details as to how they were calculated are contained in an unpublished JPL internal document which was unavailable.

Table IV is a summary of the material requirements for a TP generating system. Again, it was impossible to verify these figures

Table IV
Terrestrial Photovoltaic System
Material Requirement

Material[1]	Mass Required (Metric Ton/MWe)
Concrete	52.4[2]
Silicon	18.7
Glass	30.4[2]
Aluminum[3]	673.[2]
Coal[4]	21,005.[4]

[1]Material required for battery storage not included.

[2]Ref 8:6-14 as adjusted. See explanation of this table in text.

[3]JPL suggests that steel could be substituted for some of Aluminum in order to reduce energy of production (Ref 8:6-14). Also, steel would reduce demand for Aluminum.

[4]Coal required to provide 20% backup power supply using gasified coal generating systems (Ref 8:6-12).

because they were based on calculations contained in the unpublished JPL document described above. However, the following changes were made to the JPL results. According to JPL estimates, the land area required for a TP plant rated at 1 MWe and operating for 30 years is 112,500 m^2 (Ref 8:6-12). Using the JPL assumption of a TP plant operating with a concentration ratio of 2:1 (Ref 8:1-3), and a geometric ratio of 2.15:1, the maximum area covered by solar cells is

$$\frac{112,500 \text{ m}^2}{2.15} = 52,325.6 \text{ m}^2 \tag{25}$$

Assuming the solar cells are 10 mils thick, then the total mass of silicon required is 30,967 kg/MWe, based on a silicon density of 2.33 g/cm^3 (Ref 28:10). However, 30,967 kg/MWe is less than half of the amount which JPL calculated (Ref 8:6-12).

Since details of the JPL calculations are not available, a decision was made to recalculate the TP system material requirements based on available information. Ehricke's work indicates that the annual solar insolation[3] level for a horizontal surface in southern Arizona (32° North latitude, 115° West longitude (Ref 13:Fig. 25)) is 2.15×10^9 kwhr/km^2-year (Ref 13:51). This agrees with information published by the Smithsonian Institute for a latitude of 30° and an atmospheric transmission coefficient of just under 0.8 (Ref 29:422). It was then possible to determine the area of silicon solar cells required, assuming the rated generating capacity is to be 1 MWe = 1×10^6 J/sec and the solar cell efficiency is to be 8.37%. The following calculation was performed

$$\frac{(3.1536 \times 10^7 \text{ sec/year})(10^6 \text{ J/sec})(10^6 \text{m}^2/\text{km}^2)}{(2.15 \times 10^9 \text{ kwhr/km}^2\text{-year})(3.6 \times 10^6 \text{ J/kwhr})(0.0837)(2)} = 24,339\text{m}^2 \tag{26}$$

where the conversion factor of 1 kwhr = 3.6×10^6 J was used (Ref 43: F-167). It should be noted that the solar cell efficiency used in the

[3] _Solar insolation_ is the rate at which direct solar radiation is received on a unit horizontal surface.

calculation was 8.37% because this would be the efficiency of the silicon solar cells used in a system having a concentration ratio of 2:1. Also, a 2 appears in the denominator of Eq (26) because only 50% of the incident radiation would be received directly by the solar cells. The rest would be reflected onto the solar cells by the mirrors.

The mass of silicon required to produce 24,339 m^2 of 10 mil thick solar cells would be 14,405 kg. Since the geometric ratio would be 2.15:1, the total land required for the solar cell array would be

$$(24,339\ m^2)(2.15) = 52,330\ m^2 \tag{27}$$

Assuming that battery storage does not require additional land, allowing 1/12 of the land for miscellaneous buildings as shown in Fig. 12, and allowing 15% for space between collectors to assist in maintenance, 64,540 m^2 represents a reasonable value for land usage.

For the TP resource analysis, the following assumptions were made:

1. The total land required for a TP plant would be 64,540 m^2/MWe.

2. The mass of silicon required would be 14,405 kg/MWe plus an additional 30% for maintenance replacement.

3. The mass of non-silicon materials required would be more a function of land area covered by the generating plant than a function of silicon mass.

With the above assumptions, it was possible to construct the material summary found in Table IV. Non-silicon material requirements were obtained by multiplying the figures provided by JPL by 0.574 in accordance with assumption three above. This fraction was obtained by calculating the ratio of the land area calculated above to the land area specified by JPL:

$$\frac{64,540}{112,500} = 0.574 \qquad (28)$$

To the material requirements obtained using the factor 0.574, 30% was added to cover 30 year maintenance requirements.

Table V contains a summary of the material requirements of the OSR system. Ehricke's design calls for a 1,000 km^2 area of solar cells without concentration (Ref 13:51). However, in order to be consistent with the TP and SSP systems, this design characteristic was changed. A ground system was substituted which would produce, with concentration, the same amount of electricity as a 1,000 km^2 area of solar cells without concentration. A solar cell efficiency of 8.37% for a concentration ratio of 2:1 was used in the following equation

$$(.13)(1,000\ km^2) = (2)(q)(0.0837) \qquad (29)$$

where q is the area of solar cells required. As in the TP system, a factor of 2 appears in Eq (29) because only half of the light used by the silicon cells would actually be intercepted directly by the cells. By solving Eq (29) for q, it was determined that a solar cell area of 776.6 km^2 would duplicate Ehricke's proposed ground system if a

Table V
Orbiting Solar Reflector System
Material Requirement

Material	Mass Required (Metric Ton/MWe)
Concrete	22.5[1]
Silicon	8.05[1]
Glass	13.1[1]
Aluminum	
Ground system[2]	289.[1]
All other systems	0.947[3]
Kapton	3.73
Insulation	0.010[3]
Copper	0.063[3]
Steel	0.021[3]
Inconel[4]	0.356[3]
Electronics	0.004[3]
Water	62.2[3]
Liquid Oxygen	4395.[3]
Liquid Hydrogen	646.3[3]
TBD	
Reflectors	14.6
Space Personnel Provisions	0.18[3,5]
Space Facilities	0.079[3,5,6]
Space Transportation System	0.003[3]

Table V (continued)

Material	Mass Required (Metric Ton/MWe)
Coal[7]	21,005.

[1]Based on data from Ref 8:6-14 as modified in producing Table IV. Includes 30% for maintenance replacement.

[2]Some replacement of Aluminum with steel may be possible,as is noted in Table IV.

[3]Based on calculations as explained in Appendix D. Does not include material for periodic maintenance and refurbishment of transportation system.

[4]A nickel alloy.

[5]Partially based on Ref 33:V-14.

[6]Ref 32:9-23.

[7]Coal required to provide 20% backup power supply using gasified coal generating system.

concentration ratio of 2:1 was used. Since the geometric ratio would be 2.15:1, the total area covered by the ground portion of the OSR system would be:

$$(776.6\ km^2)(2.15) = 1670\ km^2 \qquad (30)$$

It should be noted that 1670 km^2 is a larger area than the spot size produced by the sun's image, as focused by the OSR system reflectors from a three-hour orbit. Thus, introducing a concentration system would affect the OSR system in two ways. First of all, it would expose some of the ground system's solar cells to greater sunlight intensities than they would have without concentration. Secondly, a portion of the ground system's solar cells would not be illuminated at night. On the other hand, a 1,000 km^2 system would not capture all of the sunlight reflected by the orbiting reflectors. Thus, since these factors will, to a certain extent, offset each other, it was assumed that the electricity generated by the concentration system would be identical to that generated by the ground system proposed by Ehricke.

The OSR ground system would be essentially a TP generating plant. Material requirements for this ground system were determined by extrapolating from material requirements for a TP system as listed in Table IV. The amount of each material required for the TP system was multiplied by 31,913 because 1670 km^2 is 31,913 times larger than 52,330 m^2, the land area covered by solar cells and concentrators in the TP system. The exact values listed in Table V were then obtained by dividing by 74,200, the rated electrical capacity of the system in megawatts of electrical generating capacity. As with the TP system, a 30% replacement contingency was included in the figure for the ground system materials.

The material required for the construction of the orbiting reflectors was determined to be 1.0454×10^9 kg. This was calculated by multiplying the projected mass of the reflectors, 100 tons/km^2, by the anticipated number and size of the reflectors, 1320 reflectors at 8.73 km^2 each. A portion of this mass would be made up of sodium-coated kapton. It is estimated that the kapton would be 1.3×10^{-2} mm thick. Based on this thickness, and a kapton density of 1.42 gm/cm^3 at 25°C, (Ref 12:Table 1), 1320 reflectors, at 8.73 km^2 each, would require 2.13×10^8 kg of kapton. The remainder of the reflector mass, 8.32×10^8 kg, would be made of several materials which have not yet been identified. Therefore, this 8.32×10^8 kg of material is listed in Table V as "to be determined" (TBD). The material required for operation and maintenance of the orbiting portion of the OSR system is estimated at 30% of the initial reflector mass (Ref 14:55). To account for this material requirement, the material requirements in Table V for the orbital portion of the OSR system were increased by 30%.

The final material component of the OSR system is the space transportation system. This includes personnel provisions and space facilities. Details of the calculations associated with this component are very complex and are included in Appendix D.

Table VI contains a summary of the material requirements of the SSP system. Data for the ground system was taken from JPL data. The data from JPL was divided by two to conform to the NASA conclusion that two 5 GWe space components would irradiate one ground station. The JPL study assumed that only one 5 GWe space component would be used.

Table VI
Satellite Solar Power System
Material Requirement

Material	Mass Required (Metric Tons/MWe)
Concrete	226.2[1]
Aluminum	
Ground System[2]	87.9[1]
Space System	4.33[3]
Space Transport System	0.26[4]
Insulation	0.003[4]
Copper	
Space System	0.612[3]
Space Transport System	0.017[4]
Steel	
Ground System	3.09[1]
Space System	0.128[3]
Space Transport System	0.006[4]
Inconel[5]	0.096[4]
Electronics	
Space System	0.095[3]
Space Transport System	0.001[4]
Water	13.8[4]
Liquid Oxygen	975.[4]
Liquid Hydrogen	143.[4]

Table VI (continued)

Material	Mass Required (Metric Tons/MWe)
Argon	
Space System	0.250[3]
Space Transport	7.52[4]
Silver	0.0001[3]
Platinum	0.0003[3]
Samarium Cobalt	0.053[3]
Graphite	0.216[3]
Rare Material	0.008[3]
Silicon	2.55[6]
Silicon Cover Plate (Plastic)	0.60[6,7]
Silicon Adhesive	0.675[3]
Gold Kovar	0.737[3]
Black Paint	0.737[3]
Mylar	0.959[3]
Tungsten	0.009[3]
Molybdenum	0.009[3]
Ceramics	0.0002[3]
Stainless Steel	0.0001[3]
TBD	
Ground System	2.96[1]
Space System	0.008[3]

Table VI (continued)

Material	Mass Required (Metric Tons/MWe)
TBD (continued)	
Space Personnel Provisions	0.12[4,8]
Space Facilities	0.368[4,8,9]
Space Transportation System	1.09[4]

[1]Based on Ref 8:6-14 with the exception that one ground system is assumed to be capable of producing 10 GWe by using two 5 GWe space components.

[2]Some replacement of Aluminum with steel may be possible.

[3]Ref 34:Appendix IX.

[4]Based on calculations as explained in Appendix D. Does not include material for periodic maintenance and refurbishment of transportation system.

[5]Nickel alloy.

[6]Based on solar cells with efficiency of 8.37% at 100°C and a calculation similar to that done in Ref 34:IV.B.16-6,7,8,11. This calculation used an after-solar-cell system efficiency of 57.88% (Ref 34:IV.A.1-2), and includes 11% transportation degradation (Ref 32:9-5).

[7]Based on density of plastic cover of 55 gm/m^2 (Ref 34:IV.B.1b.11) and thickness of 1 mil (Ref 34:IV.B.1b.7).

[8]Partially based on Ref 33:V-14.

[9]Ref 32:9-23.

Data relative to the materials required to construct the space portion of the SSP system and the associated space equipment and provisions were obtained from a preliminary, but very detailed analysis done by the Johnson Space Center. As with the OSR system, Appendix D contains details relative to the material requirements for the space facilities and space transportation system. Also, as with the other systems, material requirements listed in Table VI, except for those for the space transportation vehicles, include 30% for maintenance.

As a final portion of this section on material requirements, the following is a list of items which would be required for the successful operation of one or more of the systems evaluated but which were not included in Tables III through VI.

1. The materials used in the equipment and buildings needed to mine, refine, fabricate, and transport (on the Earth's surface) system components.

2. The materials used in the Earth-based portion of the space transportation system.

3. The materials used in the electrical distribution system.

4. The materials used by all levels of plant management, both private and public.

The main reason why the above items were not included, was that information relative to the materials required for these items was not available. Of all of the four items listed above, the most significant ones are the second and third. The second item was included in determining the energy requirements of each system, as will be explained later in this chapter. The third item, although significant in terms of mass required, does not appear to be a significant discriminator between systems. JPL, in their analysis, found that although the SSP system could have shorter transmission distances, this factor did not result in a significant cost savings to the system. In fact, JPL found that the transmission system would add about 20% to the cost of electricity, regardless of the generating system (Ref 8:1-4).

Although JPL did not evaluate an OSR generating system there is a good possibility that it, too, will have transmission costs in the 20% range. In their analysis, JPL found that transmission costs are primarily influenced by two factors. These are the cost of the transmission equipment, and the electricity bus-bar cost (Ref 8:5-1). Of these two costs, the transmission equipment cost of the OSR system should be identical to that of the TP system. However, the electricity bus-bar cost is tied to transmission efficiency which tends to offset the higher transmission equipment costs. Whether the larger OSR system could operate at higher transmission efficiencies than the 96.5% transmission efficiency of a 2000 mile (Ref 8:5-3) TP transmission system, has yet to be established.

Land Requirements

Electrical generating plants essentially require land for three purposes: plant sites, transmission equipment, and material supply. In this case, material supply refers to the land from which the raw materials used in construction, operation, and maintenance of a power plant are mined, refined, and fabricated. Material supply would also include the land required for the Earth-based portion of the space transportation system. Finally, material supply refers to the land from which the fuel source for the power plant is derived. However, in the case of a solar plant, there would be no land requirement for fuel supply because the sun is the fuel source.

The analysis performed in conjunction with this thesis did not consider the land requirement associated with material supply. This is a very complex area which would require a major investigation of its own and was therefore beyond the scope of this investigation.

A listing of the land required for plant sites and transmission equipment for each of the four systems evaluated can be found in Table VII. In particular, the land requirements for the TST and SSP systems were taken directly from data supplied by JPL. The only change made was that the SSP land requirement was adjusted in accordance with the NASA plan to use two space components but only one ground component. The land requirement for the SSP plant site includes the land area which would have to be fenced off, as a safety zone, to protect the public from the hazards of microwave exposure. In this study, the dimensions of the safety zone were based on the Eastern European microwave radiation standard of 0.01 mW/cm^2 for continuous exposure. However, JPL pointed out that "at this power density (0.01 mW/cm^2), side lobe overlap of rectennas in the same

Table VII
Land Requirements

Land Use	TST (m^2/MWe)	TP (m^2/MWe)	OSR (m^2/MWe)	SSP (m^2/MWe)
Generating Plant	58,500[1]	64,540	27,758	98,250[1,2]
Transmission System	49,500[3]	49,500[3]	6,671	9,750[4]
Total[5]	108,000	114,040	34,429	108,000

[1]Ref 8:6-12.

[2]Ref 8:6-13 corrected for assumption of one ground facility per 10 GWe.

[3]Ref 8:6-12, 6-13, assuming an average transmission distance of 1650 miles.

[4]Ref 8:6-12, 6-13, assuming an average transmission distance of 650 miles.

[5]Does not include land used for the ground portion of the space transportation system nor the land needed to mine, refine, and fabricate materials used to construct and maintain the systems.

region may lead to substantial increases in land area requirements" above the 98,250 m^2/MWe figure used in this study (Ref 8:6-18, 19). For purposes of comparison, the current United States microwave exposure standard for man is 10 mW/cm^2 (Ref 32:8-9), which is considerably less conservative than the Eastern European standard. However, the more conservative Eastern European standard was used in this resource analysis because, as was admitted in a recent Marshall Space Center study, the "microwave exposure standards are somewhat loose in the United States [Ref 32:8-9]."

The method used to determine the land requirement for the TP and OSR generating plants was explained in conjunction with the material requirement section and will not be repeated here. Since these systems intercept sunlight rather than microwave radiation, no safety zones would be required. However, it should be noted that, as in the TP system, 1/12 of the solar cell and concentrator area was added to the land requirement for buildings and 15% was added for maintenance.

The land requirements associated with electrical transmission equipment were taken from JPL's figures for average transmission distances of 650 miles for the SSP system and 1650 miles for the TST, TP, and OSR systems (Ref 8:6-13). The SSP system would have a shorter transmission network because sunlight is affected by weather conditions much more than microwave radiation. Therefore, it would conceivably be possible to place SSP systems closer to the consumer (Ref 8:5-5).

Water Requirement

In a conventional electrical generating plant, large quantities of water are used to cool the generating system. However, as mentioned earlier, the location of the TST system in the arid portions of the

United States requires the use of some form of dry cooling technique. The other three systems do not have conventional steam generators and, therefore, do not require extensive cooling facilities. The solar cells and rectenna would rely on passive cooling techniques, as would the orbital portions of the OSR and SSP systems.

The water requirements of each system are listed in Table VIII. In general, there would be two major requirements for water. The first would be water used to clean the energy collectors. The TST system is most sensitive to dust and dirt collection and therefore would require the most frequent cleaning. A cleaning rate of once every five weeks has been projected for the TST system. The OSR and TP systems also require clean surfaces for optimum efficiency and would therefore be cleaned every 10 weeks (Ref 8:6-13). Water would be used in cleaning the TST, TP and OSR systems at the rate of 0.75 gal/m^2 per cleaning (Ref 8:6-22). Although it is possible that the ground portion of the SSP system would need some form of water cleaning, the exact amount of water has not been estimated.

The second major requirement for water would be as it is used in the space transportation system. Information relative to this use of water can be found in Appendix D.

There is one other area in which water would be required. Water is used to mine, refine, and fabricate the materials which would be used in building and maintaining the various systems. However, this water requirement was not included in the resource study.

Table VIII
Water Requirements

Requirement	TST (10^6 liters/ MWe)	TP (10^6 liters/ MWe)	OSR (10^6 liters/ MWe)	SSP (10^6 liters/ MWe)
Collector Cleaning	27[1]	23.2[2]	10[3]	?
Space Transportation	-	-	0.062[4]	0.014[5]
Total	27	23.2	10.062	0.014

[1]Ref 8:6-12. Includes some water loss in generating process, too.

[2]Ref 8:6-22. Cleaning of mirrors and solar cells, once each 10 weeks.

[3]Based on TP results adjusted to 1670 km^2 @ 74.2 GWe.

[4]From Table V.

[5]From Table VI.

Manpower Requirement

Manpower would be required from the time that the first kilogram of ore is extracted from the Earth until the end of each system's 30 year operative lifetime. Assuming that manpower requirements are very closely tied to costs, it would be possible to roughly approximate the total manpower requirements of each system by considering the total system cost.

Another, more accurate method of determining required manpower would be to add together the manpower requirements for each portion of a given system. However, this would require a detailed analysis of each system. Since such an analysis was beyond the scope of this current research effort, a decision was made to adopt data from a JPL study.

In the JPL study, manpower requirements for material acquisition, construction, operation, and maintenance of a TP, a TST, and a SSP system were determined. The exact details of the calculations are not available. However, it is known that the maintenance manpower estimates are, in part, based on a mirror cleaning rate of 156 m^2/manhour (Ref 8:6-23).

Table IX contains a list of the manpower requirements of each system. The space/ground division of the SSP manpower requirements was accomplished by splitting the JPL manpower requirements in proportion to the ratio of the cost of one half of the space systems, including the space transportation system, to the cost of the ground systems (6.389:1). Once split, the ground manpower figures were divided by two to account for the reduced ground system size in the NASA plan.

The OSR manpower requirements were obtained by extrapolating from data for the other systems. Manpower requirements for the ground system were extrapolated from the TP data in accordance with

Table IX
Manpower Requirements

Requirement	TST (man-years/MWe)	TP (man-years/MWe)	OSR (man-years/MWe)	SSP (man-years/MWe)
Construction:				
Ground[1]	28.5[2]	12.1[2]	5.20	6.78[3]
Space[4]	-	-	95.52	69.87[3]
Operations & Maintenance:				
Ground	28.5[2]	28.5[2]	12.26	0.01[3]
Space	-	-	0.20	0.14[3]
Total	57.0	40.6[5]	113.18[5]	76.80

[1]Based on 2000 manhours/man-year.

[2]Ref 8:6-12.

[3]Ref 8:6-12 but division is based on ratio of costs.

[4]Based on 2480 manhours/man-year.

[5]Does not include manpower for ground system material acquisition.

the ratio of the OSR plant land requirement to the TP plant land requirement, as given in Table VII (0.43:1). Here land was used for comparison because the TP and OSR ground systems would be very similar.

Similarly, the manpower requirements for the space portion of the OSR system were determined by extrapolation from SSP data. In this case, the extrapolation was based on the ratio of the cost of the space portion of the OSR and SSP systems (1.37:1). Here cost was used for comparison because although the OSR and SSP systems are not very similar, it was assumed that cost is closely tied to manpower requirements.

It should be noted that the JPL data was given in units of man-hours. In converting it to man-years as expressed in Table IX, it was assumed that an average worker, on the space system, would work eight hours/day for 310 days/year. For workers on the ground system, the figure used was eight hours/day for 250 days/year (Ref 30:Table 5).

Energy Requirement

In the past, apparently very little emphasis has been placed on evaluating a system on the basis of the energy consumed by its construction. However, when dealing with an energy producing system, it is essential to ensure that it will produce more energy, in the long run, than will be consumed in its construction, operation, and maintenance. To determine if the four systems considered in this resource analysis would indeed be energy producers, this energy analysis was undertaken.

In 1975, Battelle Columbus Laboratories completed an evaluation of the energy required to mine and refine some of the materials most frequently used by United States industries. This data was used in this resource analysis, along with a small amount of data from other sources, to determine the energy required to construct, operate, and maintain each of the four systems evaluated. Unfortunately, data was not available for the amount of energy needed for fabrication of system components. Therefore, the energy of production figures contained in Tables X, XI, and XIV do not include the energy needed to take the various materials from their refined state to their finished product configuration.

Table X contains a summary of the energy of production for the materials consumed by the four systems. Those materials for which no energy data was available were not included in Table X. In each case, the energy figure listed is the energy required to mine and refine the specific material. The Battelle study assumed that any electricity required in the mining or refining process was generated at a cost of 1.05×10^4 Btu/kwhr (Ref 2:A-1).

Table XI contains a summary of the total energy required to construct, operate, and maintain each of the four systems studied, except as noted above. The total energy figures were obtained by multiplying the data from Table X by each system's total requirement for each material, as specified in Tables III through VI. Those materials from Tables III through VI for which no energy of production

Table X
Material Energy of Production

Material	Energy of Production[1] (10^6 Btu/Metric Ton)
Aluminum	269.0[2]
Copper	123.4[2]
Steel	54.6[3]
Liquid Oxygen	9.9[4]
Liquid Hydrogen	668.8[5]
Argon	10.3[3]
Graphite	176.4[6]
Silicon	6163.5[7]
Tedlar/Kapton	84.7[8]
Molybdenum	187.7[3]
Ceramic	34.1[9]
Concrete	8.4[10]
Glass	19.2[11]
Rock	0.26[12]
DDT&E	6500 Btu/$(1977)[13]

[1]Energy required to mine and refine product. It does not include the energy to transport or work with the material after it has been refined.

[2]Ref 2:5.

[3]Ref 5:145.

Table X (continued)

[4]Ref 3:6.

[5]Ref 5:145. Hydrogen is assumed produced by electrolysis.

[6]Ref 2:A-1. Approximation using the energy of production of graphite electrodes.

[7]Ref 2:A-1; Ref 28:13, 16. Using today's methods.

[8]Ref 5:145. In this resource analysis, this energy of production is assumed to be appropriate for both mylar and kapton.

[9]Ref 5:147.

[10]Ref 2:5. Approximation using the energy of production of Portland cement.

[11]Ref 2:5. Approximation using the energy of production of glass containers.

[12]Ref 2:A-1. Approximation using the energy of production of limestone.

[13]Ref 47.

was known, were included in Table XI under the category of "other." The amount of energy required for "other" was determined by taking the ratio of the mass of materials whose energy was unknown to the mass of materials whose energy was known. This ratio was then multiplied by the total energy of production of the known materials. By calculating the energy of "other" in this way, the materials in the "other" category were essentially assumed to have the same energy

10^{12} BTU ⇒ 30 MWatt-years

Table XI
System Energy of Production

System Component[1]	TST (10^9 Btu/ MWe)	TP[2] (10^9 Btu/ MWe)	OSR[2] (10^9 Btu/ MWe)	SSP (10^9 Btu/ MWe)
Aluminum	12.2	181.	78.0	24.9
Copper	-	-	0.008	0.078
Steel	45.2	-	0.001	0.176
Liquid Oxygen	-	-	43.5	9.65
Liquid Hydrogen	-	-	432.	95.6
Argon	-	-	-	0.080
Graphite	-	-	-	0.038
Silicon	-	115.	49.6	15.7[3]
Tedlar/Kapton	-	-	0.316	0.081
Molybdenum	-	-	-	0.002
Ceramic	-	-	-	7×10^{-6}
Concrete	30.8	0.440	0.189	1.90
Glass	2.55	0.583	0.252	-
Rock	0.389	-	-	-
Other[4]	2.98	-	8.69	2.18
DDT&E	0.901[5]	0.243[5]	3.00[6]	47.2[7]
Total[8]	95.02	297.266	615.556	197.585

[1]Coal for system backup was not included in this analysis.

[2]Does not include battery storage material.

Table XI (continued)

[3]An approximation based on a cutting technique which will probably not be used.
[4]"Other" includes all materials for which no energy of production was available, except as exempted in items (1) and (2) above.
[5]Based on amortization over one 10 GWe plant.
[6]Based on amortization over one 74.2 GWe plant.
[7]Based on amortization over one 10 GWe plant but doesn't include cost of 4 mil solar cell development.
[8]Does not include energy of material fabrication or of the actual construction and maintenance process. However, does include a 10% DDT&E Cost Contingency.

distribution as the rest of the system.

The energy required for system development was calculated by using additional information supplied by Battelle. In recent studies, Battelle has found that from 3100 Btu to 15,500 Btu are expended in the aerospace industry for every 1974 dollar spent. The average figure currently used by Battelle is 8000 Btu per 1974 dollar (Ref 47). This average figure was adjusted to 1977 dollars by assuming that the average aerospace inflation rate over the past 3 years has been 7% per year. Thus, an average figure of 6500 Btu per 1977 dollar was calculated. This average figure was then multiplied by the total projected developmental costs, in 1977 dollars. As a result of this procedure, it was possible to determine the estimated energy which would be expended in the development of the various systems.

It should be noted that by making the assumption of 6500 Btu per dollar, the energy required to fabricate the developmental materials was included in the total energy figure. However, because the fabrication costs of the actual generating systems are not known, this same procedure could not be used to account for the energy of fabrication of the actual generating systems themselves.

Cost Requirements

The final phase of the resource analysis was to determine the costs of development, design, testing, evaluation, construction, operation, and maintenance of each system. Table XII contains a listing of the costs per flight, in 1977 dollars, of the various space transportation vehicles along with the projected total developmental costs, in 1977 dollars, of each space transportation system. This data was projected from cost figures in 1976 dollars, assuming a 7% rate of inflation in 1976.

The summarized results of the cost analysis are contained in Table XIII. Costs of the TST and TP systems were given by a JPL study without detailed explanation. These costs include an unknown amount to cover the cost of appropriate storage systems.

The cost of the SSP system, the cost of the SSP space support facilities, and the cost of the SSP space operations were all taken directly from estimates made by the Marshall Space Flight Center.

Table XII
Space Transportation Unit Costs[1]

Vehicle	Development[2] Cost ($ 10^9)	Cost per[3] Flight ($ 10^6)
HLLV	11.8	9.6[4]
PLV	1.2	10.7[5]
$COTV_L$	1.1[6]	32.1[7]
$COTV_G$	1.1[6]	10.7[7]
POTV	1.6	12.8[7]

[1]1977 dollars.

[2]Ref 34:XI-5 as adjusted from 1976 to 1977 dollars assuming a 7% inflation rate in 1976. HLLV reference is Ref 32:12-10.

[3]Includes cost of vehicles, operations, and amortized spares/refurbishment. Cost is adjusted from 1976 to 1977 dollars assuming a 7% inflation rate in 1976.

[4]Ref 11:208.

[5]Ref 33:VI-18.

[6]The $COTV_L$ and $COTV_G$ are considered as one developmental problem. The total developmental cost was evenly split between the two systems, for this table, although the $COTV_L$ is likely to be the most costly.

[7]Ref 33:VI-19.

The design, development, testing, and evaluation (DDT&E) costs for the SSP itself were taken from work by the Johnson Space Center.

Table XIII
System Costs[1]

Cost Item	TST Cost ($10^6/MWe)	TP Cost ($10^6/MWe)	OSR Cost ($10^6/MWe)	SSP Cost ($10^6/MWe)
Ground System[2]	4.1[3]	6.5[3]	2.81	0.35[4]
Orbital System[2]	-	-	0.54[5]	1.51[4]
Space Operations and Facilities[2]	-	-	0.93[5]	1.66[4]
Maintenance[6]				
Ground System	1.2	2.0	0.84	0.10
Orbital System	-	-	0.16	0.45
Space Facilities	-	-	0.28	0.50
Space Transportation[7]				
HLLV	-	-	5.16	1.14
PLV	-	-	0.03	0.04
$COTV_L$	-	-	0.17	0.30
$COTV_G$	-	-	0.59	0.11
POTV	-	-	0.001	0.04
DDT&E				
System	0.126[8]	0.034[8]	0.21[9]	4.92[10]
Space Transport	-	-	0.21[11]	1.68[11]
Contingency[12]	0.54	0.85	1.19	1.28
Total	5.97	9.38	13.12	14.08

Table XIII (continued)

[1]1977 dollars. Source data not in 1977 dollars was adjusted to 1977 dollars assuming 7% inflation per year. In general, JPL data was in 1975 dollars, NASA data in 1976 dollars, and Ehricke data in 1977 dollars. Does not include cost of money, such as interest on debt.

[2]Initial capital investment.

[3]Ref 8:6-12.

[4]Ref 32:14-7, 8.

[5]Based on Ref 14:54. Includes cost of 1320 reflectors, each 8.73 km^2.

[6]Based on 30% of cost of initial system component.

[7]Based on results of Appendix D and Table XII. The OSR $COTV_L$ cost is for the $COTV_G$ going from 550 km to 1100 km.

[8]Based on Ref 8:1-2, amortized over one 10 GWe power plant.

[9]Ref 13:53, Ref 8:1-2, and Ref 33:X-8 for technology and advancement phase costs. Assumes no Lunetta development program. Amortized over one 74.2 GWe plant.

[10]Ref 33:X-7, 8 and Ref 34:IX-5. Amortized over one 10 GWe plant.

[11]From Table XII. Even though some of the transportation systems used by the OSR system would be smaller than those used by the SSP system (Appendix D), development costs are assumed unchanged. Amortized over 74.2 GWe plant for OSR and 10 GWe for SSP systems.

[12]10% of all costs.

However, these developmental costs are not believed to include the cost of developing a 4 mil silicon solar cell. As mentioned earlier, the cost of such a development program is not known at this time.

The total cost of the ground portion of the OSR system, $2.08 x 10^{11}, was determined by multiplying the cost of a 1 MWe TP system by 31,913. The number 31,913 was used because it is the electrical rating, in MWe, of the OSR ground system without the orbiting reflectors. This electrical rating is based on the assumption that the OSR ground system, as designed, would annually receive 2.15 x 10^9 kwhr/km^2 of solar radiation directly from the Sun (Ref 13:51).

The cost of the OSR orbital system was based on a projected reflector unit cost of \$3.5 x 10^6/km^2 (Ref 13:54). The cost of the space facilities and associated operations was based on a projected cost of \$2 x 10^6/km^2 each time the reflector is coated with sodium (Ref 14:54).

The DDT&E costs for the OSR system were extracted from projections for the OSR system (Ref 13:53). The only exception was that the SSP technology and advancement cost was added to the OSR DDT&E costs because no OSR projection for this cost was available (Ref 33: X-8).

It should be noted that although Ehricke expects the Lunetta program to precede the Soletta II program, that assumption was not made in this resource evaluation. Instead, this analysis was based on the assumption that the entire space portion of the OSR system would have to be developed solely for this program, including the sodium coating facility and the electric thrusters used for reflector control.

For all four of the systems studied, maintenance costs were assumed to be 30% of the ground and space system costs as well as 30% of the cost of space operations and facilities. Space transportation costs were based on the analysis in Appendix D. All space transportation DDT&E costs were determined from Table XII.

Resource Evaluation Summary

Table XIV is included as a summary of the results of this resource evaluation. Although it was the goal of this study to evaluate all systems on the basis of the same parameters, this was not always possible because of the unavailability of sufficient data. Therefore, the information contained in Table XIV should be used to compare the four systems only in light of the assumptions presented throughout this chapter. For a further discussion of the information contained in Table XIV, refer to Chapter VI of this thesis.

The results of a JPL resource analysis for a gasified coal (GC) generating system are also included in Table XIV. These results were not verified and may have been obtained using different assumptions than used in the resource analysis of this thesis. However, to first order, they do permit a rough comparison between conventional and solar generating systems.

Table XIV
Resource Analysis Summary[1]

Item/Units	TST	TP	OSR	SSP	GC[2]
Rated Capacity (GWe)	10.0	10.0	74.2	10.0	10.0
Material (Metric Ton/MWe)	6,368.5[3]	774.5[3]	5,456.1[3]	1,473.4	105,420.0
Land (m^2/MWe)	108,000.0	114,040.0	34,429.0	108,000.0	117,000.0
Water (10^6 liters/MWe)	27.0	23.2	10.1	0.01	15.0
Manpower (manyears/MWe)	57.0	40.6	113.2	76.8	39.6
Energy of Prod. (10^9 Btu/MWe)	95.0	297.3	615.6	197.6	--
Cost ($\$10^6$/MWe)	6.0	9.4	13.1	14.1	2.1

[1] Clarifying comments can be found in the Resource Analysis text and in Tables III through XIII.

[2] Ref 8:6-12, for a gasified coal (GC) generating system.

[3] Excluding coal for backup power system, and other exclusions listed in the text.

V. Issues that Affect the Development of Solar Electric Generating Systems

Because the four systems evaluated in Chapter IV are currently at different stages of development, each system has a different chance of becoming operational. It is the purpose of this chapter to list some of the major technological advancements that must be made, some of the environmental concerns that must be resolved, and some of the general questions that must be answered before the four systems could become operational.

This chapter is not all inclusive. Although the lists in this chapter present many of the problems that remain to be solved, it is likely that new problems will become apparent as further work is done in the field of solar electric generation. In addition, the solution to one problem may lead to another. However, it is the goal of this chapter to present a rough idea of the magnitude of the problems faced by the various systems.

Technological Advancements

A very broad definition of "technological advancement" is used in this thesis. Technological advancement is assumed to include the demonstration of the validity of assumptions made in performing the resource analysis of Chapter IV as well as the development of new technologies.

Technological advancements that must be made before one or all of the four systems can become operational, are listed in Table XV. The item numbers refer to the items listed below.

1. The impact of varying weather conditions on plant performance must be more fully understood.

2. The ERDA cost goal of \$.50 per peak watt of electrical output must be achieved (Ref 8:1-6).

3. An advanced Redox battery storage subsystem must be developed (Ref 8:1-6).

4. Techniques must be developed to mass produce 10 mil thick silicon solar cells with 13% efficiencies at air mass 1 and cell temperature of 28°C (Ref 8:4-36).

5. It must be demonstrated that sodium coating of reflectors in space is feasible.

6. Manufacturing and construction techniques for the components of an OSR system reflector must be developed.

7. Thirty year solar cell lifetimes must be demonstrated (Ref 32:A-5).

8. Service vehicles and techniques for in-orbit and ground maintenance as well as improved pressure suits for astronauts during extravehicular activity need to be developed (Ref 32:8-16, 7-108). This includes the development of protective devices for astronauts to protect them against radiation hazards in space. Unprotected astronauts working in space are expected to receive radiation dosages that exceed the suggested daily exposure limits to bone marrow and to sensitive human organs such as the ocular lens and testes. Astronauts would also receive an amount of radiation equal to the suggested exposure limit to the skin (Ref 32:8-14).

9. The economic feasibility of space construction, orbital factories, and the construction of light-weight deployable structures must be established (Ref 8:4-35).

10. The entire space transportation system must be developed. For the SSP, this includes the Cargo orbital transfer vehicle, $COTV_L$, which would use a speculative magnetoplasmadynamic

Table XV
Required Technological Advancements

Item #[1]	TST	TP	OSR	SSP
1	x[2]	x	x	-[3]
2	-	x	x	x
3	-	x	?[4]	-
4	-	x	x	-
5	-	-	x	-
6	-	-	x	-
7	-	x	x	x
8	-	-	x	x
9	-	-	x	x
10	-	-	x	x
11	-	-	-	x
12	-	-	-	x
13	-	-	-	x
14	-	-	x	x
15	-	-	-	x
16	-	-	-	x
17	-	-	-	x
18	-	-	-	x
19	-	-	-	x

Table XV (continued)

[1]Item numbers refer to items listed in text under <u>Technological Advancements</u> section.
[2]"x" means item applies.
[3]"-" means item does not apply.
[4]"?" means item may apply.

arcjet propulsion system (Ref 33:VI-8, 9, 10 and Ref 32:12-11).

11. The SSP design uses plastic silicon solar cell covers. "This assumption has little evidence to support it, and much effort will be needed to verify and test light plastic covers [Ref 32: 7-67]."

12. Techniques must be developed to mass produce 4 mil thick silicon solar cells with 13% efficiencies at air mass zero and cell temperature of 28°C (Ref 8:4-36).

13. Solutions must be found to the problem of space plasma interaction with high voltage solar arrays (Ref 32:7-32).

14. Improved techniques must be developed relative to the simulation of static and dynamic properties of large structures in space (Ref 32:7-108).

15. Advancements must be made in microwave technology relative to the amplitron, low noise levels, high efficiency, active cooling, and safeguards against arcing of the microwave system. In addition, open cathodes that can withstand the heat and high current of the SSP system for 30 years must be developed (Ref 32:7-108).

16. New materials must be developed that would be insensitive to thermal distortion but which easily conduct microwaves (Ref 32:7-108).

17. Continuously varying microwave phase shifting devices with resolutions in the order of fractions of degrees must be

developed (Ref 32:7-108).

18. A completely new method of microwave phase control must be developed (Ref 32:7-108).

19. Materials must be developed that will degrade very little over 30 years, for use in the rotary joint subsystem (Ref 32:7-109).

It should be noted that relative to the SSP system, NASA considers the demonstration of man's ability to manufacture and assemble equipment in space, and the achievement of projected mass, efficiency, lifetime, and cost goals for silicon solar cells, as the two most critical areas to the successful development of the SSP system (Ref 32:A-5).

Environmental Concerns

The TST and TP systems are not expected to have much of an impact on the environment. However, their backup systems which would probably be coal, could have an impact. However, there are a number of concerns relative to the effects of the SSP and OSR systems on the environment. Some of these concerns are:

1. What affect will prolonged microwave exposure have on humans, animals, flora, and microorganisms (Ref 10:61-62)?

2. How much thermal pollution will result from microwave or sunlight radiation directed toward the Earth? As one author put it, the absorption of energy by the ionosphere from a 5000 megawatt beam could be ". . . as much, if not more, than the energy absorbed from the Sun [Ref 10:61-62]."

3. "The projected ten or more shuttle flights per day in support of the power-satellite fleet . . . could probably cause serious and troublesome disruption of communications . . . [Ref 10:63]." A similar concern was expressed by NASA (Ref 34:IV-C-2-b-4).

4. There is concern over the potential modification of water cluster ion concentration and the modification of stratospheric--mesopheric trace gas composition, aerosol distribution, and thermal balance (Ref 10:65).

5. There is concern that a large space vehicle would abort and crash into a heavily populated area (Ref 34:6-50).

6. Heat released by heavy lift launch vehicle (HLLV) launches would be of a magnitude sufficient to cause changes in the local weather patterns. How serious this would be is unknown (Ref 34:VIII-B-5).

7. Noise pollution and associated vibrational effects would occur as a result of frequent HLLV launchings (Ref 34:VIII-B-5).

8. "Even a small fraction of the HLLV fuel could cause significant local ecological damage if the fuel were not properly contained [Ref 34:VIII-B-5]."

9. Releasing propellants into the magnetosphere from the orbital transfer vehicle could possibly cause magnetic substorms (Ref 34:VIII-B-8).

10. Space debris could increase significantly (Ref 34:VIII-B-8, 9).

11. Reflected sunlight from space will possibly interfere with astronomy.

12. Entry systems, of space transportation vehicles, are expected to create sonic booms and produce NO_x in the stratosphere (Ref 34:VIII-B-9).

13. Solar array technology may lead to problems with material toxicity, the handling of waste products, and the impact of the manufacture of large quantities of solar cells and related components (Ref 8:7-108).

Other Questions to Be Answered

In addition to the environmental concerns, it is necessary to consider the vulnerability of each generating system. Currently, the Dayton Power and Light Company services all or parts of 24 counties in West Central Ohio. For this service area, the power company uses three electrical generating plants plus parts of three others. Together these plants have a rated generating capacity of 2.4 GWe (Ref 46). Thus, the proposed SSP system would, in one plant, have a rated capacity four times greater than the capacity of the entire Dayton Power and Light Company. If that one plant stopped operating, for whatever reason, millions of people could suddenly be without electricity. The problem would be even worse for the OSR system. Although the TST and TP systems were evaluated as 10 GWe plants, they could be constructed on smaller scales to reduce this problem.

The SSP system, in a study by JPL, was found to be moderately vulnerable to sabotage and blackmail, highly vulnerable to military attack, and highly vulnerable to legal liability due to regulation and international law. However, JPL did not consider the terrestrial generating systems to be vulnerable in any of these areas (Ref 8:6-46). It can be safely assumed that since the OSR system has some space components but also has a ground component that can operate independently of the space component, its vulnerability would be somewhere in between that of the SSP and the terrestrial systems.

The vulnerability described above could be enough of a factor to end further funding of space related electrical generating systems. Especially from a military point of view, large scale dependence upon a system such as the SSP or OSR for electricity could place the United States in an extremely vulnerable position in times of war. It is only necessary to recall the recent New York blackout in order to understand the effect upon this nation of the elimination of all or most of its electrical power. Such a concern is particularly significant to the SSP system because, of the four systems evaluated, it is the only one that could not function without its orbital component. Furthermore, a foreign nation could destroy an SSP orbital component without appearing to be at war with the United States by making the destruction appear to be the result of an accidental orbital collision (Ref 8:6-47). Defense against such a threat would be difficult, if not impossible.

VI. Concluding Comments

From the feasibility study of orbiting solar reflectors, it is possible to conclude that single-mirror orbiting solar reflectors have potential uses in the space environment for in space power generation. However, their use for Earth power generation is most likely limited to a system such as the Ehricke OSR system. With this in mind, the resource evaluation was undertaken.

The purpose of the final chapter of this thesis is to comment on the questions raised in Chapter V and on the results of the resource evaluation, to put these results into perspective, and to discuss the implications of these results and the associated unanswered questions.

Resource Evaluation Comments

The results of the resource evaluation, contained in Table XIV, must be considered in light of the state of development of the four systems. Because the terrestrial systems are much closer to "state-of-the-art" systems, the data relative to their resource consumption is probably more realistic than is the data for the OSR and SSP systems. The following are a few comments relative to the uncertainty of the numbers given in Table XIV for each of the resources evaluated.

Material Requirements. The TST system would not be as significantly affected by weight growth as would the others. In fact, the use of more speculative mirror designs could reduce material requirements by as much as 60% (Ref 8:6-15). The TP system data used for this analysis was very sketchy and JPL findings were higher, as discussed in Chapter IV. Solar cell technology has changed rapidly in recent years, a fact that has made it difficult to find consistent estimates for TP material requirements.

As mentioned earlier, the OSR reflector mass could fall anywhere within a range from 50 to 150 tons/km^2 (Ref 13:26,40,43), depending upon the size of the reflector units. For the SSP system, the mass of the orbital system ranges from 71,505 metric tons to 122,045 metric tons (Ref 34:IV-A-5-9).

The space transportation system for the OSR and SSP systems are highly speculative. A weight growth of 50% is virtually certain, although this growth could be at least partially offset by improvements in current designs. For example, it has been demonstrated that hydrocarbon fuel rather than hydrogen should be used for the first stage of the HLLV because of the greater energy density of hydrocarbon fuel (Ref 34:II-2). However, because of insufficient information, hydrogen fuel was used in this resource analysis.

Land Requirements. Land requirements could increase for the SSP system if side lobe overlap problems develop. However, social concerns may have more of an impact on land requirements than anything else. The reason for this is that land requirements for solar electric generating plants are much greater than for nuclear generating plants. Construction of plants requiring hundreds of square miles of land could be delayed because of public resistance.

Water Requirements. Water requirements for the space transportation system in Table XIV may be slightly exaggerated because of the assumption that the intra-orbital vehicles were miniature HLLV systems (Appendix D). Water requirements for the ground systems of all four proposed systems are negligible when compared to conventional electrical generating plants using wet cooling techniques (Ref 8:6-12).

Manpower Requirements. Manpower requirements are somewhat uncertain at this time. Although a 120 man construction crew was assumed to be required for the orbital construction of the SSP system, estimates of required manpower place the crew size as high as 600 personnel (Ref 34:II-2).

The terrestrial systems would require considerable maintenance forces. However, this may be beneficial in that it would help to prevent the "boom/bust" phenomenon that occurs when a large construction force, required to build a big system, is suddenly laid off because of the completion of the construction (Ref 8:6-23). The exact size of this maintenance force is uncertain primarily because the required size of the plant is uncertain.

Energy Requirements. The major uncertainty in the energy of production figures is caused by the fact that the energy of fabrication of materials is unknown, that major improvements are expected in the manufacture of solar cells, and that the material requirements are uncertain.

Cost Requirements. The costs included in the resource evaluation were the nominal costs listed in the source documents with the exception of the cost of development of the SSP system. In the case of SSP system development, the upper limit cost was used because it is frequently the figure quoted by other sources (Refs 8; 16).

The uncertainty in cost is most significant for the OSR and SSP systems. JPL estimated the energy cost of a TST plant with gasified coal backup at from 60 to 120 mills/kwhr, for a TP plant with gasified coal backup at from 75 to 220 mills/kwhr, and for a SSP plant at from 60 to 500 mills/kwhr. The major factors contributing to the large cost uncertainty for the SSP system were the uncertainty of solar cell costs and the uncertainties in the cost and performance of many other major subsystems (Ref 8:1-5,6). Considering the similarity between the OSR and SSP systems, in terms of the technology they would require, it can be assumed that the OSR system would have much of the same cost uncertainties as listed above for the SSP system.

The Resource Evaluation in Perspective

Energy of Production. An electrical generating system should be an energy producer, not an energy consumer. What is meant by this statement is that the electricity required to construct, operate, and maintain an electrical generating system should be less than the amount of electricity that the system will generate during its lifetime.

The time that it would take each of the four electrical generating systems to pay back the energy requirements listed in Table XIV is:

1. Terrestrial solar thermal system - 1.48 years
2. Terrestrial photovoltaic system - 4.62 years
3. Orbiting solar reflector system - 9.56 years
4. Satellite solar power system - 2.53 years

The energy pay back times listed above were calculated by assuming that the energy consumption, for each system, listed in Table XIV could have been used instead to generate electricity at the energy consumption rate of 1.05×10^4 Btu/kwhr (Ref 2:A-1). It was then determined how long it would take each of the four systems to generate an amount of electricity equal to the effective amount that they would consume during construction, operation, and maintenance. It was assumed that the TST, TP, and OSR systems would produce electricity at a rate of 70% of rated capacity (Ref 8:1-3), while the SSP would produce electricity at 85% of rated capacity (Ref 32:14-3).

Total Investment Costs. One final way of considering cost is to consider the total cost of investment, independent of the generating capacity of the plants. The terrestrial systems could be built on small scales and would, therefore, not require investments of the magnitude required by the SSP and OSR systems. The SSP system would need to be built on a 10 GWe scale in order to offset the large support costs of such a system. Thus, the minimum investment in the SSP program, obtained by multiplying the SSP cost per megawatt in Table XIV by 10,000, would be $141 billion. This would purchase one 10 GWe system.

The OSR system cannot be built on a scale smaller than 74.2 GWe because of the limitations of solar optics discussed in Chapter II. Thus, its minimum initial investment cost, obtained by multiplying the OSR cost per megawatt in Table XIV by 74,200, would be $974 billion.

In defense of the OSR system, however, it should be noted that its cost effectiveness could be enhanced greatly by building two or more ground stations. Since the ground system cost is low compared to the cost of the rest of the system, and since one set of reflectors could easily illuminate two ground systems without decreasing the rated capacity of either ground system, this would significantly reduce the cost of electricity produced by the OSR system.

Finally, it should be noted that both the OSR and SSP systems would become considerably cheaper per copy if several were constructed. By doing this, it would be possible to spread the cost of development over several plants and thus bring down the cost of an individual plant.

Final Comments

Development of the various systems is just beginning and little is known about the solutions to the many problems that must be solved. However, it is this author's opinion that considering the environmental concerns, the vulnerability questions, the need for technological advancements, and the resource requirements, it seems difficult to justify further commitment to the OSR and SSP programs. Studies by JPL, ERDA, and NASA, as well as this one, have all shown that, at best, the SSP system would be competitive with other ground solar systems like wind, geothermal, TST, and TP (Refs 8:1-2; 34:II-3; 16:7). Thus, it seems unreasonable that billions of dollars should be spent on very uncertain programs that, even if built at the conservative cost estimates of today, would be no cheaper than terrestrial methods.

Perhaps solar power generation is not the answer to the energy crisis. Continuous operation of terrestrial solar electric generating plants is seriously hampered by the variability of sunlight. However, it is possible, as the Barstow, California TST plant demonstrates, to build a prototype terrestrial solar electric power plant today that, when operational, could be used, at a minimum, to supplement electrical supplies. If necessary conventional plants could be used during the evening hours. By using such an electrical generating plan, it would be possible to stretch out the lifetime of the finite fuel resources of the world, until a new and even better solution to the energy crisis can be found.

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Appendix A

Matrix Methods Applied to the AOSR

Since the early nineteen-sixties, the use of matrix operators to solve ray tracing problems has become increasingly popular (Ref 24: 171). The purpose of this appendix is to describe how matrix methods were used to complete a ray trace of the AOSR system.

In matrix representation, a typical ray would be written as follows:

$$\begin{bmatrix} n\ \alpha \\ y \end{bmatrix} \tag{10}$$

as described in Chapter III.

To simplify the ray trace analysis, the primary and secondary mirrors of the AOSR system were approximated as thin lenses. Mirrors and thin lenses are similar because neither requires consideration of the effects of transmission through themselves. To further simplify the analysis, primary and secondary mirrors, which in reality would have a slight curvature, were projected onto planes. The planes for the primary and secondary mirrors were defined to be perpendicular to the AOSR optical axis, intersecting the optical axis at points G and E in Fig. 15, respectively. Points G and E are the points the actual mirrors would have in common with the optical axis. The

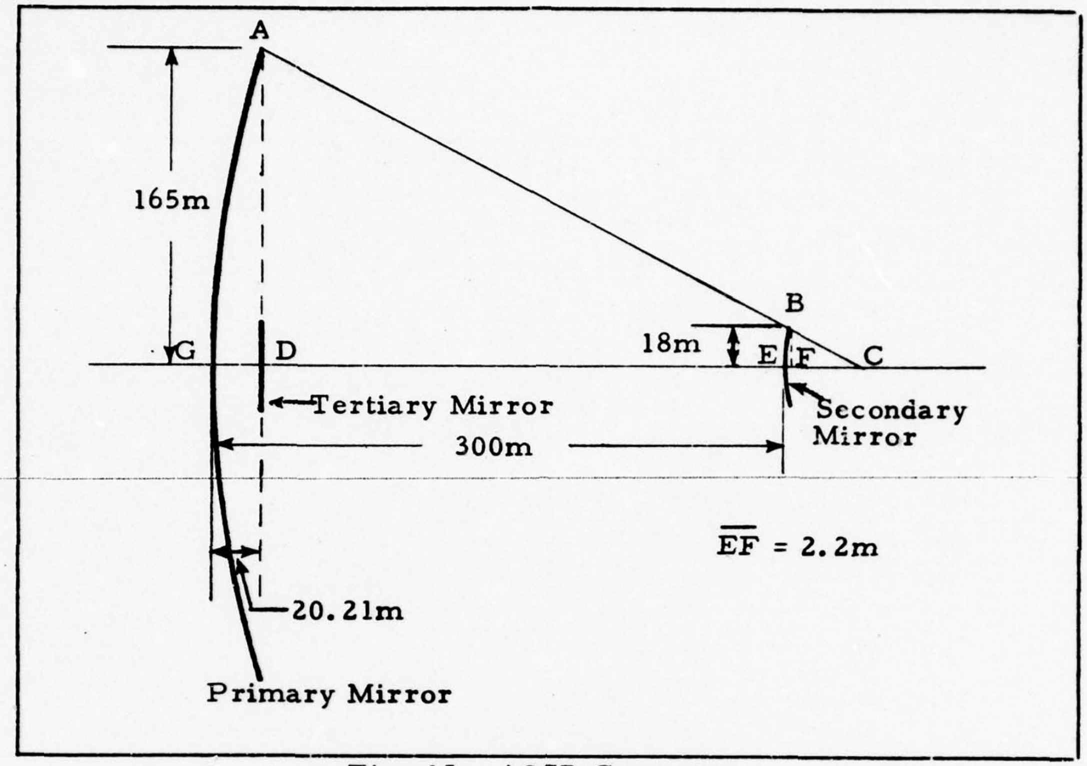

Fig. 15. AOSR Geometry

tertiary mirror required no projection because it was assumed to be a planar mirror.

Since the primary and secondary mirrors of the AOSR were approximated as thin lenses, each was represented by a matrix of the form

$$MM = \begin{bmatrix} 1 & -\frac{1}{f} \\ 0 & 1 \end{bmatrix} \tag{31}$$

where f is the focal length of the respective mirror.

To account for the amount a ray of sunlight would be displaced in traveling a distance d in the vacuum of space, the following transfer matrix was used:

$$T = \begin{bmatrix} 1 & 0 \\ d & 1 \end{bmatrix} \quad \text{(Ref 24:175)} \tag{32}$$

The matrix representing the combined effect of all of the lenses and mirrors in an optical system is called the system matrix (Ref 24:174). The following system matrix describes the path of a ray from the primary mirror to the tertiary mirror of the AOSR system:

$$Z = \begin{bmatrix} 1 & 0 \\ d_2 & 1 \end{bmatrix} \begin{bmatrix} 1 & -\frac{1}{f_2} \\ 0 & 1 \end{bmatrix} \begin{bmatrix} 1 & 0 \\ d_1 & 1 \end{bmatrix} \begin{bmatrix} 1 & -\frac{1}{f_1} \\ 0 & 1 \end{bmatrix}. \tag{33}$$

where f_1 is the focal length of the primary mirror, f_2 is the focal length of the secondary mirror, d_1 is the distance between the primary and secondary mirrors, and d_2 is the distance between the secondary and tertiary mirrors.

Values used in Eq (33) are listed in Table XVI. These values were calculated based upon three AOSR design factors: that d_1 = 300 m (Ref 22:16); that the primary and secondary mirrors would form

Table XVI
AOSR Design Parameters[1]

Parameter	Size (m)
f_1	336.73
f_2	-36.73
d_1	300.00
d_2	279.79

[1]Based on Ref 22.

a confocal system; and that the primary and secondary mirrors would have diameters of 330 m and 36 m, respectively (Ref 22:9, 15).

One additional factor was included in order to maximize the efficiency of the AOSR. By efficiency is meant the intensity of the sunlight leaving the tertiary mirror for the Earth, divided by the intensity of the sunlight incident on the primary mirror. This additional factor was that the marginal ray of the system should follow the line $\overline{ABC}$ in Fig. 15. If reflectivity losses are neglected, then under such conditions, all collimated light incident on the primary mirror would strike the secondary mirror. Yet, at the same time, the secondary mirror would experience the lowest radiation flux density possible, given the requirement that no light miss it.

Using these factors a geometrical configuration similar to that shown in Fig. 8 was constructed. Both mirrors were assumed to be parabolic such that the equation

$$y^2 = 4fx \tag{34}$$

was applicable. In Eq (34), y is the distance from the optical axis to the outer edge of the mirror, f is the focal length of the mirror, and x is the distance from the center of the mirror, measured along the optical axis, to the point from which the value for y is measured (Ref 40:38).

The actual values in Table XVI were calculated using the property of similar triangles that

$$\frac{\overline{AD}}{\overline{DC}} = \frac{\overline{BF}}{\overline{FC}} \tag{35}$$

where $\overline{AD}$, $\overline{BF}$, $\overline{DC}$, and $\overline{FC}$ are as defined in Fig. 15. Using Eq (35) it was possible to solve for $\overline{FC}$ and then to use Eqs (36) and (37) to find f_1, and f_2 where

$$f_1 \approx \overline{GE} + \overline{FC} \tag{36}$$

$$f_2 \approx \overline{FC} \tag{37}$$

Although approximate, the above calculations provided a first approximation so that Eq (34) could be used to calculate x for both the primary and secondary mirrors. These calculations gave first approximations for $\overline{GD}$ and $\overline{EF}$. Then $f_1 = \overline{GC}$ and $f_2 = \overline{EC}$ were computed, $\overline{DF}$ and $\overline{FC}$ were recomputed, and Eq (35) was applied a second time. This led to another value for $\overline{FC}$ which was used to recompute f_1 and f_2 which were then used in Eq. (34) to recalculate $\overline{GD}$ and $\overline{EF}$. The procedure described in the last two sentences was continued in an iterative fashion until recalculation did not change the value of $\overline{FC}$. The values for f_1, f_2, d_1, and d_2, when $\overline{FC}$ remained constant, are those listed in Table XVI.

When the values listed in Table XVI were substituted into Eq. (33), the following system matrix was obtained

$$Z = \begin{bmatrix} 9.1677 & 0 \\ 2865.0 & 0.10908 \end{bmatrix} \tag{38}$$

In order to complete the ray trace mentioned earlier, the tertiary mirror was assumed to be planar, and the initial ray striking the primary mirror was assumed to be, in matrix form,

$$\begin{bmatrix} 0.00466 \\ 175.54 \end{bmatrix} \tag{39}$$

The value 175.54 in Eq. (39) represents the radius of the primary mirror projected into its respective plane, as discussed earlier. This value for y in Eq. (10) represents the point A, of the marginal ray $\overline{ABC}$. (The radius of the secondary mirror projected into its respective plane would be 19.15 m.)

The value 0.00466 was selected for Eq (39) because it represents the slope of the ray having the greatest divergence at the tertiary mirror.

To completely trace this ray through the system, the individual matrices in Eq. (33) were applied consecutively until the ray left the AOSR. For example, as a first step, the effect of reflection at the primary mirror was calculated using

$$\begin{bmatrix} \alpha_1 \\ \\ y_1 \end{bmatrix} = \begin{bmatrix} 1 & \dfrac{-1}{336.73} \\ \\ 0 & 1 \end{bmatrix} \begin{bmatrix} .00466 \\ \\ 175.54 \end{bmatrix} \tag{40}$$

At the secondary mirror, the ray would undergo a reflection given by

$$\begin{bmatrix} \alpha_2 \\ \\ y_2 \end{bmatrix} = \begin{bmatrix} 1 & 0 \\ \\ 300 & 1 \end{bmatrix} \begin{bmatrix} \alpha_1 \\ \\ y_1 \end{bmatrix} \tag{41}$$

However, Eq. (41) is itself an indication that sunlight is not parallel. Although the AOSR system was configured such that all rays from the primary mirror would strike the secondary mirror, the solution of Eq. (41) is

$$\begin{bmatrix} \alpha_2 \\ \\ y_2 \end{bmatrix} = \begin{bmatrix} -0.5166 \\ \\ 20.54 \end{bmatrix} \tag{42}$$

This ray would miss the secondary mirror. Thus, from Eq. (42) it can be concluded that because sunlight is not parallel radiation, some of the sunlight striking the AOSR primary mirror would miss the secondary mirror. In fact, for the rays with slope of 0.00466, the outermost ray to strike the secondary mirror would be the ray

$$\begin{bmatrix} 0.00466 \\ 162.75 \end{bmatrix} \tag{43}$$

It is this ray that is traced in Fig. 8 by consecutively applying the individual matrices of Eq. (33).

The details of the ray trace are as follows. Eq. (44) mathematically describes the result of the initial reflection at the primary mirror. The numbers in Eq. (44) and the others below were rounded for presentation. In the actual calculations, however, they were not rounded.

$$\begin{bmatrix} 1 & -1/336.73 \\ 0 & 1 \end{bmatrix} \begin{bmatrix} 0.00466 \\ 162.75 \end{bmatrix} = \begin{bmatrix} -0.4787 \\ 162.75 \end{bmatrix} \tag{44}$$

After reflection at the primary mirror, the light ray travels to the secondary mirror as described by Eq. (45).

$$\begin{bmatrix} 1 & 0 \\ 300 & 1 \end{bmatrix} \begin{bmatrix} -0.4787 \\ 162.75 \end{bmatrix} = \begin{bmatrix} -0.4787 \\ 19.150 \end{bmatrix} \tag{45}$$

Then the ray is reflected by the secondary mirror.

$$\begin{bmatrix} 1 & 1/36.73 \\ 0 & 1 \end{bmatrix} \begin{bmatrix} -0.4787 \\ 19.150 \end{bmatrix} = \begin{bmatrix} 0.0427 \\ 19.150 \end{bmatrix} \tag{46}$$

After reflection at the secondary mirror, the light ray travels toward the tertiary mirror, but misses it, and instead strikes the primary mirror a second time.

$$\begin{bmatrix} 1 & 0 \\ 300 & 1 \end{bmatrix} \begin{bmatrix} 0.0427 \\ 19.150 \end{bmatrix} = \begin{bmatrix} 0.0427 \\ 31.966 \end{bmatrix} \tag{47}$$

Eq. (48) describes the result of a second reflection at the primary mirror.

$$\begin{bmatrix} 1 & -1/336.73 \\ 0 & 1 \end{bmatrix} \begin{bmatrix} 0.0427 \\ 31.966 \end{bmatrix} = \begin{bmatrix} -0.0522 \\ 31.966 \end{bmatrix} \tag{48}$$

The ray then travels to the secondary mirror a second time.

$$\begin{bmatrix} 1 & 0 \\ 300 & 1 \end{bmatrix} \begin{bmatrix} -0.0522 \\ 31.966 \end{bmatrix} = \begin{bmatrix} -0.0522 \\ 16.303 \end{bmatrix} \tag{49}$$

The reflection at the secondary mirror is described by Eq. (50).

$$\begin{bmatrix} 1 & 1/36.73 \\ 0 & 1 \end{bmatrix} \begin{bmatrix} -0.0522 \\ 16.303 \end{bmatrix} = \begin{bmatrix} 0.3916 \\ 16.303 \end{bmatrix} \tag{50}$$

The light ray then returns to the primary mirror a third time. At its arrival at the primary mirror it is governed by

$$\begin{bmatrix} 1 & 0 \\ 300 & 1 \end{bmatrix} \begin{bmatrix} 0.3916 \\ 16.303 \end{bmatrix} = \begin{bmatrix} 0.3916 \\ 133.801 \end{bmatrix} \tag{51}$$

Eq. (52) describes the third primary mirror reflection.

$$\begin{bmatrix} 1 & -1/336.73 \\ 0 & 1 \end{bmatrix} \begin{bmatrix} 0.3916 \\ 133.801 \end{bmatrix} = \begin{bmatrix} -0.0057 \\ 133.801 \end{bmatrix} \tag{52}$$

Following this third primary mirror reflection, the light ray departs the AOSR, heading in the general direction of the Sun. At a distance of 300 m from the primary mirror, the ray would be as calculated in Eq.(53).

$$\begin{bmatrix} 1 & 0 \\ 300 & 1 \end{bmatrix} \begin{bmatrix} -0.0057 \\ 133.801 \end{bmatrix} = \begin{bmatrix} -0.0057 \\ 132.092 \end{bmatrix} \tag{53}$$

Appendix B

Calculation of the Luminous Incidence from the Sun at 1 AU

Only a portion of the electromagnetic radiation emitted by the Sun is in the visible spectrum. Thus, while the solar constant, Eq (15), is a measure of the total radiation received from the Sun at 1 AU, it does not identify how much of that radiation is in the visible spectrum. The purpose of this appendix is to explain how the illuminance of the Sun at 1 AU, Eq (21), was calculated.

The conversion between radiometric and photometric flux is, by convention, 673 lumen/watt at 555 nm. At all other wavelengths, the conversion is given by

$$K = (y_\lambda)(673 \text{ lumen/watt}) \tag{54}$$

where y_λ is the luminous efficiency.

The solar constant is made up of finite contributions from all wavelengths in the solar spectrum. If the contribution from a given wavelength is H_λ, then the equation to be used in converting from radiometric to photometric units would be:

$$P_v = (H_\lambda)(K) \tag{55}$$

Two tables were useful in determining the total illuminance of the Sun. In each table, the visible portion of the spectrum is divided into intervals of 5 nm and 10 nm. Table 16-1b, of reference 42, lists $H_{\Delta\lambda}$ over each of these intervals; while Table 6j-1, of reference 23, lists y_{λ} for the wavelength beginning each interval.

In order to do the calculation of total illuminance, it was assumed that

$$H_{\Delta\lambda} \approx H_{\lambda} \tag{56}$$

Although Eq (56) is an approximation, it was felt that the wavelength intervals used in the tables were sufficiently small that the calculation of the total illuminance, using this approximation, would be sufficiently accurate for this thesis.

To calculate the total illuminance, P_v was calculated for each $K \neq 0$ using Eq (55). The total illuminance was then the sum of all P_v's. When this procedure was followed, the total illuminance of the Sun at 1 AU, outside the Earth's atmosphere, was found to be 1.37×10^5 lumen/m^2.

Appendix C

Potential Designs for Large Space Mirrors

One of the purposes of this thesis was to investigate the use of orbiting solar reflectors. In addition to studying the applicability of using reflectors to reflect sunlight directly to the Earth's surface, as discussed in Chapter III of this thesis, a study was also conducted concerning possible mirror designs. Since the purpose of such mirrors would be to reflect large quantities of sunlight, the design study concentrated on mirrors with maximum size to mass ratios. The purpose of this appendix is to describe the types of mirrors considered in the study and to point out their respective advantages and disadvantages in space applications.

Common Characteristics

All of the designs considered were for aluminized kapton reflectors, 330 m in diameter. Most would be parabolic in shape, although a spherical mirror would be a reasonable approximation for most applications. For example, for a parabolic mirror with a focal length of $f = 336.73$ m, the (x, y) cross sectional coordinates of the outer edge would be (165 m, 20.21 m), as determined by solving Eq (34). Similarly, the outer edge coordinates of a spherical mirror with a focal length of f, would be (165 m, 20.53 m) as derived from the equation for the circular cross section of a spherical mirror

$$y^2 = R^2 - (x - R)^2 \tag{55}$$

where R is the radius of curvature such that

$$R = 2f \tag{56}$$

(Ref 24:124, 125).

After construction, each of the mirrors considered in this study would be packaged using a radial folding technique described by the Goodyear Aerospace Corporation in a final report for the NASA Project Able (Ref 21:IV-7). Once in orbit, each mirror would be partially unfolded. This would release the elastic energy stored during the folding process. Following this unfolding, four equidistant points along the outer rim of each reflector would be moved as nearly as possible into their final positions. The final deployment sequence would then be accomplished by gradually inflating the outer rim until it becomes rigid (Ref 21:IV-8, 9).

In order to accommodate the deployment process described above, the rims of the mirrors would be made of either a wire-film truss, or a pressurized torus, as shown in Fig. 16. Triangular in shape, the truss would be made of an aluminum wire mesh covered with 0.35 mil kapton film on both sides (Ref 21:III-7). During deployment, a gas would be applied to pressurize the wire-film tubes, forcing them to take a predetermined shape. During the last part of this

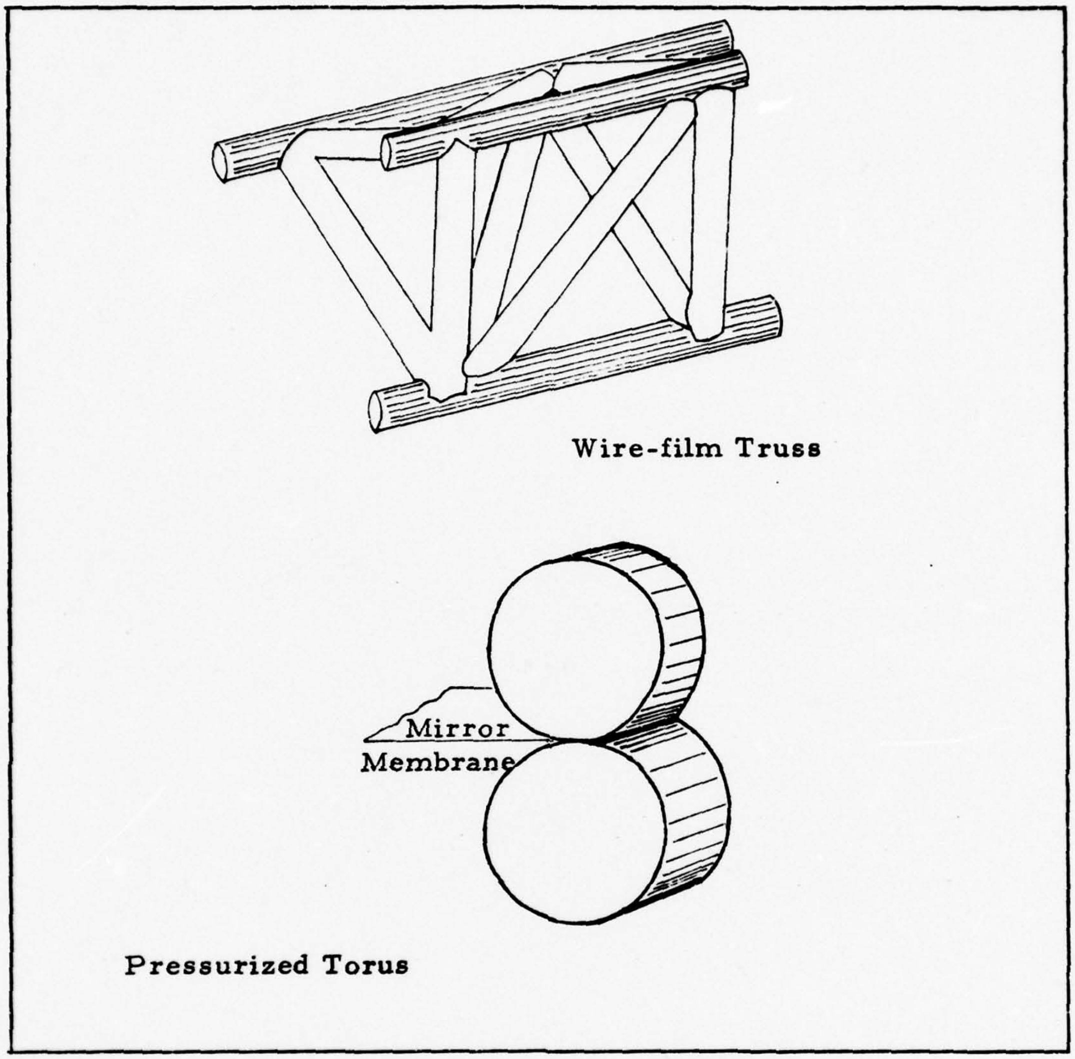

Fig. 16. Two Mirror Rim Designs
(Based on Ref 21:III-18, III-20)

pressurization, the pressure would stress the wire-mesh part of the tubes slightly beyond their yield point. This would permanently set their shape. Once this shape had been set, the rim would maintain its shape and rigidity without the need of internal pressurization (Ref 21:III-19, III-20). In addition, even after the gas is vented, the kapton film would continue to provide shear stiffness for the wire grid

(Ref 21:IV-12).

A pressurized torus would be made of the same material as the reflector, most probably aluminized kapton. This would eliminate thermal expansion problems associated with dissimilar materials. The torus would resemble a figure eight in cross section (Ref 21:III-20).

There are both advantages and disadvantages to a pressurized torus and a wire-film truss. The wire-film truss would be relatively unaffected by meteoroids since it would not require constant pressurization for rigidity. However, Goodyear found that the truss design could not withstand as much stress as could the pressurized torus (Ref 21:X-2, IV-11). This could become a deciding factor when working with mirrors that are 330 m in diameter.

On the other hand, the pressurized torus would be very susceptible to meteoroid punctures. For example, Goodyear conducted a design study of a flat reflector, 2250 ft in diameter, made of 0.35 mil aluminized kapton. To support the mirror, Goodyear concluded that the diameter of each of the two sections in the torus rim would have to be 43 feet. Goodyear suggested that a subliming material such as sulfur, along with a powder, such as carbon black, should be dusted into the torus during fabrication. These materials would help to seal micrometeroid punctures. Yet, even with these materials, Goodyear estimated that 150 lb of gas would be needed to inflate the torus and that within one year, 350 lb of gas would be lost between the time a puncture developed and was sealed, and 1120 lb of gas would escape through openings too large for the particles to plug (Ref 21:III-20—III-22).

The ultimate goal in developing a rim design would be to provide adequate tension while minimizing rim mass. Depending on the particular mirror design, it may be possible to use a wire-film concept. However, even if the wire-film concept is not feasible, it may still be possible to avoid the large gas losses associated with the pressurized torus design. This could be done by pressurizing a torus rim and then chemically treating it with a rigidizing process such as one of those suggested by the Itek Corporation in their feasibility study of a 30 m space mirror. Itek categorized these techniques as "plasticizer boil-off, ultraviolet and infrared cured plastic resins, and gas catalysis curing techniques [Ref 4:62]." Finally, as another alternative to pressurization, it might be possible to fill the torus with a polyurethane or epoxy foam which, after filling the torus, would solidify. However, in a study by the Goodyear Aerospace Corporation, it was concluded that for mirrors larger than 10 or 15 feet in diameter, a chemical discovery would be necessary to make foam rigidization work. Such a chemical would have to remain fluid for at least 3 to 10 minutes but would then have to foam and gelatin within a matter of seconds (Ref 41:45). In addition, without some sort of breakthrough, foams may prove to be too heavy. For example, at 75°F, a rigid polyurethane foam with a density of 2 lbs/ft^3 has a compressive strength of only 36 psi $\pm$ 10% parallel to the foam rise and a shear strength of 23 psi $\pm$ 10% parallel to the foam rise. To compound the problems, strength decreases as density decreases and compression strength perpendicular to foam rise is only approximately half that parallel to foam rise (Ref 27:70-71).

Possible Mirror Designs

In this appendix, the following mirror designs will be discussed: radiation pressure inflated, electrostatically inflated, gas filled, foam supported, catenary supported, and solid back Fresnel. A sketch of each is included in Fig. 17.

Radiation Pressure Inflated Mirror

One of the first mirrors considered was one which would use the force of solar radiation pressure to maintain its shape. This concept is very attractive because it would require less support equipment than the other concepts.

The general equation for stress on a continually inflated structure is given by

$$P = \frac{\sigma_x}{R_x} + \frac{\sigma_y}{R_y} \tag{57}$$

where P is the pressure, R_x and R_y are the principal radii of curvature, and σ_x and σ_y are the principal stresses. For a sphere, where $R_x = R_y$, the general formula reduces to

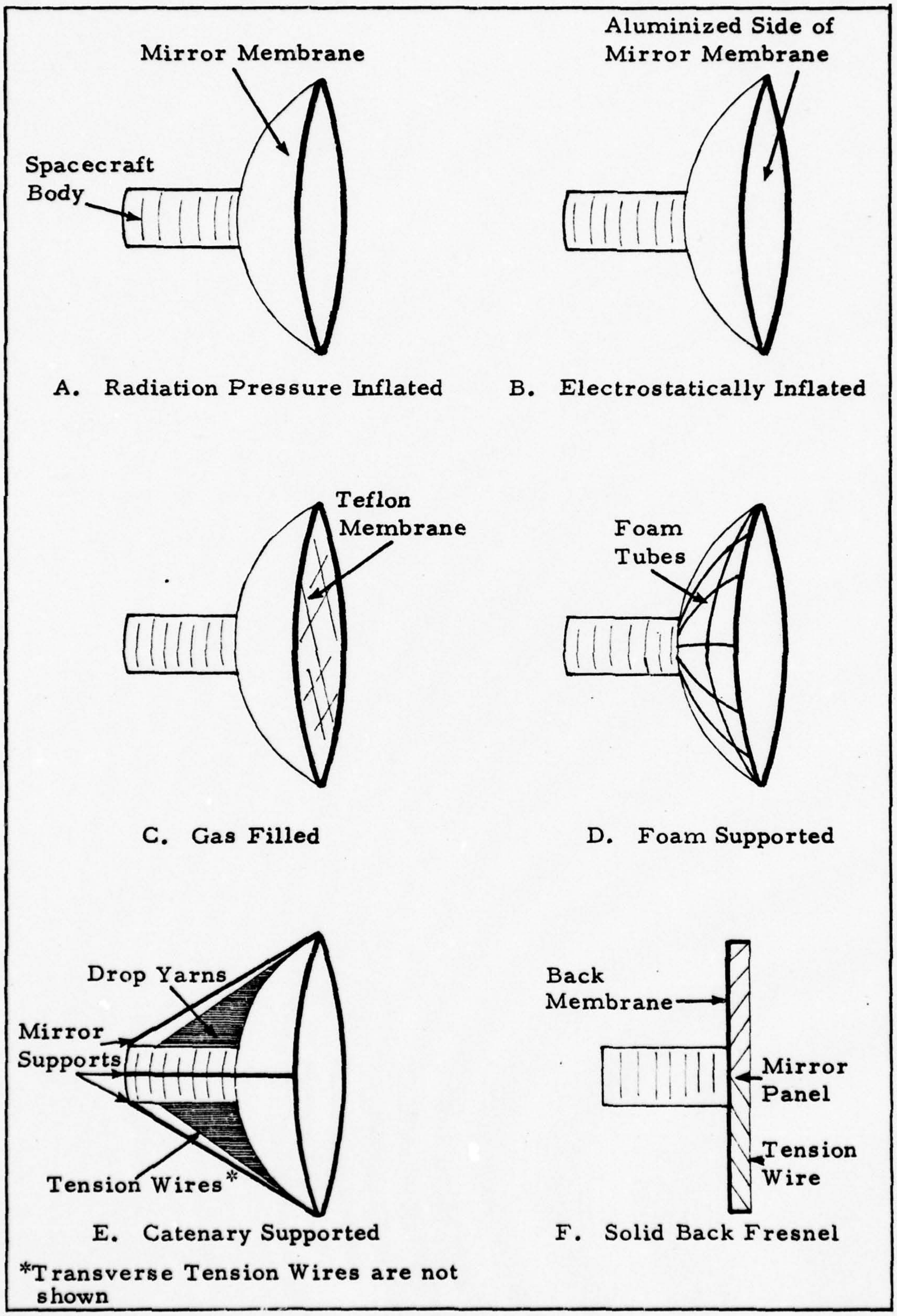

Fig. 17. Mirror Designs

$$\sigma = \frac{PR}{2} \tag{58}$$

(Ref 35:238).

In applying Eq. (58), the solar pressure at 1 AU was assumed to be 0.9×10^{-5} N/m^2 (Ref 44:1). The radius of curvature was approximated as R = 165 m since the solar pressure would act only over the 330 m cross section of the mirror. Substituting these values into Eq. (58), it was determined that

$$\sigma = 7.425 \times 10^{-4} \text{ N/m} \tag{59}$$

Thus, if solar pressure is to be sufficient to inflate the mirror, then a stress of 7.425×10^{-4} N/m must be sufficient to remove all wrinkles from the aluminized kapton. To determine if this level of stress would be sufficient to remove all wrinkles, a simple experiment was devised.

In the environment of space, the orbiting reflector would be essentially weightless. But on Earth, it is possible to use the force of gravity to approximate the effect of solar pressure. Newton's equation

$$F = ma \tag{60}$$

was used to convert 7.425×10^{-4} N/m to 7.57×10^{-5} g/mm, where a, the acceleration due to gravity, was assumed to be 9.81 m/sec^2.

In the experiment, a 20 mm by 10 mm piece of 0.1 mil aluminized mylar, supplied by Sheldahl of Northfield, Minnesota, was attached to a glass rod as shown in Fig. 18. A fold was made 5 mm from the bottom of the mylar and three holes were made along the bottom of the mylar as shown in Fig. 18. The density of the 0.1 mil mylar used, was 3.4×10^{-6} g/mm^2. Thus, since 100 mm^2 of mylar hung below the fold, the mass of the mylar below the fold was 3.4×10^{-4} g. This meant that solar pressure would not be sufficient to inflate the mirror, unless a mass of

$$(7.57 \times 10^{-5}\ \text{g/mm})(20\ \text{mm}) - (3.4 \times 10^{-4}\ \text{g}) = 1.174 \times 10^{-3}\text{g} \qquad (61)$$

would be sufficient to completely remove the fold from the mylar.

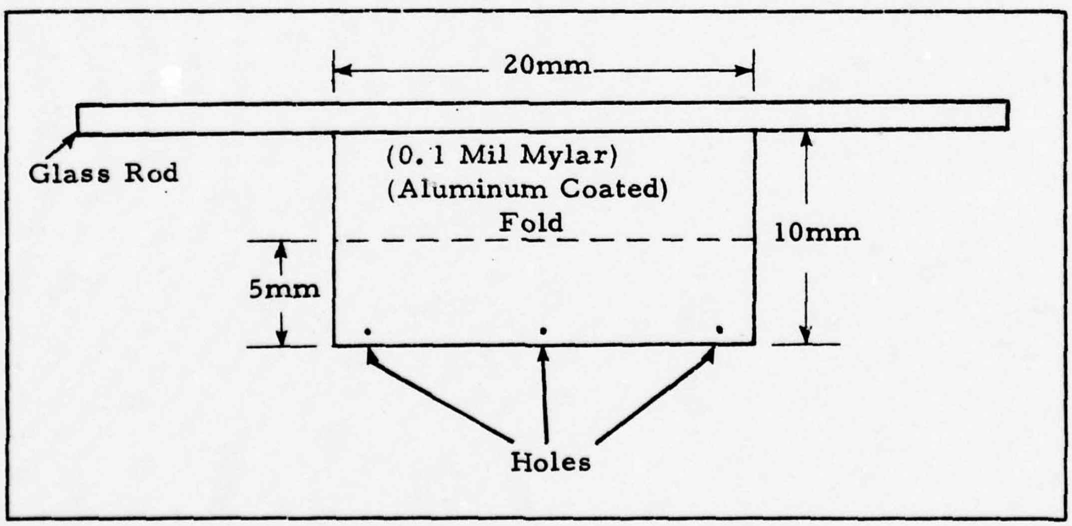

Fig. 18. Experimental Setup for Solar Pressure Experiment

In the experiment, aluminum wire was hung as uniformly as possible, from the three holes in the mylar film. It took 1.427 grams of aluminum wire to take most of the wrinkles out and much more than that to come close to taking out all of the wrinkles.

Therefore, it was concluded that solar pressure would not be sufficient to remove all of the wrinkles and properly inflate a 330 m diameter mirror.

Electrostatically Inflated Mirror

One of the problems encountered in trying to determine if solar radiation pressure would inflate a large space mirror was the electrostatic potential that built up on the mylar any time it was handled. This led to a hypothesis that the mirror could be properly inflated by building up a large electrostatic charge on it. It was theorized that the charged film would repel itself and thereby take its manufactured shape.

To test the electrostatic hypothesis, a miniature spherical mirror was constructed using a cardboard rim and 0.1 mil aluminized mylar. A Welch winshurst generator was used to generate the electrostatic potential.

During the experiment, the aluminized mylar was wrinkled and its spherical shape was distorted. Then the generator was cranked until a charge had built up on the miniature mirror. In all cases, when the experiment was conducted in air, the mylar film did take on a more spherical shape. However, it was not possible, with the equipment available, to build up enough charge to remove all wrinkles from the miniature mirror surface.

However, when the experiment described above was done in an evacuated bell jar to more closely simulate the space environment, it was not possible to build up enough charge on the mirror surface to cause it to even slightly change shape. This occurred because too much of the charge continually leaked off to the evacuation system. Time did not permit further investigation of this concept. Therefore, it may deserve continued consideration because should it prove feasible, it would offer a lightweight method of inflating large space mirrors. Additional consideration should also include a study of the affects on the mirrors of charged particles that the mirrors would encounter in space.

The Space Division of Rockwell International has proposed a similar concept. They proposed that a large wire net be constructed and that the electrostatic attraction of the membrane to the adjacent wire net be used to inflate the membrane (Ref 39:10).

Gas Filled Mirror

Rockwell International in a proposal to NASA has also suggested that a mirror could be made which would use gas pressure to maintain its shape. The gas filled mirror, as envisioned by Rockwell, would be constructed of metallized kapton but would, in addition, have a piece of 0.5 mil transparent FEP Teflon stretched across the rim like the top of a drum. The gas would be placed between the teflon and the metallized side of the mirror, at a pressure of at least 4 x 10^{-5} N/m^2 (Ref 39:12).

The obvious disadvantage of this type of system is that the sunlight must pass through the gas twice and through the teflon membrane twice. In addition, micrometeoroid damage would be a problem just as it was for the pressurized torus rim described earlier.

The main advantage of a gas filled mirror over the other mirrors described in the remainder of this appendix, is that it could more closely approximate a perfect parabola or sphere. Whether this advantage would offset the need for at least 1000 kg/yr of makeup gas Rockwell predicts would be necessary to keep the system inflated because of meteoroid damage, has yet to be demonstrated.

The last three mirror designs to be discussed have not been considered experimentally but are included in the event that they may prove to be of value in initiating further research in this field.

Foam Supported Mirror

In the case of a foam supported mirror, small tubes of kapton would be secured in the seams of the aluminized kapton membrane. They would be so designed, that when inflated, they would bulge on the non-aluminized side of the membrane and cause the membrane to assume a nearly parabolic or spherical shape. Tubes would extend radially from the center of the mirror to the rim as well as transversely.

As mentioned earlier, finding an acceptable foam could be a problem. Additionally, the fact that the mirror would not be perfectly spherical or parabolic between the foam tubes, may prove troublesome. Although this latter problem can be reduced by putting the tubes closer together, this would at the same time increase the total mass of the mirror. Thus, in order to determine the optimum spacing of the tubes it would be necessary to do a design study to determine the reflectivity as a function of tube spacing.

<u>Catenary Supported Mirror</u>

The catenary supported mirror would have problems similar to those of the foam supported mirror. However, it would not require foam injection through small tubes over long distances, a problem which may doom the foam supported concept. In the case of a catenary supported mirror, tension wires would be attached between the spacecraft body and the rim. Drop yarns would then be attached between the aluminized kapton membrane and the tension wires. The drop yarns would be made of varying lengths such that when the tension wires are in place, the mirror membrane would be forced into its proper shape. As with the foam supported mirror, both radial and transverse tension wires could be required. All of the drop yarns of a transverse tension wire would be the same length. The transverse wires are not shown in Fig. 17.

As with the foam supported mirror, some reflectivity would be lost because the mirror membrane would not be perfectly parabolic or spherical between drop yarn attachment points. Also, it appears that this method would exert greater tension on the outer rim of the mirror. As with the foam concept, further study would be required to evaluate the stresses involved and to determine the optimum spacing of the drop yarn attachment points.

Solid Back Fresnel Mirror

The final mirror design concept to be discussed is the solid back Fresnel mirror concept. This concept is based on the same principles as those for a Fresnel lens. As shown in Fig. 17, a flat and rather sturdy back membrane would be stretched tightly within the rim of the mirror. To this back membrane would be attached rings of flat panels of aluminized kapton of varying widths. The widths would vary so that the outer edge of all panels would be an identical distance from the back membrane. Wires would extend from the central hub of the mirror to the outer rim and would be attached to the outer edge of each panel. These wires would be a constant height above the back membrane and would provide the rigidity necessary to maintain the proper panel configuration.

The geometry of the solid back Fresnel mirror is somewhat simpler than the geometry of the previous concepts. However, it is not clear if the mass of the back membrane and the increased stress on the rim caused by the back membrane and wires would negate any mass savings obtained by not having to use foam or drop yarns. Other possible disadvantages are that the wires would block some sunlight and that the flat panels would only approximate a parabolic surface, as in the case of foam and catenary supported mirrors. Just what effect this approximation and the shadowing by the wires would have on the reflectivity of the mirror remains to be determined. It is also possible that the Fresnel mirror would not dissipate absorbed sunlight as easily as the other concepts and may therefore have to operate at a higher temperature.

The final disadvantage of the Fresnel mirror design is that a portion of the sunlight reflected by a given panel would be intercepted by the adjacent inner panel. To determine what percentage of the reflecting surface is essentially lost because of this effect, the following computer analysis was made. To simplify the analysis, only incident radiation perpendicular to the back membrane was considered.

The first step in the computer analysis was to determine the distance between the outer edge of the panels and the back membrane. A value of 13 cm was arbitrarily selected. Figure 19 illustrates the geometry of this problem. Since all incident radiation was assumed to be perpendicular to the back membrane, all incident rays striking a given panel between points J and Q would be intercepted by the adjacent inner panel. These intercepted rays can be thought of as lost

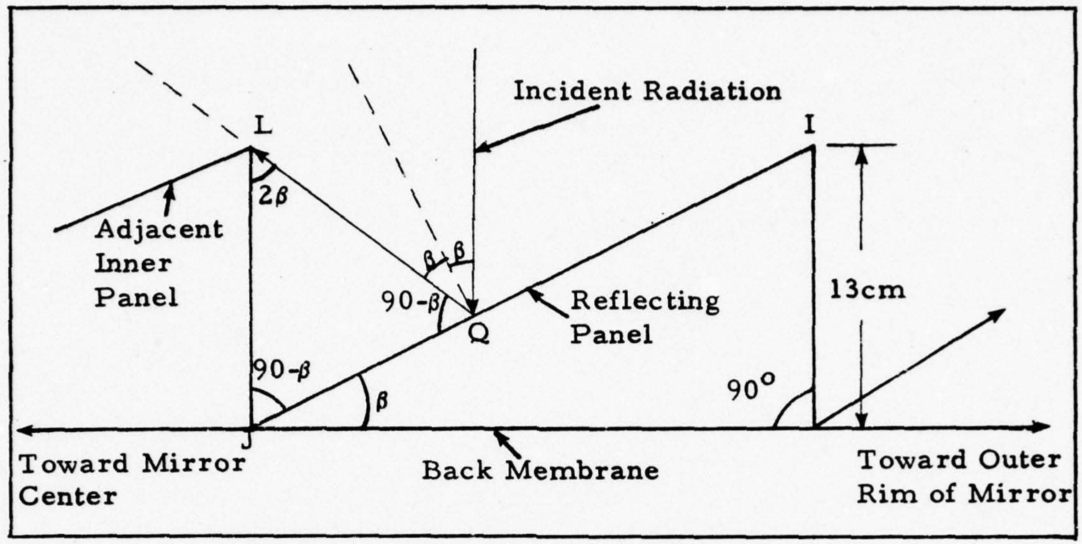

Fig. 19. Geometry for Fresnel Mirror

since they represent a decrease in the reflectivity of the mirror surface. By the law of sines,

$$\overline{JQ} = \frac{(\overline{JL})(\sin 2\beta)}{\sin (90-\beta)} \tag{62}$$

where the lengths and angles are as specified in Fig. 19.

The next step in the analysis was to determine the surface area effectively lost as a result of intercepted reflected light. This was done by considering the length lost, $\overline{JQ}$, as the lateral surface of a cone. Then the surface area lost on a given panel would be

$$\text{Area Lost} = \pi(\overline{OJ} + \overline{OQ})\overline{JQ} \tag{63}$$

where $\overline{OJ}$ and $\overline{OQ}$ are the distances from the center of the mirror to the points J and Q, respectively. To simplify Eq. (63), $\overline{OQ}$ was written as

$$\overline{OQ} = \overline{OJ} + \overline{JQ} \cos \beta \qquad (64)$$

For panels with outer edges 13 cm from the back membrane, a 330 m diameter spherical mirror with a radius of curvature of 668.48 m, would have approximately 159 panels. Therefore, combining Eqs. (63) and (64), the total mirror surface area lost as the result of intercepted sunlight would be

$$\text{Total Area Lost} = \sum_{J=1}^{159} \pi [2(\overline{OJ}) + (\overline{JQ}) \cos \beta] (\overline{JQ}) \qquad (65)$$

When a computer program was generated to solve Eq. (65), it was found that the

$$\text{Total Area Lost} = 5364.4 \text{ m}^2 \qquad (66)$$

For comparison, the total surface area of the mirror, as determined by using the equation for the surface area of a spherical cap, would be

$$\text{Total Area} \approx 2\pi R x \qquad (67)$$

where R is the radius of curvature of the cap and x is the cross sectional height of the cap (Ref 40:9). For a mirror of height 20.68 m and radius of curvature equal to 668.48 m, Eq. (67) yields

$$\text{Total Area} = 86,859.8 \text{ m}^2 \qquad (68)$$

As a final step in the computer analysis, the percentage of the total surface area lost was determined by dividing the total area lost in Eq. (66) by the total area in Eq. (68). From this calculation it was determined that 6.18% of the surface area of the mirror would be useless because of interception of reflected sunlight by the adjacent inner panel.

It should be noted that a more precise calculation would take into consideration the affect of sunlight incident at other angles of incidence. However, since the incident sunlight would be equally distributed through an arc of 0.5°, there should be roughly as many rays of sunlight intercepted after striking a given panel between points Q and I of Fig. 19, as there would be rays not intercepted which were headed for points between J and Q of Fig. 19. Therefore, the result of a more precise calculation would be roughly the same.

Whether any of the mirror designs presented in this appendix will prove to be of use in future space applications remains to be demonstrated. Perhaps in the long run, large parabolic or spherical space mirrors will prove to be less desirable than techniques which employ several smaller mirrors. However, it is hoped that should additional work be done in this field, that the suggested designs described in this appendix will serve to suggest feasible alternatives.

Appendix D

Space Transportation System

The purpose of this appendix is to describe the space transportation system used in the analysis in Chapter IV of this thesis. The transportation system would be used to cheaply and efficiently transport all space related materials for the OSR and SSP systems from the Earth to various Earth orbits.

This appendix is divided into three sections. The first section consists of a description of the various vehicles to be used in the space transportation system. The second section contains the SSP operations scenario and details of the calculations relative to the SSP transportation system. The third section is similar to the second except that it applies to the OSR system.

Transportation System Vehicles

The transportation system would utilize four types of space vehicles: a heavy lift launch vehicle (HLLV), a personnel and priority cargo launch vehicle (PLV), a cargo orbital transfer vehicle (COTV), and a personnel orbital transfer vehicle (POTV). In effect, however, there would actually be five vehicles because the COTV system would use two vehicles: the $COTV_L$ and the $COTV_G$.

The labels $COTV_L$ and $COTV_G$ originated with NASA studies of the SSP system. Two options were considered by NASA in determining the SSP construction scenario. Either the space portion of the SSP could be constructed in low Earth orbit (LEO) and then transported to geosynchronous orbit (GEO) when nearly completed, or the space portion of the SSP could be constructed entirely in GEO.

If constructed in LEO, the $COTV_L$ vehicles would be used for the orbital transfer to GEO. To accomplish this, the $COTV_L$ vehicle would have to be capable of transporting large payloads, although the transfer could take several months (Ref 32:12-6).

If constructed in GEO, long transfer times would be unacceptable because they would delay construction. In addition, GEO construction would not require the transportation of large payloads from LEO to GEO. For this type of construction scenario, the $COTV_G$ was designed.

Results of studies by NASA indicated that even in a LEO construction scenario, both vehicles would have a role. The studies also demonstrated that construction in LEO would be more economical (Ref 33:VI-18). The uses of the $COTV_L$ and $COTV_G$ in LEO construction will be explained more fully in later portions of this appendix.

Table XVII contains a list of the major features of each of the proposed transportation system vehicles. The HLLV vehicle described in Table XVII is a ballistic single stage-to-orbit (SSTO) vehicle. In considering HLLVs, NASA evaluated the potential of three types of

Table XVII
Space Transportation Vehicles

Characteristic	HLLV	PLV	$COTV_L$	$COTV_G$	POTV
Payload (Metric Tons)	221[1]	36[2]	1000[3]	250[3]	20[4]
Payload (Passengers)	-	50[2]	-	-	75[5]
Propellant[6] (Type of Fuel)	LO_2/ LH_2	LO_2/ LH_2	Argon[7]	LO_2/ LH_2	LO_2/ LH_2
Propellant (Metric Tons/ Round Trip)	9304.5[8]	1892[2]	800[3]	475[3]	159[3]
Inert Mass (Metric Tons)	853.2[8]	256[2]	166[9]	35[3]	19[3]
Cost[10] ($\$10^6$/Flight)	9.6[1]	10.7[11]	32.1[3]	10.7[3]	12.8[3]
Lifetime[12] (Flights)	300[13]	100[14]	10[3]	30[3]	30[3]

[1]Ref 11:208, where 7 metric tons have been subtracted for mass of orbital maneuvering system (Ref 33:VI-2).

[2]Ref 33:VI-9. Payload can be either 36 metric tons or 50 passengers, but not both.

[3]Ref 33:VI-19.

[4]Ref 33:VI-17, but only from LEO to GEO.

[5]Ref 33:VI-17, while also carrying 20 metric tons of cargo.

[6]Assumption for this thesis was that all but the $COTV_L$ would use liquid oxygen/liquid hydrogen fuel (LO_2/LH_2). In actuality, the HLLV and PLV systems would use LO_2/LH_2 plus other fuels such as kerosene, hydrocarbon (Ref 33:VI-8), propane, and RP-1 (Ref

Table XVII (continued)

33:VI-3). However, the amount of these other fuels needed has not been specified.

[7]Ref 33:VI-10.

[8]Ref 34:Appendix IX, p 13, 13A. Inert mass includes 115 metric tons of water which is assumed expended after each flight.

[9]Ref 33:VI-19. Includes 110 metric tons expended per flight.

[10]Includes cost of vehicles, operations, and amortized spares/refurbishment adjusted from 1976 to 1977 dollars assuming a 7% inflation rate in 1976.

[11]Ref 33:VI-18.

[12]All vehicles are also limited to 10 year lifetimes, if that occurs before flight limits (Ref 32:6-4).

[13]Ref 33:VI-2.

[14]Ref 26:45 based on current space shuttle lifetime.

vehicles. These included the SSTO, a two-stage ballistic vehicle, and a two-stage winged vehicle.

For the transportation analysis described in this appendix, the SSTO vehicle was selected for two reasons. First of all, more data was available relative to the SSTO vehicle. Secondly, NASA, in a preliminary analysis, found that of the three HLLVs listed above, the SSTO vehicle would be of intermediate cost (Ref 33:VI-6).

The HLLV, shown in Fig. 20, would be used to transport all SSP and OSR materials, including all space support facilities, from

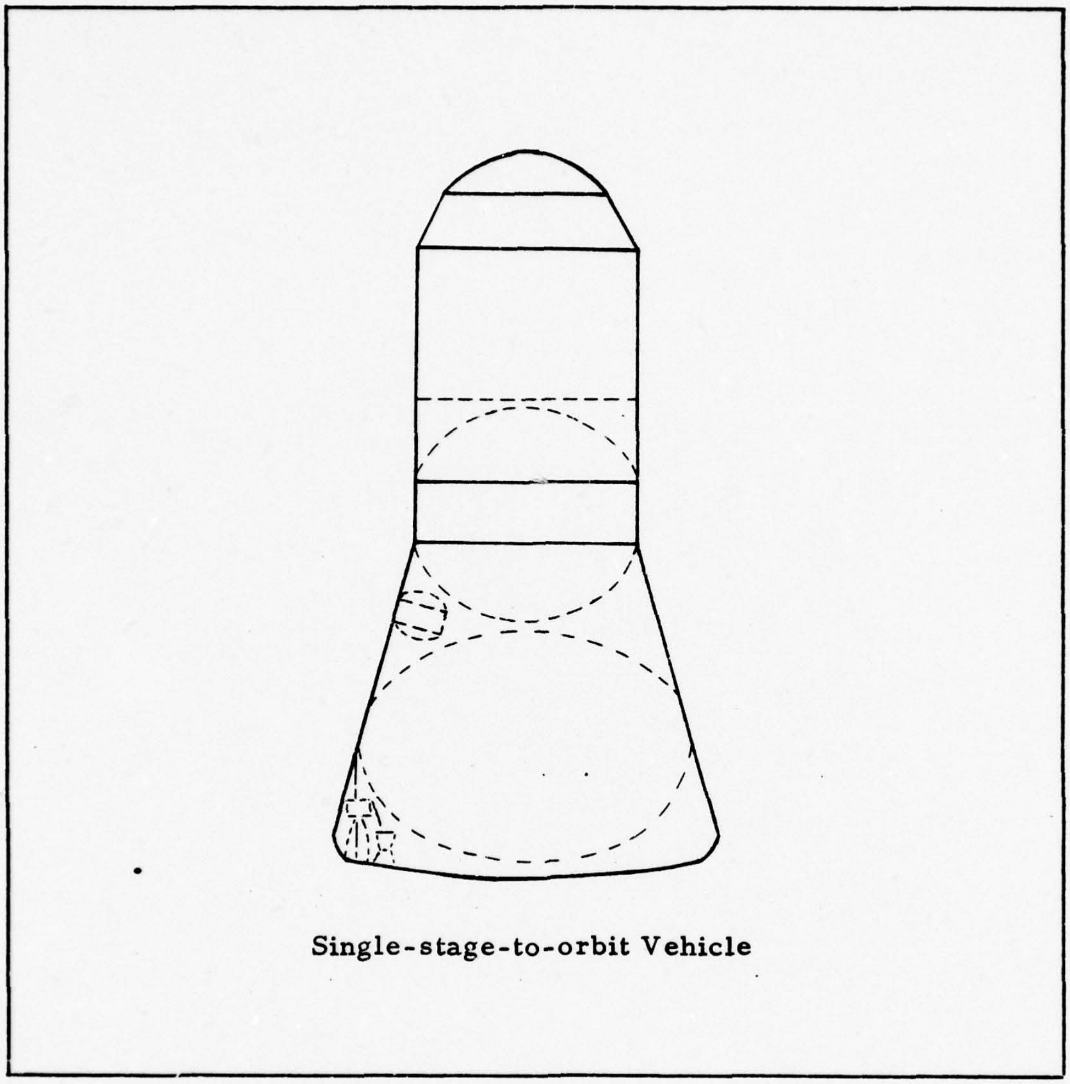

Fig. 20. Heavy Lift Launch Vehicle (Ref 32:12-9)

the Earth to LEO. Current specifications list the payload at 228 metric tons (Ref 11:208), of which 3% would be lost in order to carry the orbital maneuvering system (OMS) used for final HLLV rendezvous (Ref 33:VI-2). However, the Johnson Space Center has projected that the eventual nominal HLLV payload will be 700 metric tons (Ref 33: VI-18).

The PLV, shown in Fig. 21, would be used to transport all personnel from the Earth to LEO, although it could also be used to make high priority deliveries of small payloads. The design would be essentially a modified space shuttle (Ref 33:VI-8).

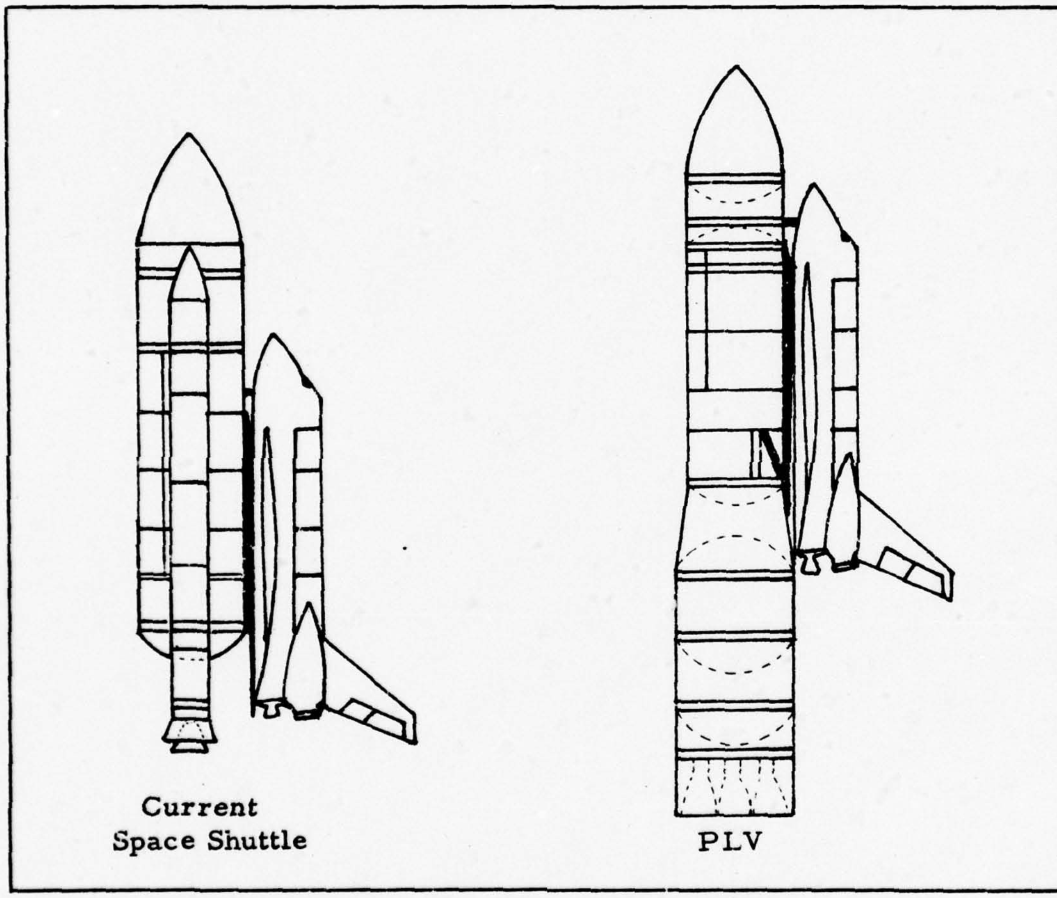

Fig. 21. The Current Space Shuttle Compared to the PLV (Based on Ref 33:11-9)

The $COTV_L$ would be a space-based system used to transport large payloads from LEO to GEO. It would be delivered to LEO by the HLLV. It would be unacceptable for use with the OSR system because its propulsion system requires its payload to supply electrical energy. A magnetoplasmadynamic arcjets propulsion system with argon fuel is tentatively being projected for use with the $COTV_L$ (Ref 33:VI-8, 9, 10).

The $COTV_G$, shown in Fig. 22, would not require electricity from its payload. Recent studies have indicated that an oxygen/hydrogen propulsion system would be most advantageous. Delivered to LEO by the HLLV system, a $COTV_G$'s primary mission would be to transport supplies from LEO to GEO and to return to LEO materials no longer needed at the GEO location (Ref 33:VI-12).

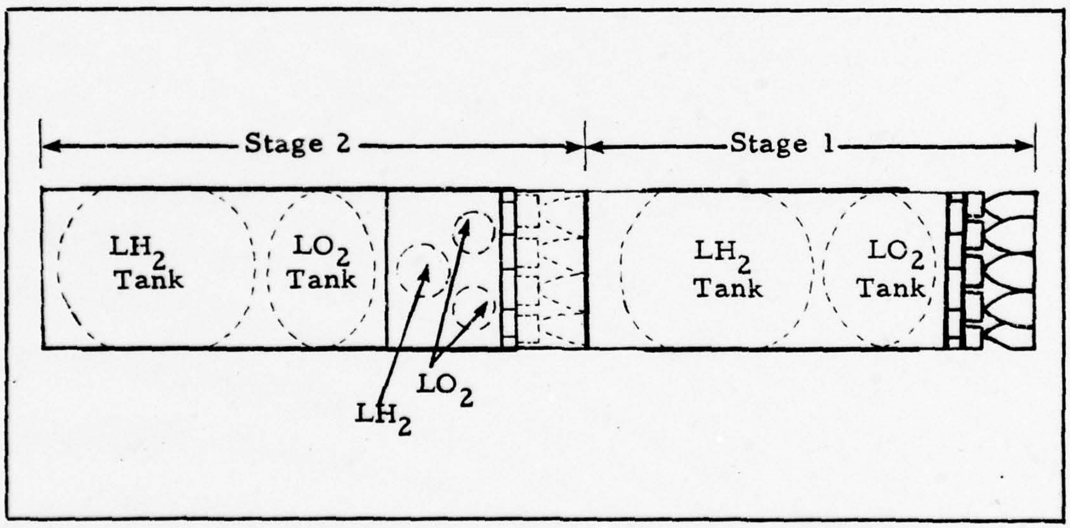

Fig. 22. Cargo Orbital Transfer Vehicle ($COTV_G$)
(Based on Ref 33:VI-14)

The final required space transportation vehicle would be the POTV, shown in Fig. 23. It would be used to transport all personnel on round trip missions between LEO and GEO. One round trip mission

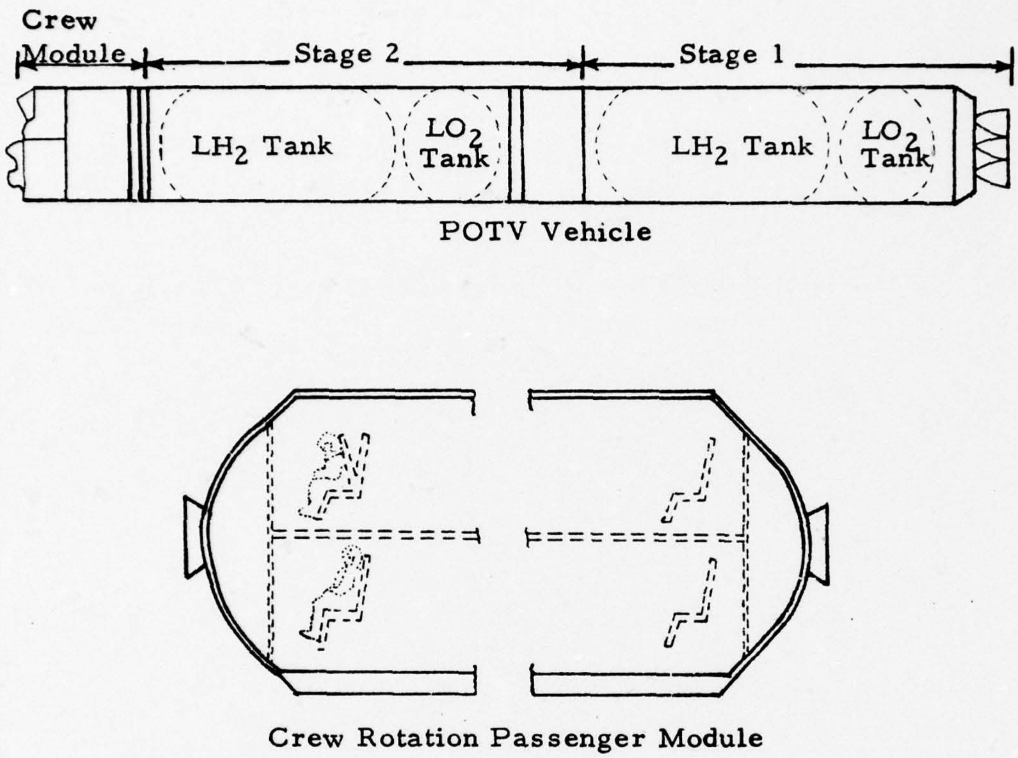

Fig. 23. Personnel Orbital Transfer Vehicle (POTV)
(Based on Ref 33:VI-17)

would take less than a day between LEO and GEO. In addition to personnel, or in place of personnel, the POTV would also be capable of carrying high-priority cargo. Like the $COTV_L$ and $COTV_G$, the POTV would be delivered to LEO by the HLLV system (Ref 33:VI-12, 14, 15, 16).

It should be noted that data relative to the transportation system described above is very preliminary. However, with the data available, it is possible to get an idea of the magnitude of the transportation system that would be required for both the SSP and the OSR systems. For example, the size of the payloads of the various systems is flexible. The actual payload mass could be adjusted to a higher or lower amount as necessary as long as the propellant mass, inert mass, and cost were equally adjusted. Additionally, the lifetime of a given vehicle would have to be reconsidered. In essence, what is important is the relationship between these various specifications because it is this relationship that determines the magnitude of the space transportation system.

In this appendix, data for both the SSP and OSR systems will be presented in tabular form. In addition, for each transportation vehicle, the requirements that determine the amount of material that must be transported will be listed and where necessary, an explanation will accompany the requirement to clarify its origin. Following these lists will be an explanation of how the vehicle, mass, and cost figures were calculated.

The SSP Transportation System

The SSP transportation system is designed for LEO construction. Construction would require 120 people working for 330 days in LEO. Transfer to GEO would be accomplished with a $COTV_L$ and would take

60 days. At GEO, a crew of 12 would complete the SSP assembly. This crew would be transported to GEO by a POTV and would work out of a space station and logistics depot connected directly to the SSP. Once in operation, the rated electrical capacity of the SSP system would be 10 GWe (Ref 32:9-5).

The main component of the space transportation system would be the HLLV. It would place its payload in 90 km by 500 km orbits at an inclination of 28.5°. NASA anticipates that lift-off would be from the NASA John F. Kennedy Space Center (Ref 33:VI-2).

Vehicular Requirements and Costs. Table XVIII contains a list of the number of flights that would be required of each vehicle in constructing, operating, and maintaining one 10 GWe SSP system. It also contains a list of the number of copies of each vehicle that would be required, based on the anticipated lifetime of each vehicle. Finally, it contains a list of the cost, in 1977 dollars, of each vehicular system based on the cost per flight from Table XVII. The determination of the required number of flights was made in accordance with the following requirements:

POTV Requirements:

1. GEO operations crews would require 12 people/year, working for 30 years in 1 year shifts (Ref 32:7-66).

2. Personnel provisions required for GEO operations would be 2.5 metric ton / person-year (Ref 33:V-14).

Table XVIII
SSP Space Transportation Vehicle Requirements

Vehicle	Number of Flights	Number of Vehicles	Cost[1] ($\$10^6$)
HLLV	1190	4[2]	11,424.
PLV	33[3]	3	353.1
$COTV_L$	94	10	3,017.4
$COTV_G$	102	4	1,091.4
POTV	30[4]	3	384.

[1]Based on cost per flight from Table XVII.

[2]This number could be as high as 6 since most of the trips would occur early, although a few would stretch over 30 years.

[3]Based on need for 120 men in first year and only 12 for the following 30 years. Assumes a slight design modification, to take 12 men and 30 metric tons of cargo per flight.

[4]Assumes one flight per year.

$COTV_G$ Requirements:

1. SSP maintenance would require 850 metric tons/year for 30 years (Ref 33:V-14).

$COTV_L$ Requirements:

1. The mass of the SSP to be transported to GEO would be 9.346×10^4 metric ton (Table VI). This mass includes an additional 11% of solar collector and solar propulsion system mass to account for solar cell losses which would occur during transit through the radiation belt.

2. The mass of the operations and maintenance facilities that would have to be transported to GEO would be 112

metric tons (Ref 32:9-11, 9-23).

PLV Requirements:

1. GEO operations crews would consist of 12 people/year working in 1 year shifts (Ref 32:7-66).

2. LEO construction crew would require 120 people for a 1 year period (Ref 32:9-5).

3. Personnel provisions would be 2.5 metric tons/person-year (Ref 33:V-14). Only the operations crew's provisions would fit in the PLV. The construction crew's provisions would be transported in an HLLV.

HLLV Requirements:

1. The total mass of the POTV, $COTV_G$, and $COTV_L$ vehicles, as listed in Tables XX and XXI would have to be placed in LEO.

2. The 9.346×10^4 metric ton (Table VI) mass of the SSP would have to be placed in LEO.

3. The 2.55×10^4 metric ton SSP maintenance material would have to be placed in LEO (Ref 33:V-14).

4. A mass of 1000 metric tons would have to be placed in LEO for maintenance material for the space construction and maintenance facilities (Ref 33:V-14).

5. The mass of the construction and maintenance facilities themselves would require that 2676 metric tons be placed in LEO (Ref 32:9-23).

6. Personnel provisions, at the rate of 2.5 metric tons/person-year (Ref 33:V-14) would have to be placed in LEO for the construction crews.

Vehicular Masses. Table XIX includes data relative to the material components of an HLLV. The materials are divided into two categories: those that would be required once in a vehicle's lifetime, and those which would have to be replaced after each flight.

Table XIX
HLLV Material Components[1]

Component	Mass Metric Ton
Per Vehicle Requirements	
Aluminum	498.1
Insulation	5.5
Copper	33.0
Steel	10.9
Inconel[2]	187.1
Electronics	2.0
TBD	1.6
Per Flight Requirements	
Water	115.0
Propellant	9,304.5

[1]Ref 34:Appendix IX, p 12-13A. Does not include material requirement for periodic maintenance and refurbishment of transportation system.

[2]A nickel alloy.

No data was available relative to the material components of the other space vehicles. However, since all but the $COTV_L$ would use propellant systems similar to the HLLV propellant system, it was decided that, as a first approximation, all but the $COTV_L$ could be considered miniature HLLVs.

Since data was available relative to the total inert mass and propellant mass of each vehicle, these parameters were used as a basis for the miniaturization. The total propellant and inert masses were not adjusted. However, each vehicle was assumed to contain inert mass components in the same proportion as the HLLV. To calculate the exact proportionality factor, the total inert mass of each vehicle was divided by the total inert mass of an HLLV, as given in Table XVII. For example, for the PLV the ratio of proportionality is

$$\frac{256}{853.2} = 0.3000 \tag{69}$$

This ratio was then used to conclude that a PLV would contain 30% of the aluminum contained in 1 HLLV, 30% of the insulation contained in 1 HLLV, and so forth for each HLLV component except propellant.

Table XX contains a list of the ratios for the PLV, $COTV_G$, and POTV systems. It also contains a list of the material requirements for these systems based on the number of flights required of each system from Table XVIII. The inert material requirement, except water, was determined by multiplying the vehicles ratio of proportionality by the number of vehicles required from Table XVIII. The water requirement was determined by multiplying the ratio by the number of flights required from Table XVIII because water was considered to be an expended resource. The propellant required was determined by multiplying the mass of propellant required per flight from Table XVII

Table XX
SSP PLV, $COTV_G$, and POTV Requirements

Item	PLV	$COTV_G$	POTV
Ratio: System Inert / HLLV Inert	0.3000	0.0410	0.0223
Inert Mass, except water (# of HLLVs)[1]	0.9	0.164	0.0669
Water (# of HLLVs)[1]	9.9	4.182	0.669
Propellant (Metric Tons)	62,436.	48,450.	4,770.
Total Mass[2] (Metric Tons)	64,239.	49,052.	4,896.

[1]Units are: the equivalent number of HLLVs.

[2]Based on 1 HLLV = 738.2 metric ton for non-water inert.
1 HLLV = 115.0 metric ton for water.

by the number of flights required from Table XVIII.

Table XXI contains data similar to that found in Table XX. However, since the $COTV_L$ propulsion system would not be similar to the HLLV's propulsion system, the inert mass could not be considered as a miniature HLLV. The inert mass requirement was determined by multiplying the inert mass per vehicle from Table XVII by the number of $COTV_L$s required from Table XVIII. The propellant requirement was determined by multiplying the propellant per flight from Table XVII by the number of flights required from Table XVIII.

Table XXI
SSP $COTV_L$ Requirements

Component	Mass (Metric Tons)
Inert (TBD)[1]	10,900.
Argon Fuel	75,200.
Total	86,100.

[1]Based on 10 units of 166 metric ton and 84 units of 110 metric tons to account for expendable inert mass. (Data from Tables XVII and XVIII.)

Finally, in Table XXII, all of the data relative to the mass of the SSP space transportation system is assembled.

The OSR Transportation System

For this analysis, it was assumed that the OSR transportation system would use the same vehicles used in the SSP system. The reason for this assumption was that NASA is moving in the direction of development of the vehicles described in this appendix. Therefore, it is unlikely that an entirely different set of vehicles would be developed for the OSR system. However, as mentioned earlier, it is possible to scale the vehicles to different sizes and this was done as necessary in this appendix.

The OSR system differs from the SSP system in several ways. First of all, essentially only two construction functions would be

Table XXII
SSP Space Transportation Mass Requirements

Item	HLLV Factor	Mass (10^3 Metric Ton)
Aluminum	5.131[1]	2.556
Insulation	5.131[1]	0.028
Copper	5.131[1]	0.169
Steel	5.131[1]	0.056
Inconel	5.131[1]	0.960
Electronics	5.131[1]	0.010
TBD		
Non-$COTV_L$	5.131[1]	0.008
$COTV_L$	--	10.9
Water	1204.751[2]	138.546
Propellant		
LO_2[3]	--	9753.65[4]
LH_2[3]	--	1434.36[4]
Argon	--	75.2
Total Mass	--	11,416.4

[1]Based on 4 HLLV vehicles plus HLLV units for inert mass of PLV, $COTV_G$, and POTV, Table XX.

[2]Based on 1190 HLLV flights plus HLLV units for water of PLV, $COTV_G$, and POTV, Table XX.

[3]LO_2/LH_2 ratio based on ratio of approximately (6.8):(1) used in Ref 5:145-147 energy analysis.

[4]Based on 1190 HLLV flights and propellant required from Table XX.

performed in space: reflector coating (Ref 14:56) and reflector assembly (Ref 13:45). Secondly, the system would use a Sun-synchronous orbit. This highly retrograde orbit would increase transportation costs by at least 100% by reducing the allowable payload (Ref 14:56). To compensate for this fact in this analysis, the HLLV and PLV payloads listed in Table XVII were assumed to be cut in half. Thirdly, the OSR would require a large quantity of kapton reflector material. This material would be bulky when packaged. Thus, an HLLV loading factor was devised.

The component parts of an OSR system reflector would be smaller reflectors known as Lunettas. One Lunetta could be carried by the current space shuttle and would have a mass of 19.77 metric tons (Ref 13:27). Since the maximum payload of the current space shuttle is 29.485 metric tons (Ref 26:49), the loading factor for the OSR reflectors would be

$$\frac{19.77}{29.485} = 0.67 \tag{70}$$

Thus, it was assumed that the HLLV could, by weight, be loaded at 67% of its anticipated payload capacity with OSR system reflectors. This loading factor reduced the HLLV payload to 74 metric tons. It should be noted that in this analysis, the loading factor was applied only to the initial OSR space system mass, and not to support equipment or replacement parts.

A fourth fundamental difference between the OSR and SSP systems is that the OSR system would not use a geosynchronous orbit. In the transportation scenario, the HLLV and PLV systems would transport people and supplies to a Sun-synchronous service orbit at 550 km (Ref 14:35). After assembly, each reflector would then be transferred to a sodium coating facility located in a 1100 km orbit (Ref 14:56). After coating is complete, each reflector would be transferred again to a Sun-synchronous 4184 km orbit from which it would operate (Ref 13:43). It is expected to take 15 years to get the entire system into operation (Ref 13:55).

At the 10 and 20 year points of each reflector's lifetime, the reflector would be transferred back to the 1100 km orbit for recoating (Ref 14:56). $COTV_G$ vehicles would be used for transfer from the 1100 km orbit to the 4184 km orbit in each case and would be supplemented with solar pressure for orbital plane changes (Ref 13:28).

Vehicular Adjustments. Since the intra-space transportation vehicles would not have to travel as far as geosynchronous orbit, it follows that they would need less propellant to transport a given payload. To determine how much the propellant requirement could be reduced, the following analysis was completed.

The potential energy of a satellite system is given by

$$U(r) = \frac{-G_e m_e m_s}{r} \tag{71}$$

where G_e is the universal gravitational constant, m_e is the mass of the Earth, m_s is the mass of the satellite, and r is the distance from the center of the Earth to the satellite (Ref 38:406-407).

The kinetic energy of a satellite system is given by

$$K(r) = \frac{G_e m_e m_s}{2r} \tag{72}$$

(Ref 38:412). The difference in energy between two different orbits, r_1 and r_2, would be the difference in the kinetic and potential energies of the two orbits. This difference would be

$$\mathcal{E} = \frac{-G_e m_e m_s}{r_2} + \frac{G_e m_e m_s}{2r_2} - \left(\frac{-G_e m_e m_s}{r_1} + \frac{G_e m_e m_s}{2r_1}\right) \tag{73}$$

$$= G_e m_e m_s \left(\frac{1}{2r_1} - \frac{1}{2r_2}\right) \tag{74}$$

where r_2 is greater than r_1.

Equation (74) was applied to the OSR system by calculating the energy difference between 550 km and 1100 km orbits, between 1100 km and 4184 km orbits, and between 550 km and 3.59×10^4 km geosynchronous orbits. The radius of the Earth was assumed to be 6376 km and the mass of the Earth was assumed to be 5.98×10^{24} kg. The results of the calculations were:

1. The 550 km to 1100 km energy difference was determined to be $\frac{2.12\ km^2/sec}{m_s}$.

2. The 1100 km to 4184 km energy difference was determined to be $\frac{7.79\ km^2/sec}{m_s}$.

3. The 550 km to 3.59×10^4 km energy difference was determined to be $\frac{24.08 \text{ km}^2/\text{sec}}{m_s}$.

Assuming that the amount of propellant required by a given spacecraft is directly proportional to the change in energy that it must undergo, then it is possible to determine how much of the propulsion system would no longer be required for OSR applications. For example, a transfer from 550 km to 1100 km, based upon the above calculations, would require only

$$\frac{2.12}{24.08} = 0.088 \tag{75}$$

or only 8.8% of the propellant needed by a vehicle carrying the same payload to geosynchronous orbit. Similarly, the 1100 km to 4184 km orbital change would require only 32.35% of the propellant needed for geosynchronous transfer.

In this analysis, the POTV was reduced in size and cost by multiplying by 8.8%. The payload was assumed to be held constant and the inert mass propellant, and operational cost of the POTV was scaled down by multiplying the Table XVII values by 8.8%.

The $COTV_G$ vehicles, however, were first increased in size to accommodate one OSR reflector. A reflector would have a mass density of 90.72 metric tons/km^2 and have an area of 8.73 km^2 (Ref 13:26,43). Thus, one reflector would have a mass of 792 metric tons. The $COTV_G$ listed in Table XVII would have a mass of 250 metric tons. Thus, the OSR $COTV_G$ would have to be 3.168 times larger. The mass and operation cost figures would also be 3.168 times larger.

After adjusting the figures for the $COTV_G$ as described above, these figures were then multiplied by 8.8% and 32.35% to account for orbital transfer to 1100 km and 4164 km, respectively. However, the payload was held constant at 792 metric tons. Table XXIII contains appropriately adjusted data for the entire OSR space transportation system.

Vehicular Requirements and Cost. Table XXIV contains data relative to the number of flights, number of vehicles, and total cost that would be required to construct, operate, and maintain one 74.2 GWe OSR system. Data for the manpower requirements was based on the ratio of the SSP orbital mass to the OSR reflector mass. In particular, the orbital mass of the SSP system would be 9.346×10^4 metric tons. The orbital mass of the OSR system would be 1.0454×10^6 metric tons based on 1320 reflectors at 8.73 km^2 each and a mass of 90.72 metric tons/km^2 (Ref 13:26,43). Thus, the scaling factor would be

$$\frac{1.0454 \times 10^6}{9.346 \times 10^4} = 11.18 \qquad (76)$$

Data for OSR manpower requirements was determined by multiplying the corresponding SSP figures by 11.18. This was done based

Table XXIII
OSR Adjusted Space Transportation System Vehicles[1]

Characteristic	HLLV	PLV	$COTV_G$[2]	$COTV_G$[3]	POTV
Payload (Metric Tons)	110.5[4]	18	792	792	20
Payload (Passengers)	-	25	-	-	75
Propellant (Type of Fuel)	LO_2/ LH_2	LO_2/ LH_2	LO_2/ LH_2	LO_2/ LH_2	LO_2/ LH_2
Propellant (Metric Tons/ Round Trip)	9304.5	1892	132.4	486.8	14.0
Inert Mass (Metric Tons)	853.2	256	9.8	35.9	1.7
Cost ($\$10^6$/ Flight)	9.6	10.7	3.0	11.0	1.1
Lifetime (Flights)	300	100	30	30	30

[1]Based on material from Table XVII, adjusted as described in the text.

[2]For orbital transfer from 550 km to 1100 km.

[3]For orbital transfer from 1100 km to 4148 km.

[4]For OSR material use 74 metric ton to account for packing factor.

Table XXIV
OSR Space Transportation Vehicle Requirements[1]

Vehicle	Number of Flights	Number of Vehicles	Cost[2] ($\$10^6$)
HLLV	39,889	133	382,934.4
PLV	215	3	2,300.5
$COTV_G$[3]	4,198	140	12,594.0
$COTV_G$[4]	3,960	132	43,560.0
POTV	60	3	66.0

[1]Based on calculations similar to those done to compile Table XVIII.

[2]Based on per flight cost of Table XXIII.

[3]For orbital transfer from 550 km to 1100 km.

[4]For orbital transfer from 1100 km to 4148 km.

on the assumption that manpower required for space operations is directly proportional to the amount of material being assembled and maintained.

The specific requirements used to formulate Table XXIV were:

POTV Requirements:

1. Orbital operations and maintenance would require 134 people/year, working in 1 year shifts for 30 years, at 1100 km. The 11.18 factor was used here.

2. Personnel provisions required would be 2.5 metric tons/person-year (Ref 33:V-14), working at 1100 km. However, only 20 metric tons/flight would be carried by the POTV. The rest would go in the $COTV_G$.

$COTV_G$ Requirements:

1. OSR maintenance would require 30% of the original OSR reflector mass, or 3.13×10^5 metric tons over a 30 year period (Ref 14:55). This would be transported to 1100 km.

2. The OSR mass of 1.0454×10^6 metric tons would be transported once to 1100 km.

3. The OSR mass of 1.0454×10^6 metric tons would be transported three times between 1100 km and 4184 km, for sodium coating purposes.

4. Personnel provisions for the operations and maintenance crews at 1100 km which would not fit in the POTV would be carried. This would amount to 8850 metric tons based on POTV requirements, above.

5. The mass of the operations and maintenance facilities to be transported to 1100 km would be approximately 4282 metric tons. This value was determined by assuming that each person would require an equal amount of equipment. Since the OSR construction would take 15 years (Ref 13:55), only 89 people/year would be required for construction. Thus, while the SSP would need 120 people for construction and 12 for maintenance in a given year (a total of 132), the OSR would need only 89 for construction and 134 for maintenance (a total of 223). Therefore, the ratio of OSR to SSP construction and maintenance facility needs should be about 1.6:1.

6. The mass of material required to maintain the operational facilities in 1100 km orbit would be 1600 metric ton. This, too, was based on the ratio of 1.6:1 as explained above.

7. The mass of $COTV_G$ system required for the 1100 to 4184 km orbital transfer would have to be transported from 550 km to 1100 km.

PLV Requirements:

1. Operations and maintenance crews at 1100 km orbit would require 134 people/year, working in 1 year shifts for 30 years.

2. Construction at 550 km would require 89 people/year for 15 years.

HLLV Requirements:

1. The total mass of the POTV and $COTV_G$ vehicles required, must be placed in 550 km orbit.

2. The 1.0454×10^6 metric ton mass of the OSR must be placed in 550 km orbit, at 74 metric ton/HLLV flight.

3. The 3.13×10^5 metric ton OSR maintenance material must be placed in 550 km orbit.

4. The mass of 1600 metric tons for maintenance material for the operations and maintenance facilities must be placed in 550 km orbit.

5. The mass of the operations and maintenance facilities of 4282 metric tons must be transported to 550 km.

6. All personnel provisions, at 2.5 metric tons/person-year must be transported to 550 km.

Vehicular Masses. Tables XXV and XXVI contain information relative to the mass of the OSR transportation system. This information was calculated in a manner similar to the calculations for the SSP system. Therefore, the explanation of these calculations will not be repeated.

Table XXV
OSR PLV, $COTV_G$, and POTV Requirements[1]

Item	PLV	$COTV_G$[2]	$COTV_G$[3]	POTV
Ratio: System Inert / HLLV Inert	0.3000	0.0115	0.0421	0.0020
Inert Mass except water (# of HLLVs)[4]	0.9	1.61	5.557	0.006
Water (# of HLLVs)[4]	64.5	48.277	166.716	0.12
Propellant (10^3 Metric Tons)	406.8	555.8	1927.7	0.8
Total Mass (10^3 Metric Tons)[5]	414.9	562.5	1951.0	0.8

[1]Based on calculations similar to those done to compile Table Table XVIII.

[2]For orbital transfer from 550 km to 1100 km.

[3]For orbital transfer from 1100 km to 4148 km.

[4]Units are: the equivalent number of HLLVs.

[5]Based on 1 HLLV = 738.2 metric ton for non-water inert.
1 HLLV = 115.0 metric ton for water.

Table XXVI
OSR Space Transportation Mass Requirements[1]

Item	HLLV Factor	Mass (10^3 Metric Ton)
Aluminum	141.073	70.268
Insulation	141.073	0.776
Copper	141.073	4.655
Steel	141.073	1.538
Inconel	141.073	26.395
Electronics	141.073	0.282
TBD	141.073	0.226
Water	40,168.6	4619.389
Propellant		
LO_2	-	326,084.67
LH_2	-	47,953.63
Total Mass	-	378,761.83

[1]Based on calculations similar to those done to compile Table XXII.

Vita

Rolf Enger was born [redacted] the son of [redacted] He graduated from [redacted] in 1968. Upon completion of the necessary course work in 1972, he received the degree of Bachelor of Arts, Summa Cum Laude, in Physics, from St. Olaf College. Also upon graduation, he received a commission in the USAF through the ROTC program. Prior to entering the Air Force Institute of Technology in June 1976, he was assigned to the 44th Strategic Missile Wing at Ellsworth Air Force Base, South Dakota. While at Ellsworth, he served as a missile combat crew commander, as a deputy missile combat crew commander, and as a missile combat crew instructor. As a missile officer, he was in direct command and control of up to 50 Minuteman Intercontinental Ballistic Missiles. In [redacted] and on [redacted] at the close of this research effort, the couple was blessed by the birth of their first child, [redacted]

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The potential consumption of natural resources by four solar electric generating systems was evaluated. They included a terrestrial solar thermal, a terrestrial photovoltaic, an orbiting solar reflector, and a satellite solar power system. Each was evaluated on its projected consumption of materials, land, water, manpower, energy, and money. The evaluation demonstrated that, per megawatt of electrical generating capacity, the terrestrial systems — over

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SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered) would consume less resources mainly because they would not require massive space transportation and construction systems and expensive developmental programs. It was also shown that construction of terrestrial systems would require fewer technological advancements and would pose less of a threat to the environment. A feasibility study of orbiting solar reflectors demonstrated that single-mirror systems may be useful for intra-space power generation. The report contains a 47-item bibliography.