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FINAL REPORT

LASER ROCKET SYSTEM ANALYSIS

PREPARED UNDER CONTRACT NO. NAS3-20372
BY
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Section 1
SUMMARY

1.1 OBJECTIVES

The objective of the Laser Rocket Systems Analysis is to identify promising missions and potential laser rocket systems, establish critical technology areas, and compare the potential laser rocket system with conventional space propulsion systems of equal capability relative to technology requirements and cost effectiveness. The study includes three laser rocket systems whose laser transmitters are space-based, ground-based, and airborne, respectively.

1.2 STUDY SCOPE

The Laser Rocket Systems Analysis Study investigated parametrically the feasibility and utility of laser propulsion units with the laser devices remotely located to provide velocity increments to orbiting payloads for drag makeup, orbit and plane changes, and other orbital maneuvers required of space missions including interplanetary missions. Parametric limits established by the Statement of Work included: Laser power, 5 to 1000 MW; wavelengths, 0.5 to 10.6 μm; aperture diameters, 3 to 30 m (9.84 to 98.43 ft); engine thrust, 445 to 133,440 N (100 to 30,000 lbf); and specific impulse, 9800 to 19,600 N-s/kg (1000 to 2000 lbf-s/lbm). The velocity increments for missions ranged from 90 to 20,100 m/s (300 to 66,000 ft/s).

Also investigated was the use of solar electric transfer vehicles for comparison with the Laser Rocket Systems and conventional chemical systems.

1.3 STUDY RESULTS

The results of the study showed that, with advanced technology, laser rocket systems with either a space- or ground-based laser transmitter could reduce the national budget allocated to space transportation by 10 to 345 billion dollars over a 10-year life cycle when compared to advanced chemical propulsion systems (LO_2-LH_2) of equal capability. The variation in savings depends upon the projected mission model. The mission models used for comparison ranged from 460 shuttle-type payloads to the 460 shuttle-type payloads plus 13 space power satellites. The 10-year life cycle cost ratios of LO_2-LH_2 systems to laser rocket systems ranges from 2.4 to 6.9. To achieve these savings, technology developments are required in the areas of the laser device; large, lightweight adaptive optics; laser propulsion engines; and pointing and tracking. Laser rocket systems with ground-based transmitters require higher level technology developments than space-based transmitters because of the atmospheric effect on beam propagation. Laser rocket systems with airborne transmitters proved to be noncompetitive. The use of gallium-arsenide solar arrays reduces the laser transmitter weight and the cost of deployment. However, the overall effect to the 10-year life cycle costs is small. Solar electric propulsion may be competitive with laser rocket systems, but the long trip times and production requirements may not be compatible with mission models developed in this study.

1.4 CONCLUSIONS

Laser rocket systems offer the potential to reduce cost of orbit-to-orbit transfer of payloads requiring high velocity increments (e.g., geosynchronous). The high specific impulse coupled with relative short trip times provides for the minimum number of vehicles to perform a given mission model, plus the high specific impulse minimizes the resupply of propellant to low earth orbit — the primary life cycle cost driver.

The potential savings justify a continuing effort to develop critical technologies through theoretical studies and experiments.

Section 2

INTRODUCTION

2.1 BACKGROUND

The use of laser energy for propulsion of space vehicles has been of interest since laser devices with significant power outputs appeared feasible. Among the earlier studies, Kantrowitz (Ref. 1) examined the use of ground-based lasers to provide the energy for launching vehicles to orbit. Minovitch (Ref. 2) and Pirri (Ref. 3) also were early investigators of laser propulsion. More recent investigations of various aspects of laser propulsion (Refs. 4, 5, 6, 7) have been, and are being, conducted for NASA Lewis Research Center, Defense Advanced Research Projects Agency, and Air Force Rocket Propulsion Laboratory. These and other studies and experimental programs have made significant advances in critical technologies such as lasers, large optics, and pointing and tracking. With the recent advancements, it is becoming more and more evident that laser rocket systems have more potential for significantly improving space propulsion capabilities than any other systems currently under study. Nuclear rocket systems offer high specific impulse but are penalized by the device and shielding weights. Electric propulsion systems also have the potential of high specific impulse but are restricted to very low thrust. Cryogenic chemical rocket systems such as LF_2-LH_2 show improvements over current systems because of the improved propellant bulk density but have limited specific impulse.

The laser rocket systems investigated in this study were for orbital transportation using space-based, ground-based and airborne laser transmitters as depicted in Figure 1. The propulsion unit (Figure 2) of these systems utilizes a continuous wave (CW) laser beam focused into a thrust chamber which initiates a plasma in the hydrogen propellant (possibly seeded to improve absorptivity), thus heating the propellant and providing thrust through a suitably designed nozzle and expansion skirt. The specific impulse is limited only by the ability to adequately cool the thruster and the amount of laser energy entering the engine. The energy generation hardware is remote, and no chemical combustion is present which could result in "hard starts," therefore development and testing of multiple-start engines for reusable vehicles should be much easier and less expensive. With the continued advancement in laser output powers and shorter wavelengths; large lightweight, adaptive optics; reflective coatings; laser engines; and pointing and tracking technologies, the laser rocket system could overcome the constraints and penalties of other types of space propulsion systems and provide a significant step forward in the utilization and exploration of space. While the orbit-to-orbit laser rocket system will greatly reduce the transportation costs of today's sophisticated and expensive satellites, the real impact will be that relatively inexpensive bulk payloads can be transported at affordable costs to build large space stations, space manufacturing facilities, and space power systems.

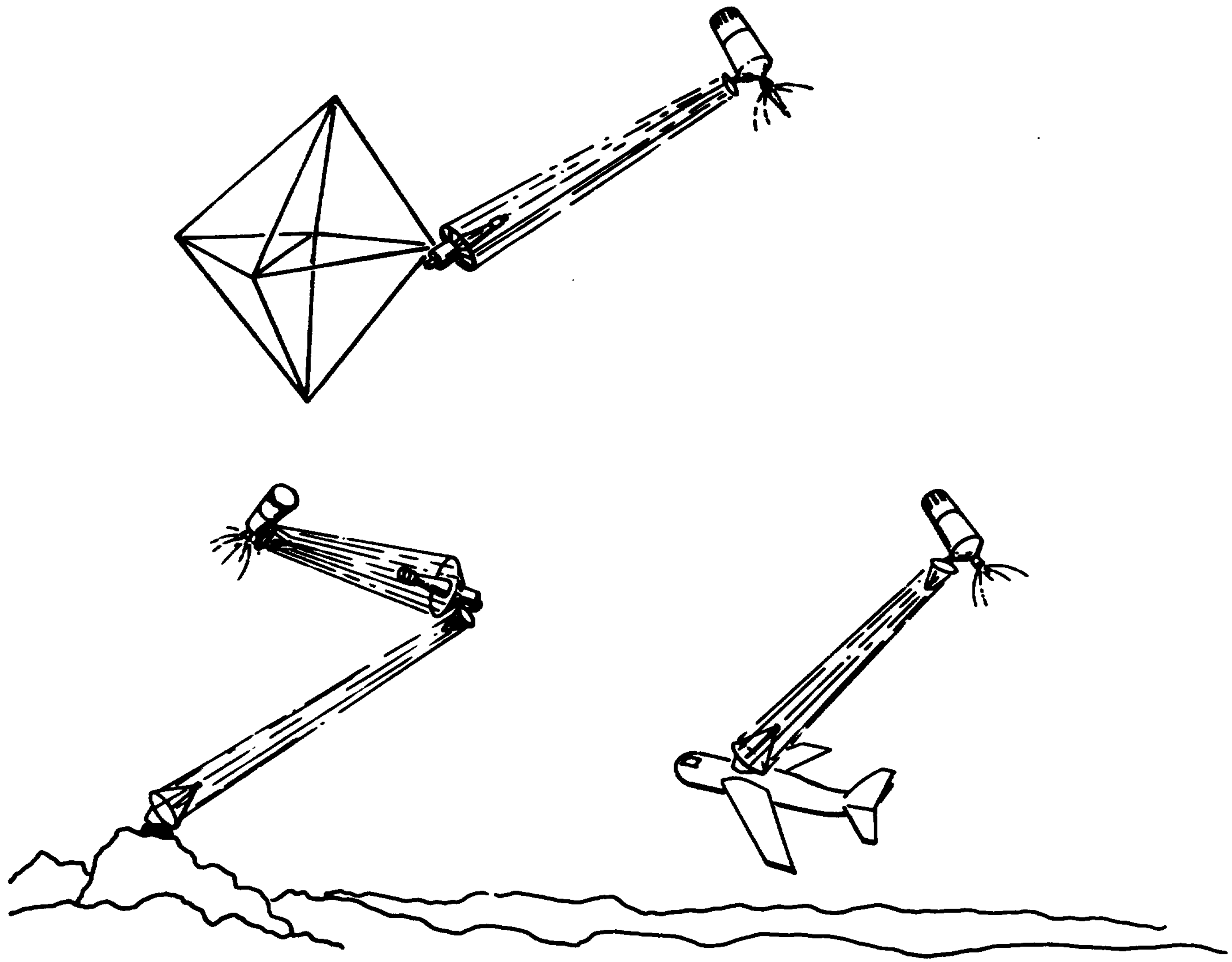

Figure 1. Laser basing concepts

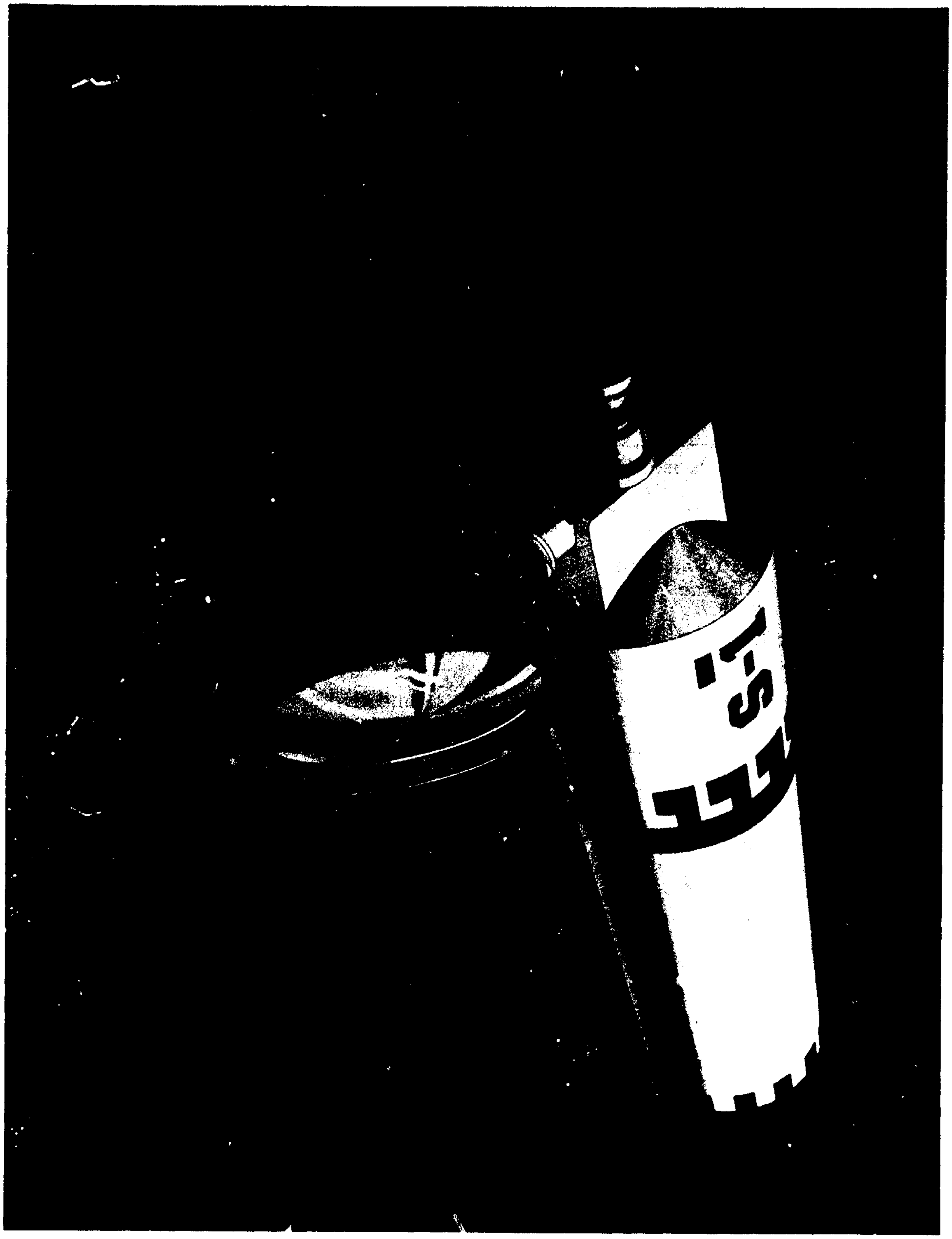

Figure 2. Propulsion unit concept

2.2 STUDY DESCRIPTION

The objective of the Laser Rocket System Analysis Study is to identify promising missions and potential laser rocket systems, establish critical technology areas, and compare the laser rocket systems with conventional space propulsion systems relative to performance, technology requirements, and cost effectiveness. The study includes laser transmitters that are space-based, ground-based, and airborne.

The Laser Rocket System Analysis is generally bounded by the Statement of Work as illustrated in Table I. The technology bounds varied from the minimum technology required to perform the mission to those maximum levels beyond which the performance improvements are negligible.

TABLE I: STUDY BOUNDS FROM STATEMENT OF WORK

Parameter	Minimum	Maximum
Laser Power, CW or Pseudo CW (MW)	5	1000
Wavelength (μm)	0.5	10.6
Lasers per Transmitters	1	> 1
Number of Transmitters	1	> 1
Transmitter Aperture Diameter (m/ft)	3/9.8	30/98.4
Receiver Aperture Diameter (m/ft)	3/9.8	30/98.4
Engine Thrust (N/lbf)	445/100	133,440/30,000
Specific Impulse (N-s/kg/lbf-s/lbm)	9800/1000	19,600/2000

Considering these limits, a mission model was established in Task I ranging in mission velocity requirements of 90 m/s (300 ft/s) for drag made up to 20,100 m/s (66,000 ft/s) for an interplanetary mission. Payload weights ranged from 500 kg (1100 lbm) for a Saturn/Uranus probe, to 148,000 kg (326,300 lbm) for a single load of a space power satellite. The mission model was made flexible by the use of activity multipliers that could be used to increase activity or to zero out each mission. With these additional limits established, Task II, Parametric Analysis: Space-Based Laser, was performed by parametrically synthesizing laser propulsion units throughout the range of limits. Upon determining the most likely propulsion units that could perform the basic mission model, complete systems were synthesized varying laser transmitter basing modes and other parameters to minimize total weight on orbit which usually result in minimum design, development, test and engineering (DDT&E) and investment costs. From the system syntheses, it was determined that the primary driver of life cycle costs (LCC) would be the transportation of replenishment propellant to space. Further system syntheses were performed to balance minimum weight on orbit and propellant requirements. This parametric analysis resulted in two laser rocket systems with space-based laser transmitters being recommended to the NASA. The smaller system was conceived to accomplish a mission of transporting from low earth orbit a 2268 kg (5000 lbm) payload round trip to geosynchronous equatorial orbit (GEO). This system will have excess capability when performing less demanding missions. The second system was conceptually designed to transport from low earth orbit (LEO) a 148,000 kg (326,300 lbm) payload one way to GEO and return to LEO empty. The two systems required laser output powers of 16 and 490 MW, respectively.

Task III, Conceptual Design: Space-Based Laser, included the preparation of inboard profiles and weight statements to the subsystem level for each major unit in the systems (transmitter unit, propulsion unit, and energy relay unit).

Task IV, Concept Evaluation and Costs: Space-Based Laser, included the synthesis of conventional-type LO_2-LH_2 systems with performances equal to the two laser rocket systems. The conventional systems were costed for DDT&E and first unit. Using the flexible mission model developed in Task I, life cycle costs were established for both conventional and laser systems. The mission model variations ranged from all small payloads that could be accomplished by the smaller system to exclusively heavy payloads requiring the larger system only. Various mixes of small and large payloads and activity levels also were examined. In each case the laser rocket system was more cost effective. The development, investment, and 10-year life cycle costs were spread assuming a 1990 Initial Operating Capability (IOC). These costs were discounted to determine their present value. Again in each case, the laser rocket system was more cost effective by factors ranging from 2.4 to 6.9.

Tasks V, VI, and VII involved the parametric analysis, conceptual design, and concept evaluation tasks for laser rocket systems with ground-based laser transmitters. The analyses were performed similar to the space-based systems except the added atmospheric effects on the laser beam were modeled. Also, it was found that energy relay units would be required at about 6500-km (3500-nmi) altitude to maintain manageable orbit phasing with respect to the various missions and the laser transmitter location on earth. Because of the atmospheric losses, the laser output power requirements (to perform identical mission models as the space-based systems) increased from 16 to 37.5 and 490 to 1000 MW, respectively, for the small and large payloads. Evaluation of the ground-based systems showed that they also resulted in lower life cycle costs than the conventional systems. The space-based and ground-based laser rocket systems were compared to one another and found to be about equal in life-cycle costs. The ground-based systems would have higher technology requirements for laser and optics development because of the atmospheric effect on laser beam propagation.

Tasks VIII, IX, and X involved the parametric analysis, conceptual design, and concept evaluation tasks for airborne laser transmitters. However, during the ground-based studies, it became apparent that an airborne system would not be competitive because of the weight and volume constraints, if it could perform the missions at all. As a result, the analysis of airborne laser rocket systems was discontinued.

Tasks XII and XIII were added to evaluate the impact of advanced solar cells and to compare solar electric propulsion, respectively.

2.3 SYSTEMS DESCRIPTIONS

The laser rocket system concepts developed in this study are based on the basic mission model shown in Table II. Four systems were developed — two for space-based

TABLE II. LASER ROCKET SYSTEMS MISSION MODEL

Mission Model		ΔV (fps)	Payload Weight (lb)	Nom. Steady State Activity
Geocentric Missions				
Current Projected				
Hi. Ellip., Hi. Inc.		9,000	3,000	5/yr
Geosynchronous		14,100	5,000 RT	15/yr
	9 Reusable			
Advanced				
Geosynch. Space Station		14,100	55KD/25 KR	5/yr
Geosynch. SPS		14,100	326,000	400/yr
Extreme Lat. Coverage	80 Reusable	20,000/yr	5,000	6/yr
Orbit Maint. of Lg. Structures (LEO)		300	100,000/700,000	2/yr
Interplanetary Missions				
Current Projected				
Mercury Orbiter		17,000	9,200	4/10 yr
Pioneer Saturn/Uranus/Titan Probe		40,000	1,100	2/10 yr
Advanced	130 Expendable			
Neptune Jupiter Flyby		40,000	7,000	2/10 yr
Uranus Orbiter – 3.5 yr Trip Time		66,000	2,000	2/10 yr
Nuclear Waste Disposal		30,000	10 – 30 K	40/yr

RT = Round Trip
D = Deliver
R = Return

laser transmitters and two for ground-based transmitters. The two systems in each basing mode are: (1) for 2268-kg (5000-lbm) payloads round trip to GEO; and (2) for 148,000-kg (326,300-lbm) payloads one way to GEO and return empty. The four systems are briefly described as follows:

2.3.1 Laser Rocket Systems With Space-Based Transmitters

The space-based laser rocket systems consist of three basic units – the laser transmitter unit (LTU), the propulsion unit (PU), and an energy relay unit (ERU). The LTU is deployed in a 500-km (270-nmi) circular orbit with a 28.5° inclination (basic shuttle delivery orbit). The ERU is deployed at GEO. The mode of operation is for the PU to pick up its payload at its basic deployment orbit, then the LTU transfers energy to the PU to raise the apogee to synchronous altitude. This is accomplished by thrusting near perigee and may take more than one burn. Plane change and orbit circularization at GEO is accomplished by LTU transferring energy to the ERU which in turn refocuses, corrects wavefront errors and relays the energy to the PU. The return trip is a reverse procedure.

- <u>Laser Transmitter Unit, 2268-kg Payload</u>

The LTU (Figure 3) for a 2268-kg (5000-lbm) payload is assumed to have a closed-cycle, 16-MW laser device operating at 0.5-μm wavelength (EXCIMER type). The device requires a 131-MW supporting electrical power system which uses a 0.972-km^2 (0.375-mi^2) solar collector with a 2:1 solar concentration on silicon cells. The transmitting aperture of the LTU is a Cassegranian, segmented, 30-m (98.4-ft) diameter, adaptive optical system which focuses the laser beam on the receiving aperture (PU or ERU). The LTU, including the electrical power supply, which dominates the unit, weighs 668,000 kg (1,500,000 lbm). The electrical power supply and power conditioning takes up 609,000 kg (1,400,000 lbm) of the total weight.

- <u>Propulsion Unit, 2268-kg Payload</u>

The PU (Figure 4) for the 2268-kg payloads has a 1000-N (225-lbf) thrust engine requiring 13.4-MW laser power input to provide a specific impulse of 19,600 N-s/kg (2000 lbf-s/lbm). The receiver is an off-axis, monolithic, adaptive aperture for a maximum continuous operating time of 10,000 s. No wavefront error correction is required and beam jitter is not expected to be a problem because the energy transfer range to the thruster is only a few meters. The PU receiver optics rotate 360° about an axis normal to the vehicle center line which, combined with vehicle roll, provides a pointing capability of 4π sr. The propellant management system is designed for 3,131 kg (6,903 lbm) of LH_2 which provides a velocity increment of 10.5 km/s (34,450 ft/s) with a 2268-kg (5000-lbm) payload. This velocity increment includes the gravity and off-optimum trajectory velocity losses for the round trip to GEO. The propulsion unit total weight is 5291 kg (11,665 lbm).

- <u>Energy Relay Unit, 2268-kg Payload</u>

The ERU (Figure 5) for the 2268-kg payload has an off-axis, segmented, adaptive optics receiver which must be near-diffraction limited quality to avoid inducing

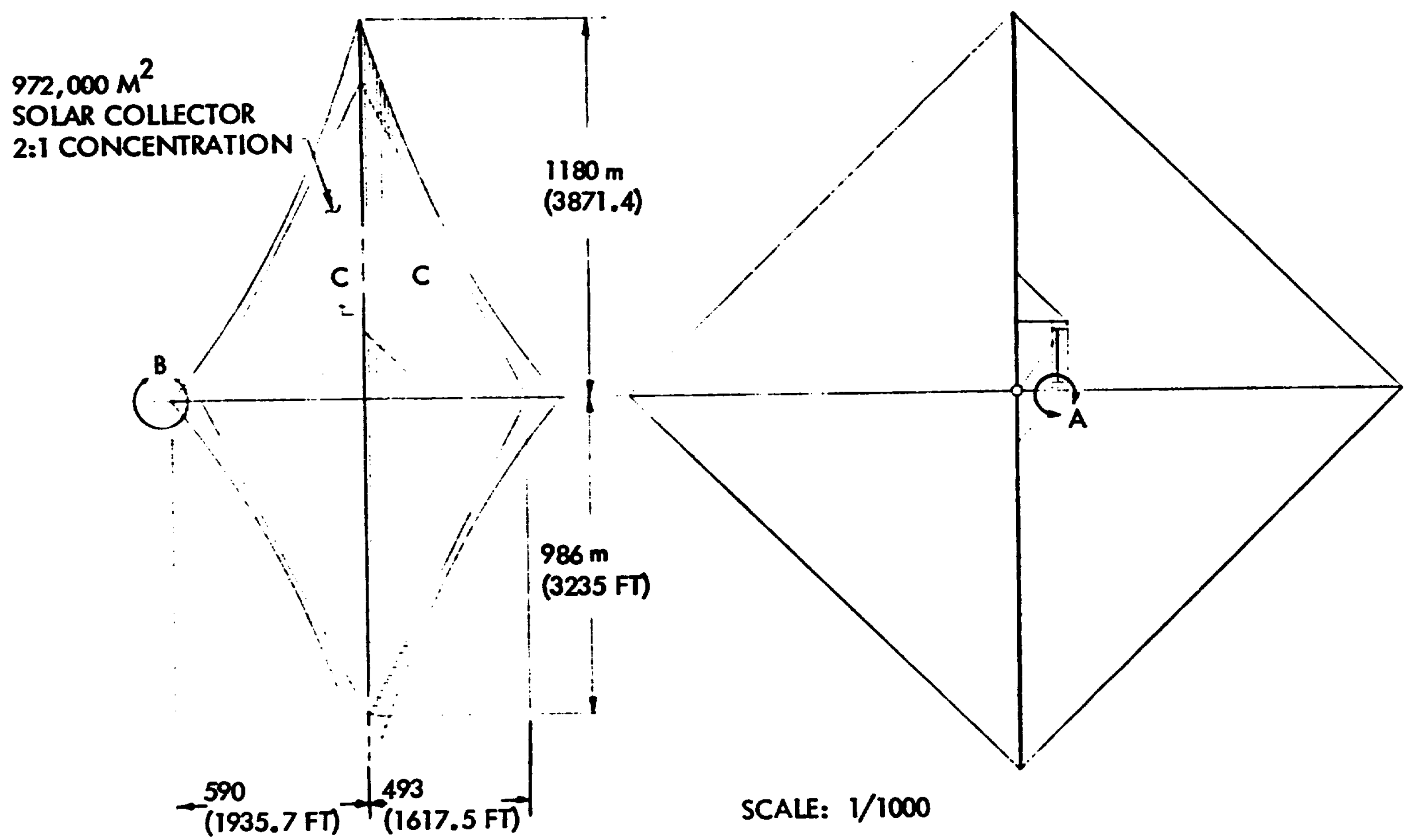

Figure 3. 16-MW laser transmitter unit

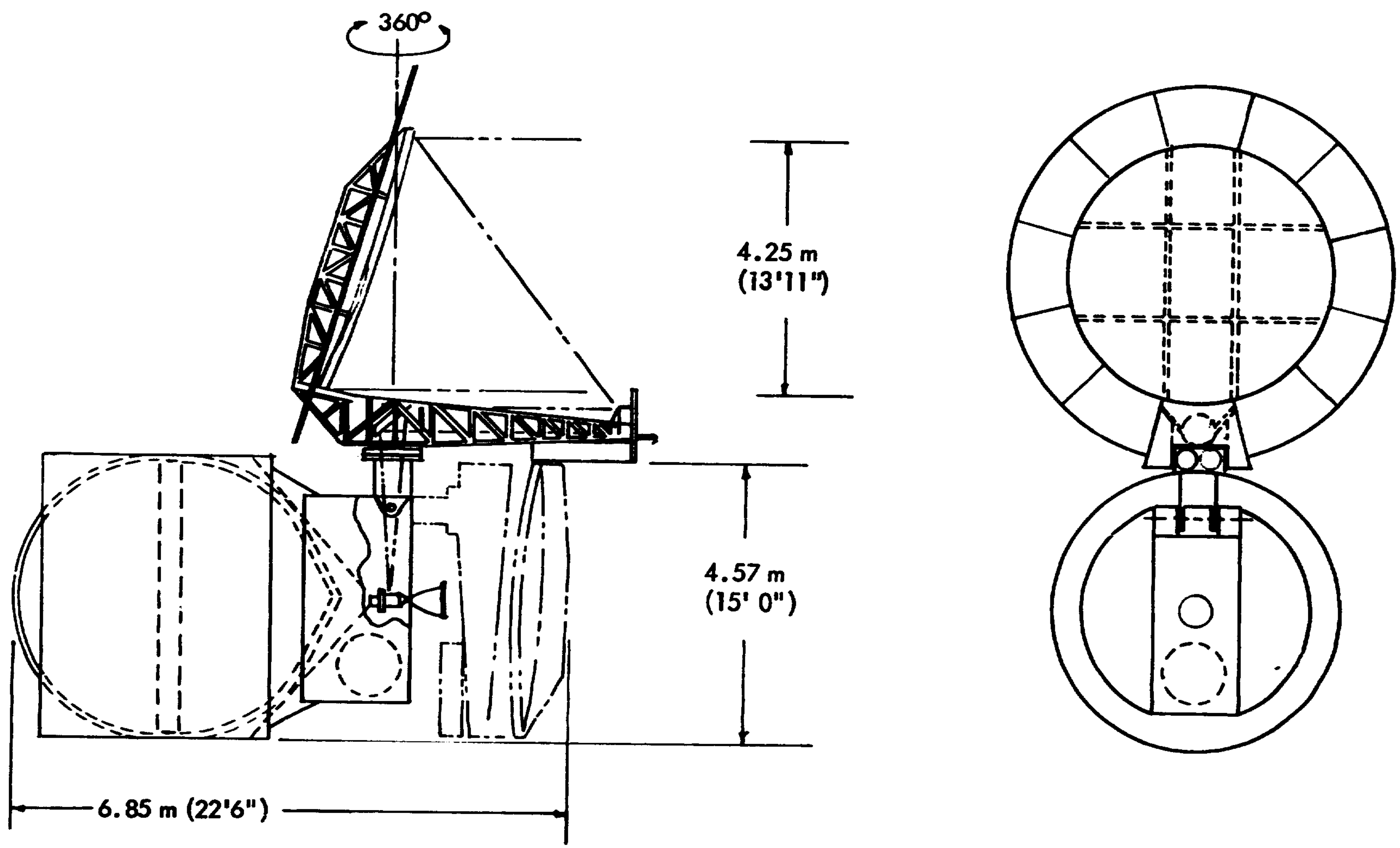

Figure 4. Propulsion unit for 2268-kg payloads

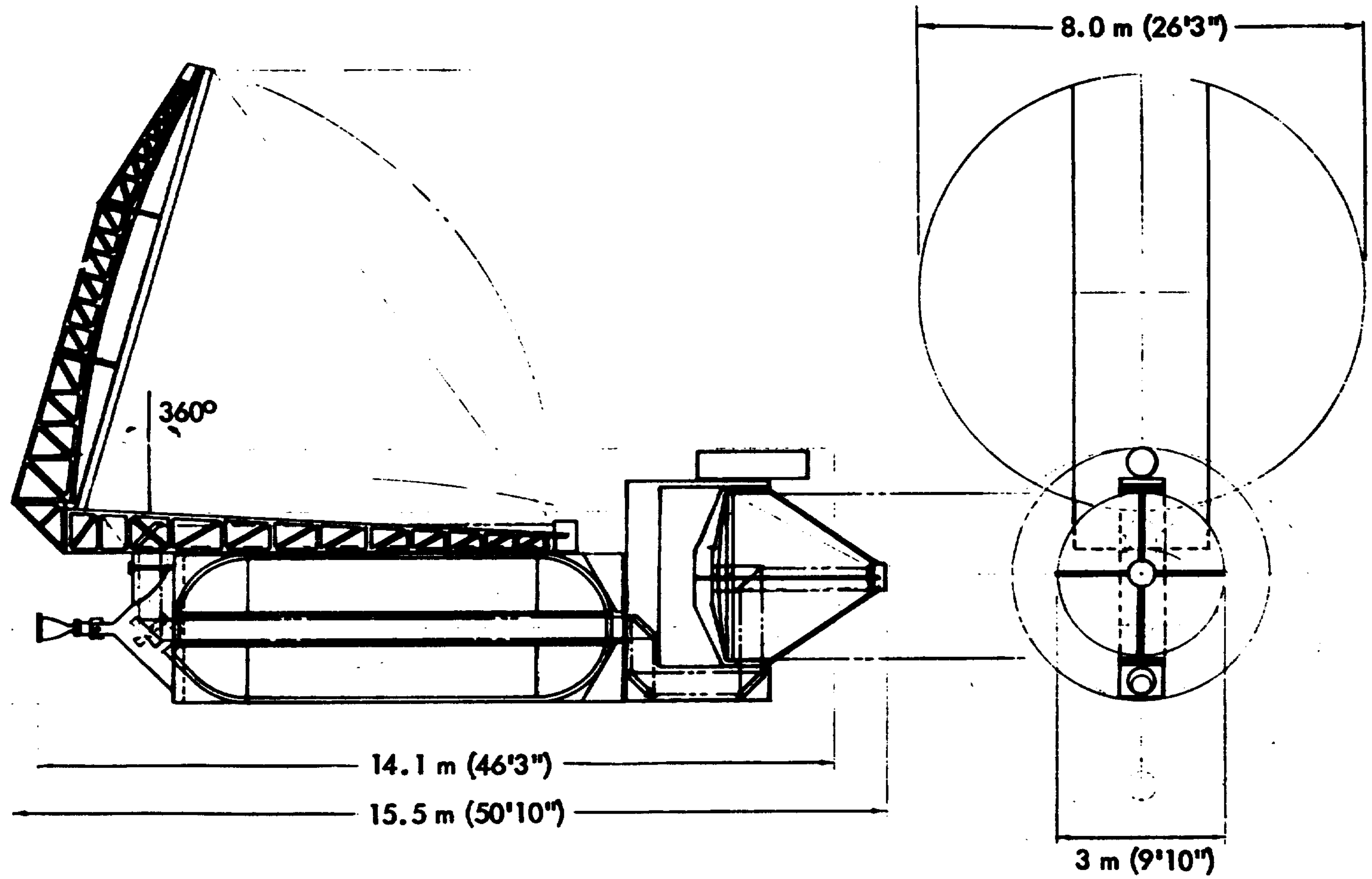

Figure 5. Energy relay unit for 16-MW system

additional wavefront errors. Like the PU receiver, the ERU receiver rotates 360° about an axis normal to the vehicle centerline providing 4 π-sr pointing. The beam is reduced and transferred via transfer mirrors to the ERU transmitter aperture which corrects wavefront errors, refocuses and transmits the energy beam to the PU. The transmitting aperture is Cassegrainian, monolithic and adaptive. The transmitting aperture has greater than 2 π-sr pointing capability and when coupled with vehicle attitude, and the receiver gives the ERU the capability to receive and transmit from and to any direction. The ERU also includes a laser propulsion system to carry itself to GEO. The thruster is the same as used on the PU. The energy is diverted from the normal optical path and focused into the thruster during the required propulsive burns. The ERU receiver size permits a transmit range from LEO to GEO for plane change and circularization. By replenishing the propellant, the relay can also be brought down to LEO for maintenance and servicing. The total weight of the energy relay unit is 8,500 kg (18,700 lbm) with 2,130 kg (4,690 lbm) being LH_2 propellant.

- Laser Transmitter Unit, 148,000-kg Payload

The LTU for the 148,000-kg (326,000-lbm) payload is a larger version of the 2268-kg payload transmitter unit. The system requires a 490-MW device with a 4-GW electrical power supply. The total transmitter unit weighs 12,300,000 kg (27,000,000 lbm).

- Laser Propulsion Unit, 148,000-kg Payload

The PU for 148,000-kg payloads have the same features as the 2268-kg payload system. The engine has a thrust 31,150 N (7,000 lbf) requiring 418-MW laser input. Because of the higher flux density in the diffraction pattern rings, the receiver is 4.5-m (14.76-ft) diameter. The total weight of the unit is 115,000 kg (253,000 lbm), including 71,000 kg (158,000 lbm) of LH_2 propellant.

- Energy Relay Unit, 148,000-kg Payload

The ERU for the 148,000-kg payload has the same features as the 2268-kg payload ERU including the diameters of both the receiver and transmitting apertures. The primary difference is the cooling system to handle the much larger absorbed energy level. The total weight of the relay unit is 69,000 kg (152,000 lbm) of which 16,800 kg (37,000 lbm) is LH_2 propellant to carry itself to GEO.

2.3.2 Laser Rocket Systems With Ground-Based Transmitters

The laser rocket systems with ground-based transmitters are also designed around the 2268-kg payload round trip to GEO and the 148,000-kg payload one way to GEO, returning empty. The ground-based system, however, has four basic units: the laser transmitter unit (LTU); the propulsion unit (PU), identical to space-based systems; the Medium Earth Orbit (MEO) energy relay units (MEO-ERU); and the GEO-ERU. The LTU is located in Hawaii at approximately 3,650-m (12,000-ft) elevation. The location is approximately 20° north latitude which will permit a series of MEO-ERUs (6) to be deployed in an equatorial plane to provide constant contact with the LTU. The mode of operation for the ground-based laser rocket system is to beam the energy through the atmosphere to the MEO-ERU that is in view of the LTU. This MEO-ERU refocuses, corrects the wavefront errors, and relays the energy to the propulsion unit either directly or via another MEO-ERU. The plane change and circularization is accomplished by relaying the energy to the GEO-ERU which refocuses, correct the wavefront errors, and relays the energy beam to the propulsion unit.

- Ground-Based Transmitter Units

The ground-based laser transmitters for the 2268- and 148,000-kg payloads have closed-cycle, EXCIMER type laser devices operating at 0.5-μm wavelength and 37.5- and 1000-MW CW laser outputs, respectively. The electrical power systems supporting the laser device are alternators driven by JP fueled turbines. The primary apertures are Cassegrainian, segmented 10-m (32.81-ft) adaptive optical systems. To control the beam spread through the atmosphere, approximately 10,000 actuators are required (8.7-cm apart). The difference between the optical systems for the 37.5- and 1000-MW laser devices is determined by the amount of cooling required to dissipate the absorbed energy and maintain the optical surfaces within their thermal design limits.

- Propulsion Units

The propulsion units for the laser rocket systems with ground-based transmitters are identical to those previously described for the space-based transmitter systems.

- MEO Relay Units

The MEO-ERUs are deployed in a 6580-km (3550-nmi) circular orbit with zero inclination (equatorial). Six relays are equally spaced so that the zenith angle of the LTU does not exceed 61°. The receiving aperture is an off-axis, segmented, 8-m (26.25-ft) diameter, adaptive system. Near diffraction-limited optical quality is required to avoid additional wavefront errors. The MEO-ERU transmitting aperture is a Cassegrainian, segmented 8-m (26.25-ft) diameter adaptive system. Wavefront error sampling and correction are required. Both the receiver and transmitting apertures are double gimbaled to provide receiving and transmitting from and to any direction. The MEO-ERUs do not have a self-contained propulsion system. Again, the weight difference between the MEO-ERUs for the two payloads is due to the mirror cooling requirements. The total weights for each MEO-ERU is 7,450 kg (16,500 lbm) and 43,000 kg (95,000 lbm), respectively, for the 2268-kg and 148,000-kg payloads.

- GEO Relay Units

The GEO-ERUs for the ground-based transmitter rocket systems are located at geosynchronous-equatorial orbit. The receiving 10-m (32.8-ft) diameter optics are off-axis, segmented, and adaptive. Near diffraction-limited optical quality is required. The 4-m (13.12-ft) diameter transmitting optics are Cassegrainian, monolithic, and adaptive. Wavefront error sensing and correction are required. Both receiving and transmitter apertures are double-gimbaled to provide receiving and transmitting from and to any direction. The GEO-ERU also includes an integral propulsion system to transfer itself to GEO. The total weights are 10,000 kg (22,000 lbm) and 45,000 kg (99,200 lbm), respectively, for the 2268- and 148,000-kg payloads.

2.4 CONCLUSIONS

Laser rocket systems are very attractive when compared to conventional, chemical propulsion systems of equal capability on the basis of 10-year life cycle costs. The development of laser rocket systems will require substantial technology advancements, but the payoff is also substantial as shown in Table III. The cost ratios shown in Table III are the factors by which conventional LO_2-LH_2 systems cost more than an equivalent laser rocket system. For example, the 450 payloads for reusable vehicles and 10 payloads for expendable vehicles (projected shuttle use) show that the LO_2-LH_2 system will cost 2.37 times the cost of a space-based laser rocket system or 2.25 times the cost of a ground-based laser rocket system.

The laser rocket systems are also the best choice from a cost point of view when compared to solar electric propulsion except where the flight activity drops to a low level.

TABLE III. LIFE CYCLE COST RATIOS OF CONVENTIONAL/LASER SYSTEMS

2268-kg Payloads		148,000-kg Payloads		Cost Ratio	
Reusable Vehicle	Expendable Vehicle	Reusable Vehicle	Expendable Vehicle	Space Laser	Ground Laser
450	10	0	0	2.37	2.25
0	0	4500	14	5.90	5.77
400	85	4000	0	5.23	5.15
425	14	8000	0	6.91	6.69

2.5 RECOMMENDATIONS

The continuation of technology development programs directed toward critical technology areas of laser rocket systems is recommended. Specific analytical and experimental programs recommended include:

- Continuation of the CW Rocket Thruster Program
- Initiation of a Laser Rocket System Technology Development and Program Plan Study
- Initiation of a Space-Based Electrical Power Systems Analysis for a power system in the range required by space-based laser rocket systems

Section 3
TECHNICAL DISCUSSION

The Laser Rocket Systems Analysis investigated space-based systems first, followed by ground-based and airborne systems and will be discussed in that order. Task I, Model Definition, is used for all systems regardless of laser transmitter basing. The Model Definition Task included Subtasks A, Mission Model; B, Orbital Model; C, Energy Delivery System Model; and D, Performance Model. No effort was expended in subtasks IB, IC, and ID as these models were already developed. The discussions of these models are contained in the tasks where they were used.

3.1 TASK I: MODEL DEFINITION

In Subtask IA, a mission model was established for the 1990 – 2005 time frame covering missions which would utilize Orbit Transfer Vehicles (OTV) to transport payloads between LEO and GEO or the Outer Planets. This mission model provides an adequate baseline for a systems analysis of the laser rocket concept both for technical and cost analyses.

In the search for deterministic projections of future space activity, few studies were found. Post-shuttle era space activity depends, in part, on the response to the space shuttle in terms of cost effectiveness and exploitation of new possibilities. Additionally, various economic, political, environmental, and military factors will determine the outcome of space utilization concepts presently under consideration. By synthesizing the activities from the studies utilized and organizing the mission model by categories and activity level multipliers, a realistic model has been established.

The primary sources utilized in generating the mission model were:

- Mission Analysis of Future Military Space Activities, 1980 – 2000
- Future Space Transportation Systems Analysis Study, Boeing Aerospace Company for JSC
- Initial Technical, Environmental, and Economic Evaluation of Space Solar Power Concepts, Johnson Space Center
- Feasibility of Space Disposal of Radioactive Nuclear Waste, Lewis Research Center

Other studies as well as current trends were drawn on for additional information in constructing the mission model.

Table IV, the Laser Rocket Systems Mission Model, provides a summary of the missions by category, ΔV requirements, typical payload weight, and nominal steady

TABLE IV. MISSION MODEL – 1995–2005

	Δ VELOCITY (m/s)	Δ VELOCITY (ft/s)	PAYLOAD WEIGHT (kg/lb)	NOMINAL NUMBER PER YEAR	STEADY ACTIVITY PER 10 YR	ACTIVITY LEVEL MULTIPLIERS CASE 3	CASE 6	CASE 8	CASE 11
GEOCENTRIC MISSIONS									
CURRENT PROJECTED									
HI. ELLIP., HI. INC.	2,743	9,000	1,361/3,000	5	50	2	0	1	3
GEOSYNCHRONOUS	4,298	14,100	2,268/5,000 RT	15	150	1	0	1	0.5
ADVANCED			11,340/25,000 R						
GEOSYNCHRONOUS SPACE STATION	4,298	14,100	24,949/55,000 D	10	100	0	5	0	0
GEOSYNCHRONOUS SPS	4,298	14,100	148,000/326,000	400	4,000	0	1	1	2
EXTREME LAT. COVERAGE	6,095	20,000/YR	2,268/5,000	10	100	1	0	0	1
ORBIT MAINT. OF LG. STRUCTURES (LEO)	91	300	45,000/99,000 TO 340,000/750,000	10	100	1	0	2	1
INTERPLANETARY MISSIONS									
CURRENT PROJECTED									
MERCURY ORBITER	5,182	17,000	4,173/9,200	–	4	1	0	1	2
PIONEER SATURN/URANUS/TITAN PROBE	12,192	40,000	499/1,100	–	2	1	0	0	2
ADVANCED									
NEPTUNE JUPITER FLYBY	12,192	40,000	3,175/7,000	–	2	1	0	2	0
URANUS ORBITER – 3.5-YR TRIP TIME	20,117	60,000	907/2,000	–	2	1	0	1	1
NUCLEAR WASTE DISPOSAL	9,144	30,000	4,536/10,000 TO 13,609/30,000	30	300	0	0.05	0.25	0

NOTE: RT – ROUND TRIP PAYLOAD WEIGHT
D – DELIVERED PAYLOAD WEIGHT
R – RETURNED PAYLOAD WEIGHT

state activity. The Mission Model is composed of mission categories and activity level multipliers. There are several advantages to implementing a mission model based on mission categories and activity level multipliers. The most important advantage is that the minimum activity level for specific missions or groups of missions that provide economic justification for the advanced design concepts can be easily defined. Another advantage of the mission-category, activity-multiplier concept is the ability to generically define the missions which, in some cases, are utilized by classified and nonclassified programs alike. The composition of individual programs which make up the activities in any given mission category, while based on overall scenario objectives, can be kept anonymous where required. Furthermore, the role of anonymous missions in justification of an advanced system can be measured in relation to the total activity level required for the allowable development costs to exceed the projected development costs. Finally, it is possible through zeroing out particular mission categories in any one performance assessment to quickly and easily go from current or near-term mission applications to models which reflect the most demanding space utilization considered to date. In this way, special or peculiar missions, which by themselves may have a significant impact on space transportation demand, can be treated separately or as a part of a mission scenario.

Specific activity level multipliers represent various levels of future demand for on-orbit transportation. By applying multipliers ranging from 0 to 5 to each nominal activity level, a high demand for "advanced" interplanetary missions can be modeled or such a demand can be zeroed out. The same is true for all other mission categories. The indicated cases are those selected from the repertoire generated which are useful in sensitivity analyses. Cases not listed were determined to be minor alterations of those listed and deleted from this final report in order to avoid redundancy. In keeping with the conceptual design nature of this study, the values shown for ΔV, payload weight, and nominal activity are approximations to specific missions or average values for a group of individual missions. As such, these values are representative of potential transportation demands but do not necessarily sum up to a specific total mission model or reflect all the different missions in such total models. The significant concept inherent in the mission model format used for this study is the ability to easily and quickly identify those missions and their activity which are required to demonstrate the economic effectiveness of on-orbit laser rocket propulsion systems.

Discussion of each major category of missions shown in Table IV is provided in the following paragraphs.

- Geocentric Missions

Missions included under this heading include all earth orbital operations. The heading includes both current projected and advanced concepts.

Current Projected

Two separate orbital transfer requirements are set forth under this heading. The first type of mission, labeled HI. ELLIP. (highly elliptic), HI. INC. (highly inclined) has a 12-hr period with perigee at approximately 460-km (250-nmi) altitude positioned at the sourthern most latitude of its earth track. This category is one of those given in the Mission Analysis on Future Military Space Activities, 1980 to 2000. The orbit inclination is 63.4°. It is assumed that the payload is initially injected into a 100-nmi circular orbit at 63.4° inclination. The second transportation requirement is a geosynchronous mission. This payload is initially in a 185-km (100-nmi) circular orbit at 28.5° inclination and its destination is a 35,800-km (19,370-nmi) circular orbit at 0° inclination. No special longitude of injection is specified for this mission. The reference for this mission stems from the current Space Shuttle mission model with a 2268-kg (5000-lbm) one-way payload capability representing a design objective of early orbit transportation systems.

Advanced

Geosynchronous Space Station. The Future Space Transportation Systems Analysis Study depicts an 8-man modular geosynchronous space station (GSS). The orbit transfer vehicle needed to supply an 8-man GSS would be required to deliver 25,000 kg (55,000 lbm) to orbit and return with 11,340 kg (25,000 lbm). Typical of low-earth to geosynchronous transfers, the orbit transfer geometry is identical to the current projected missions.

SPS. The technical requirements are not yet specified for the SPS due to uncertainties remaining in design specifications. From the initial technical evaluation* performed at Johnson Space Center, it is clear that it would be necessary to deliver a large mass to geosynchronous orbit yearly. Each SPS would require about 90,000 metric tons (200,000,000 lbm) delivered to geosynchronous orbit from low-earth orbit. If each payload were sized at 225** metric tons (496,000 lbm), it would be necessary to make 400 trips from low-earth to geosynchronous orbit per year. In the projected 30-year span, during which time some 112 satellites would be constructed, the average annual payload weight to be delivered would be about 340,000 metric tons (750,000,000 lbm) necessitating about 1500 flights per year. Typically such a flight requires a ΔV of about 4298 m/s (14,100 ft). The orbital transfer geometry is identical to the current projected missions, however, payload weight was reduced to 148,000 kg (326,000 lb) to avoid high laser power requirements.

Extreme Latitude Coverage. This type mission will provide full-time coverage of extreme latitudes with the minimum number of satellites. Current vehicles providing this type coverage are generally placed in an orbit of 63.4° inclination to maintain apogee at the northern-most latitude with a payload weight of approximately 910 kg (2000 lbm). The projected mission would be in polar orbits with ΔV periodically applied to prevent the line of apsides from rotating. Along with the capability of maintaining this orbit, satellite weights are projected to increase to 2268 kg (5000 lbm).

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*JSC, Initial Technical, Environmental, and Economic Evaluation of Space Solar Power Concepts, August 31, 1976, JSC 11568.
**op. cit. page IV-A-9, Volume II.

Preliminary analysis shows that approximately 6,100 m/s (20,000 ft/s) per year will be required to maintain the orbit. It is estimated that 235 m/s (770 ft/s) will be applied at intervals of 2 weeks.

Orbit Maintenance of Large Structures. The orbit maintenance of large space structures was assumed to be a potential application of orbiting laser rocket propulsion systems due to the large payload weights and large cumulative ΔV requirements. Typical of future large orbiting structures are the solar power satellites (SPS) at syn-eq orbit and the low-earth manned space stations. Since other mission categories already contain the transfer of the SPS from low earth to synchronous altitude and because the model should contain some low-altitude missions, orbit maintenance of low earth orbit space stations was chosen for this mission model. A 12-man station which represents a wide range of future space station applications including manufacturing and various observation functions was chosen as a typical low-earth orbit station. Based on the Boeing study of "Future Space Transportation Systems Analysis Study," it was further assumed that such a station would be at 275-km (150-nmi) circular orbit, weigh between 45,000 and 340,000 kg (99,000 and 750,000 lbm), and have a worst case frontal area as large at 750 m^2 (8,010 ft^2). Under these conditions, a ΔV of 90 m/s (295 ft/s) per year would be required primarily to maintain the stations altitude in the presence of free molecular drag.

- Interplanetary Missions

Missions included in this section are all missions requiring earth escape velocities although in not every case is the destination another planet.

Current Projected

Mercury Orbiter. This interplanetary mission payload is scheduled for 2 missions in 1987 with 4,175-kg (9,200-lbm) payloads. The departure declination is -17.6° and is planned for a C_3 of 18.9 km^2/s^2. The Civil Payload Model for IUS, Marshall SFC 22 January 1976, was utilized to document this mission. While the 1987 date is before the time period of laser rocket systems, it is assumed that similar missions will exist between 1990 and 2000.

Pioneer Saturn/Uranus Flyby. This mission payload is planned with a payload mass of slightly over 500 kg (1100 lbm) and a C_3 of 139 km^2/s^2 which corresponds closely with a departure ΔV of 12,200 m/s (40,030 ft/s). The departure declination is 17.9°. The source for this mission data is the NASA and Civil Payload Model for IUS, Marshall SFC, 22 January 1976.

Advanced Missions

Neptune/Jupiter Flyby. In obtaining typical cases of a medium range ΔV but large payload, numerous contacts were made with JPL, Ames, JSC, and some private vendors. For this particular case, Columbus, a future mission analysis performed at JPL was utilized with a December 1992 departure, a declination of 7.74°, and a departing C_3 of 140 km^2/s^2 the payload would reach Jupiter in March of 1994.

Uranus Orbiter. In obtaining the data for this representative case, many contacts were made with interplanetary mission personnel. Dr. Ed Tindle of the NASA Ames Research Center, was able to provide us with this mission requiring only 3.5 years to Uranus based on a C_3 of 330 km^2/s^2 corresponding closely with departing ΔV of 20,115 ms (66,000 ft/s) on a declination of 21° of the departing asymptote.

Nuclear Waste Disposal. Lewis Research Center, NASA, has completed a study* on the technical aspects of space disposal of nuclear wastes. The nominal destination for nuclear disposal is solar system escape requiring a ΔV of about 12,200 m/s (40,030 ft/s). The Lewis report assumes an annual mission rate of 40 per year. The payload weight per mission, assuming some type of shuttle system to low-earth orbit, would be about 4,536 to 13,600 lbm. The range in payloads is based on the report of two studies. The Lewis report projects a payload weight of 4,536 kg while the Future Space Transportation Systems Analysis Study projects 13,600 kg by grouping several payloads together. By the year 2000, the yearly nuclear space disposal rate could reach 200 missions per year.

The payload weights were grouped into two major categories: those which were roughly 2268 kg (5000 lbm), and the larger category of about 148,000 kg (326,000 lbm). Admittedly, design and operational changes would need to be made before actual implementation of the mission model; however, for purposes of this study it was deemed safe to make the groupings listed above.

3.2 TASK II: PARAMETRIC ANALYSIS, SPACE-BASED LASERS

The purpose of this parametric analysis is to define concepts of space laser rocket systems having the capability to perform the mission model established in Task I, then select two systems with the most potential which will be further defined in Task III.

This analysis considered orbital propulsion missions using laser energy from a remote orbiting laser transmitter to heat a working fluid and provide the necessary thrust to accomplish the missions. During the analysis, an energy relay unit was found to be advantageous at Geosynchronous Equatorial Orbit (GEO) for those missions requiring propulsive maneuvers in that region. As a result, the concepts derived during the analysis have three remotely located units, laser transmitter (LTU), propulsion (PU) and energy relay (ERU), that must interface and interact with one another as a coordinated system. Two system concepts (Table V) resulted with different capabilities because of the wide variations in payload weights established in the mission model. One system was conceptually designed around the current projected geosynchronous mission with a 2268-kg (5000-lbm) payload round trip. This system has excess capability for the less demanding missions. The other system was conceived to perform the advanced Space Power Satellite (SPS) mission to GEO. This mission is only the transporting of 148,000-kg (326,300-lbm) segments of the SPS and does not include other supporting missions.

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*Feasibility of Space Disposal of Radioactive Nuclear Waste, Lewis Research Center, May 1974, NASA TM X-2911, 2912.

TABLE V. SYSTEM SPECIFICATIONS

	Small (2,268 kg) Payload	Large (148,000 kg) Payload
Laser Transmitter Unit		
Laser Device Type	Closed Cycle EXCIMER	Closed Cycle EXCIMER
Laser Power (MW)	16	490
Transmitting Aperture Diameter (m/ft)	30/98.4	30/98.4
Obscuration (ID/OD)	0.2	0.2
Number of Gimbals	2	2
Electrical Power Supply (MW)	131	4,000
Orbit	Circular	Circular
Altitude (km/nmi)	500/270	500/270
Inclination (deg)	28.5	28.5
Propulsion Unit		
Payload (kg/lbm)	2,268/5,000	148,000/326,000
Required Input Power (MW)	13.4	418
Receiving Aperture Diameter (m/ft)	4.25/13.94	4.5/14.76
Obscuration (ID/OD)	0	0
Number of Gimbals	1	1
Velocity Increment (m/s/ft/s)	10,500/34,450	10,000/32,810
Thrust (N/lbf)	1,000/225	31,100/7,000
Energy Relay Unit		
Receiver Aperture Diameter (m/ft)	8/26.25	8/26.25
Obscuration (ID/OD)	0	0
Number of Gimbals	2	2
Transmitter Aperture Diameter (m/ft)	3/9.84	3/9.84
Obscuration (ID/OD)	0.2	0.2
Number of Gimbals	2	2
Integral Propulsion	Yes	Yes
ΔV Capability (m/s/ft/s)	5,250/17,225	5,250/17,225

The parametric analysis from which the above systems resulted was bounded by the statement of work as shown in Table VI. The mission model established additional bounds such as payload weights, ideal velocity requirements, and orbital parameters. With these data as inputs, the parametric analysis of space-based laser rocket systems was accomplished as described below.

TABLE VI. STUDY BOUNDS FROM STATEMENT OF WORK

Parameter	Min	Max
Laser Power (MW)	5	1,000
Wavelength (μm)	0.5	10.6
Laser per Transmitter	1	> 1
Number of Transmitters	1	> 1
Transmitter Aperture Diameter (m)	3	30
Receiver Aperture Diameter (m)	3	30
Engine Thrust (N)	445	133,440
Specific Impulse (N-s/kg)	9,800	19,600

3.2.1 Transmitter Deployment, Transmission Opportunities, and True Velocity Requirements

The LPROP computer program models the orbital dynamics of two space vehicles, one of which is the laser and the other the propulsion unit (PU) in powered flight. The modeling for the PU powered flight includes the velocity losses due to gravity and non-optimum trajectories. Table VII shows the input requirements. The output is variable and may be specified to print results of each integration step or any increments thereof. The output includes the total elapsed time, perigee, apogee, inclination, final thrust/weight, and the total velocity increment. With the LPROP program, a series of runs were made for the first burn to GEO varying laser orbital parameters, thrust/weight ratios, laser range, and true anomalies of both the laser transmitter unit (LTU) and PUs. This provided a matrix of timelines and true velocity requirements from which laser deployment could be selected based on the first burn requirement. With energy transfer ranges of 10,000 to 15,000 km (5400 to 8100 nmi), energy transfer time available did not vary significantly for the lower altitude LTU deployment schemes [500 to 10,000 km (270 to 5400 nmi)] whether the inclination was zero or 28.5°. The curves shown in Figure 6 are typical of the matrix of timelines and true velocities. The velocity increments are highly dependent on the thrust/weight ratios and the specific impulse. The range relative to time is dependent upon the starting orbital parameters and the initial thrust/weight. Energy transfer opportunities (line-of-sight) with ranges of 10,000 km or greater usually last for more than 1 hr on the first controlled phasing of the two vehicles. If more energy is required than is transferred during the first phase, then additional opportunities will occur for various time intervals as the vehicles continue through their orbits and again come into line-of-sight of one another within the range constraint.

TABLE VII. LPROP COMPUTER MODEL INPUTS

Input target perigee altitude (km) 185
Input target apogee altitude (km) 186
Input target inclination (deg) 28
Input target longitude of node (deg) 0
Input target argument of perigee (deg) 0
Input target true anomaly (deg) 0
Input inplane thrust angle (deg) 0
Input out-of-plane thrust angle (deg) 0
Input initial thrust to weight of target 0.01
Input target specific impulse (sec) 2000
Type integration step size (sec) 60
Printout every input computed point (integer) 20
Type stopping code (integer) 2
Input value of stopping parameter 36440
Input max laser range (km) 13000
Input laser perigee altitude (km) 6580
Input laser apogee altitude (km) 6581
Input laser inclination (deg) 0
Input laser longitude of node (deg) 0
Input laser argument of perigee (deg) 0
Input laser true anomaly (deg) 10

The maneuvers required in the vicinity of synchronous altitude were also analyzed using the LPROP computer program. Again, the LTU was put in low, medium, and synchronous altitude orbits with the PU having the orbital parameters attained with the first burn. As the PU is in an elliptical orbit, its velocity near apogee is relatively slow, and the energy transfer opportunity time is long. Also, because out-of-plane thrust was not applied during first burn, the perigee altitude increased and the velocity increment required to change planes and circularize the orbit is lessened. These two events assure that ample time is available for the synchronous maneuvers. The matrix of time lines and velocity requirements for the second burn maneuvers showed that the system was relatively insensitive to the LTU orbit; however, from experience it was recognized that the receiver optics diameter increases with range and

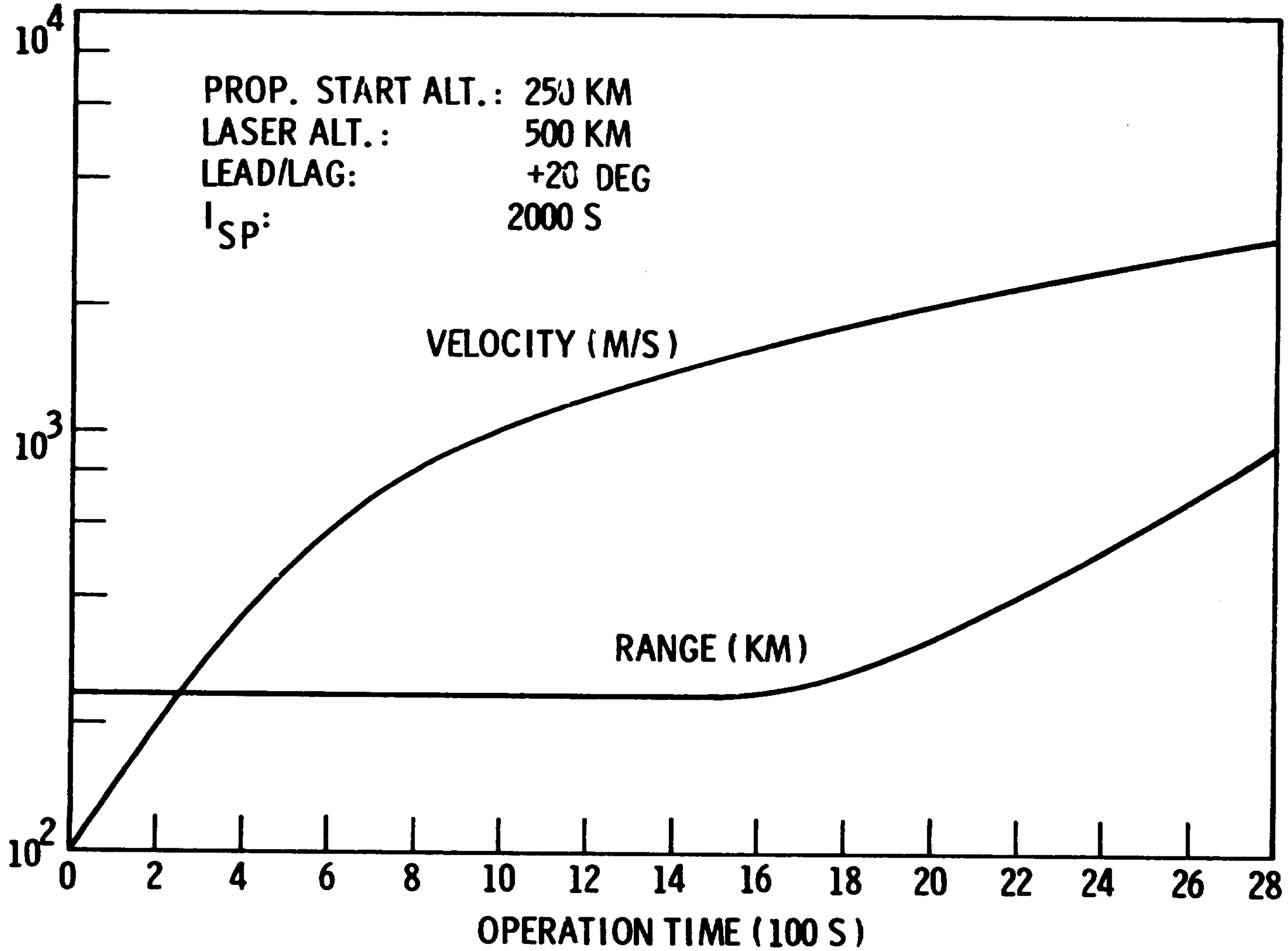

Figure 6. Range and velocity versus time for thrust/weight = 0.1

could be a problem weight-wise at the long ranges from low- or medium-earth orbit to synchronous altitude. The laser rocket system PU is weight critical the same as any other space propulsion vehicle. This indicated that possibly two laser units, one in low-earth orbit and one at GEO, could be useful from a system standpoint to shorten ranges and reduce the PU receiver diameter requirements (many PUs versus two LTU). Also from experience, it was realized that LTU would be the dominating single unit in the system regardless of laser power requirements; therefore, the use of an energy relay unit (ERU) was considered. An ERU would have the disadvantage of reducing the power at the PU thruster because of the second diffraction loss plus the energy absorbed in the ERU mirrors. The loss of laser energy input into the thruster would reduce the specific impulse making the PU less efficient. A quick analysis showed that the specific impulse would be reduced on the order of 16% which would still be an excellent efficiency relative to chemical systems (i.e., 2000 s I_{SP} would be reduced to 1680 s). To simulate the use of a relay at GEO, additional runs with the LPROP program were made with the PU operating at the reduced specific impulse. The burn-time increased as expected, but was still well within the available time. The velocity requirement also increased; however, the impact to the PU (increased propellant) appeared to be acceptable which was confirmed in a later analysis.

A third approach to the energy transfer at synchronous altitudes is the deployment of the laser transmitter in an elliptical orbit so that the apogee is at synchronous altitude (500 by 35,860 km, 10.6-hr orbit). The longer orbital period causes phasing problems between the LTU and PU and significantly increases the time between opportunities to transfer energy. Also, the available time during an energy transfer opportunity at low earth orbit is significantly less. For example, the relative velocity between the PU in a 185-km (100-nmi) circular orbit and LTU in the same orbital plane, but a 500-km (270-nmi) circular orbit is 180.5 m/s (592 ft/s). The relative velocity with the LTU in the elliptical orbit is 2190 m/s (7186 ft/s) at perigee. Another very significant disadvantage is the cost of placing the LTU in the elliptical orbit. This would require consideration of a special vehicle to perform that mission.

The result of the analysis was the establishment of LTU deployment parameters, transmission opportunities, and the true velocities required for various thrust/weight ratios.

- <u>Laser Transmitter Unit Deployment</u>

The LTU is deployed in a circular orbit at 500-km (270-nmi) altitude with an inclination of 28.5°. The altitude is high enough to avoid large drag forces requiring excessive drag make-up capability. The LTU will have a slower inertial velocity than the PU which assures phasing of the LTU and PU so that energy transfer can be accomplished.

- <u>Transmission Opportunities</u>

Transmission opportunities occur when the laser and propulsion are within sight and range. A matrix of transmission opportunities and times available to perform the energy transfer was established for initial thrust/weight ratios from 0.002 through 0.1.

These transmission opportunities can be preplanned to provide maximum available time by scheduling the PU to be in the correct inertial position relative to the position of the LTU and the trajectory to be accomplished. If sufficient time is not available to transfer the required energy at the first planned encounter, then additional opportunities will occur on subsequent orbits.

- True Velocity Requirements

The velocity requirements in the mission model are characteristic velocities associated with each mission and do not account for any losses that occur during the performance of the missions. However, to size a propulsion system to perform a given mission, the losses must be considered. There are four basic types of losses – gravity, thrust alignment, drag, and earth oblateness. Drag and earth oblateness are long-term losses that have a negligible effect to the velocity requirements of orbit changing, and are not considered in this analysis. Gravity losses are related to the thrust/weight (T/W) ratio and can become very significant with low T/W ratios. Thrust alignment losses are related to the angle of thrust required to perform particular maneuvers. The LPROP computer program models the orbital conditions and velocities for power flight incrementally including the losses due to gravity and both in-plane and out-of-plane thrust angles. A matrix of true velocity requirements for the GEO mission was generated for T/W ratios varying from 0.002 to 0.1. Table VIII shows the true velocity requirements for a round trip to GEO from LEO for two T/W ratios. The trajectories chosen were in-plane orbit change to raise and lower the apogee. All plane changes were accomplished near synchronous altitudes. Table VIII also illustrates the penalties incurred in low T/W ratios. Of particular note in Table VIII is the lesser velocity requirement of the T/W = 0.04 case for the maneuvers at synchronous altitude. This occurs because no in-plane thrust angle was used during the in-plane maneuver to maintain a constant perigee resulting in the perigee being much higher and thereby requiring less velocity to circularize the orbit.

TABLE VIII. TRUE VELOCITY REQUIREMENTS

	T/W = 0.1		T/W = 0.04	
	km/s	ft/s	km/s	ft/s
Increase Apogee	2.79	9,154	3.55	11,648
Circularize and Change Plane	1.78	5,840	1.59	5,217
Decrease Perigee and Change Plane	1.79	5,873	1.71	5,611
Decrease Apogee (Circularize)	2.66	8,727	3.10	10,171
Total	9.02	29,594	9.95	32,647

3.2.2 Interactions Between System Units

The space-based laser rocket system is composed of three separated units (LTU, PU, ERU) that must be coordinated to act as a single unit. The most significant interactions concern the relationship between the transmitting and receiving aperture diameters and the pointing and tracking capability. Not only must these be coordinated between units, they must be coordinated with the other subsystems of their individual units. Therefore, it is necessary to examine the total rocket system and the individual unit systems at the same time to assure that decisions for coordination between system units are acceptable within the individual units.

The coordination of the LTU and PU apertures involves consideration of the maximum transmission range, the boresight error, and the total beam spread angle as well as the effect to the LTU and PU as individual units. The same is true in coordinating the LTU and ERU and the ERU and PU. Of these system coordinations, the most sensitive is the coordination between the LTU and the PU, because the relative velocity of the LTU and PU at the lower altitudes reduces the time available to transfer energy and the energy required to be transferred is the highest.

- Coordination of LTU and PU Aperture Diameter

The PU being weight critical would benefit from the receiving aperture being as small as possible since all weight additions are bootstrapping. That is, for each unit of inert weight added to the PU, additional propellant is required to maintain the velocity capability and each addition of propellant requires bigger tanks which adds inert weight which requires more propellant, etc. Therefore, if the penalty to the rest of the total system is not too great, then the PU should have as small an aperture as possible. This means that the LTU aperture must be the maximum 30-m (98.4-ft) diameter specified in the statement of work. Evaluating from that point, the 2σ-central spot diameter was plotted against range for various wavelengths as shown in Figure 7. As may be noted in the figure, the beam jitter, wavefront error, and beam quality are also considered. (Beam quality, as used herein, is the beam spread angle as it exits the laser and is expressed as a factor times the diffraction limited beam spread.) The curves were generated using the following equations:

$$\text{Diffraction Half-Angle } (\sigma_o) = \frac{1.3\,\lambda}{\pi D}$$

where

λ = Wavelength (m)

D = Transmitting aperture diameter (m)

$$\text{Total Beam Spread Half-Angle } (\sigma_T) = \left[Q^2\,\sigma_o^2 + \sigma_j^2 + \left(\frac{0.05\,\lambda}{D}\right)^2 \right]^{1/2}$$

where

Q = Optical quality (defined as a factor relative to diffraction limit)

σ_j = Jitter (m)

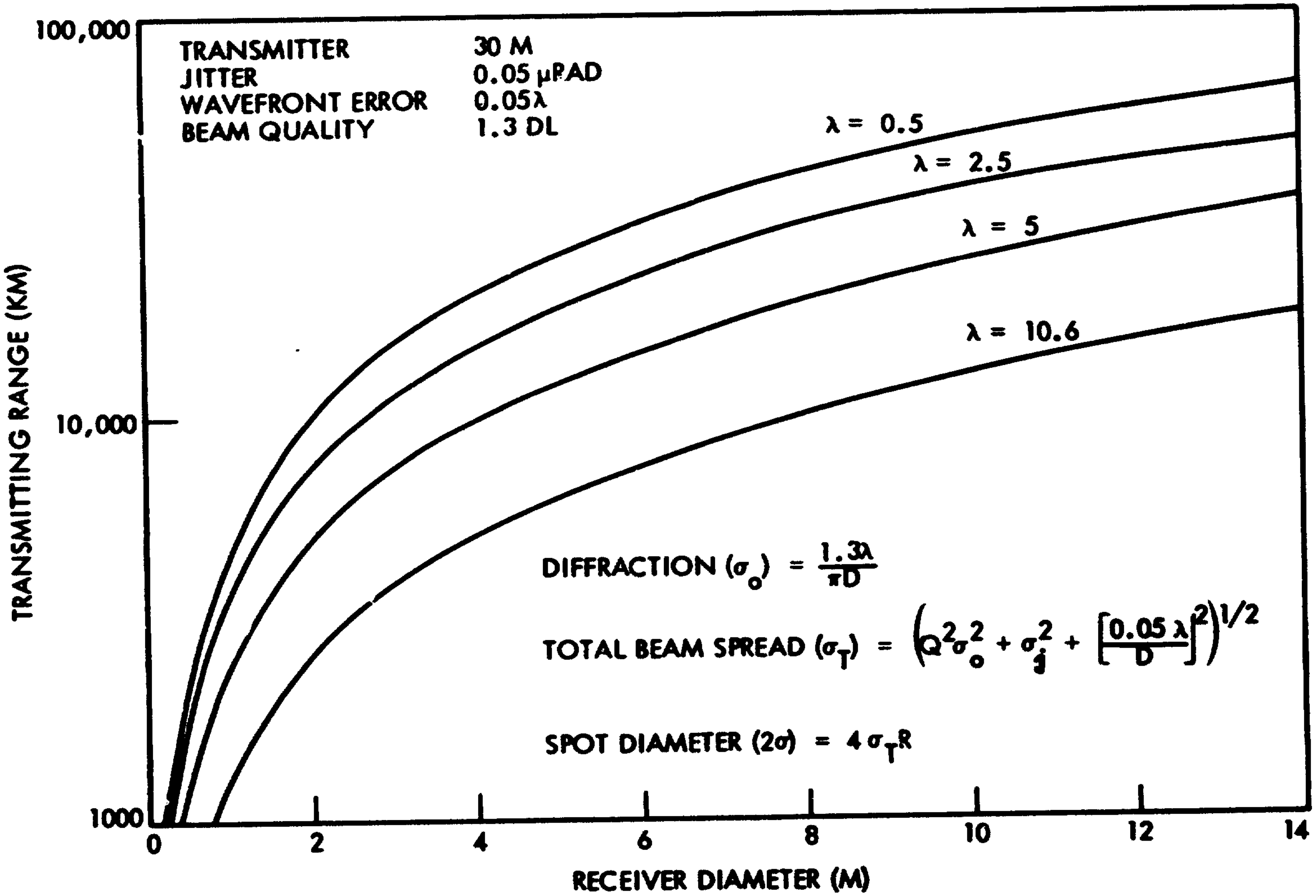

Figure 7. Two-sigma central spot

$$2\sigma \text{ Spot Diameter } (d) = 4\,\sigma_T\, R$$

where R = Range (m)

In selecting a wavelength for the system, several important points must be considered simultaneously – the technology status of the laser device, and the flux density of the energy outside 2σ central spot. The CO_2 closed cyclc laser device would probably require less development to reach the range of power levels to be investigated, but has the disadvantage of a large wavelength ($\lambda = 10.6\ \mu m$) which could require very large receiver apertures on the PUs or additional relay units to assure shorter ranges. The CO laser device ($\lambda = 5.0$) requires a supersonic flow of very cold gasses which causes the weight to increase substantially. The HF chemical laser device ($\lambda = 2.7$) has the highest energy conversion efficiency but has the disadvantage of being open cycle which would require huge quantities of reactants to be transported to low-earth orbit. The EXCIMER-type laser devices ($\lambda = 0.2$ to 0.9) or other types of lasers with like wavelengths have a large advantage with the shorter wavelengths, but the disadvantage of being less developed. The flux density of the outer rings of the diffraction pattern must be examined to assure that flux levels outside the receiver diameter are not detrimental to the structural components on which it could impinge. The flux density of the outer rings is a function of laser power, transmitter obscuration ratio, range, and beam jitter. Beam jitter tends to lower the peak flux density because of the spreading effect to the dark ring areas. Table IX is taken from an output from the DIFPAT computer program which models Gaussian approximation data for obscured circular aperture diffraction. As may be noted, ring 8 has a total power of 113.2 kW and a peak intensity of 0.44 W/cm^2 (2.84 W/in.^{2}) which is enough to cause concern for energy transfer times that may take several thousand seconds of continuous irradiation. Using these type data, energy relay units (ERUs) were synthesized for the primary candidate laser wavelengths (λ). Table X shows the subsystem weights and diameters. Obviously, the wavelength is a strong driver to aperture diameters which is the primary driver of weight.

Upon consideration of the advantages and disadvantages of primary candidate lasers with their respective wavelengths, the shorter wavelength devices (EXCIMER or other type) benefitted the system significantly enough to outweigh their being in the early development stage. Also, the history of laser development shows that power increases tend to be in orders of magnitude rather than factors of 2 or 3. With a 0.5-μm wavelength laser device and a maximum range of 15,000 km (8100 nmi) to the PU, the receiving aperture of the PU could be less than 5-m (16.41-ft) diameter. An ERU at geosynchronous orbit would have a maximum range from the laser transmitter unit (LTU) of approximately 40,000 km (21,600 nmi) which would require an 8-m (26.25-ft) diameter receiver to reduce the spillover energy intensity to an acceptable level.

- <u>Acquisition, Pointing, and Tracking Coordination</u>

The transmitting apertures of the LTU and ERU must have the capability to direct and maintain the laser beam on the receiving aperture. The receiving apertures of the ERU

TABLE IX. DIFFRACTION PATTERN DATA

Assumed laser beam quality = 1.3
Beam quality equivalent wavefront parameter is 5.996443912
Obscuration ratio 0.2
16 MW, 30-m diameter
0.5 μm at a range of 15,000 km
Max intensity is 19301.94526 W/cm^2
1st dark ring, E = 0.2 is 29.16266957-cm radius

Wavefront error is λ over 20, 0.05 μrad jitter

7.587871646 times diffraction limit
Actual peak irradiance is 335.244088 W/cm^2

Lockheed Palo Alto Research Laboratory
3-17-77 Version

Gaussian Approximation Data
for Obscured Circular Aperture Diffraction

Linear Obscuration = 0.20

	Power In Image Element (MW)	Next Diff. Minimum (m)	Power to Diff. Min (MW)
Central Disk	12.2204	0.292	4.7872
Ring 1	2.1847	0.589	8.4586
Ring 2	0.1154	0.772	10.3114
Ring 3	0.6373	1.093	12.7210
Ring 4	0.0540	1.291	13.7141
Ring 5	0.1333	1.558	14.5806
Ring 6	0.1120	1.837	15.0879
Ring 7	0.0169	2.026	15.2919
Ring 8	0.1132	2.345	15.4701
Ring 9	0.0122	2.540	15.5385
Ring 10	0.0393	2.810	15.6005
Ring 11	0.0358	3.086	15.6446
Ring 12	0.0064	3.277	15.6676
Ring 13	0.0456	3.595	15.6974
Ring 14	0.0052	3.790	15.7112
Ring 15	0.0185	4.061	15.7259

TABLE X. RELAY WEIGHT STATEMENT (kg)

	$\lambda = 0.5$	$\lambda = 2.7$	$\lambda = 5.0$	$\lambda = 10.6$
ACQUISITION	102	102	102	102
TRACKER	114	114	114	114
RANGER	23	28	28	28
TRANSMITTER BEAM EXPANDER	742	5,495	17,102	72,079
BEAM EXPANDER DIAMETER (m)	(3)	(15)	(27)	(58)
OPTICAL TRAIN	1,281	1,281	1,281	1,281
GIMBALS AND CMGs	426	3,153	9,815	41,366
RECEIVER TRACKER	114	114	114	114
RECEIVER BEAM EXPANDER	1,937	3,072	5,167	57,319
BEAM EXPANDER DIAMETER (m)	(8)	(11)	(15)	(51)
OPTICAL TRAIN	952	952	952	952
GIMBALS AND CMGs	469	469	469	469
ASTRIONICS	490	573	662	1,034
ELECTRICAL POWER	126	346	692	2,822
STABILIZATION AND CONTROL	191	405	821	3,767
PROPULSION SYSTEM	2,983	6,686	14,322	72,362
TOTAL	9,955	23,065	52,423	267,223

and PU must have the capability to point toward the transmitting aperture with an accuracy sufficient to maintain alignment of the reduced beam diameter on the secondary, and through the transfer mirrors and in the case of the PU for focusing through the thruster window into the thruster heating chamber. The ranges over which the beam is transmitted from the LTU and ERU require pointing and tracking angle accuracies in the submicroradian regime. However, with cooperation between the transmitting and receiving units, the problem is significantly relieved. Several optical configurations, as shown in Figure 8, were investigated and all appear to be capable of meeting the requirements. For example, an acquisition, pointing, and tracking subsystem could be designed around modest power (tens of mW) laser illuminators, centroid detectors, and spectral sharing of the transmitting and receiving apertures. Since only modest amounts of power are required, almost any visible light laser is a viable candidate. Detectors are limited to those which have gain so that signal-to-noise limited operation is possible. One illuminator/detector combination could be an He-Ne laser operating at 6328 Å with an image-intensified silicon quadrand array detector. Both are currently available commercially; however, a means of spectrally sharing apertures efficiently will require some development. This concept operates in both the acquisition and tracking modes. Acquisition can be accomplished in at least two ways, depending upon the original uncertainty of location of the units.

In the first method, Unit A transmits a scanning beacon while Unit B inserts a retroreflector in its optical train. Unit A scans until return radiation is received at which point Unit A proceeds to track. Unit B then removes the retroreflector and locks onto Unit A's beacon and transmits its own beacon for lock-on by Unit A. When both units are locked onto one another, fine tracking begins. This acquisition concept is used when the location uncertainty between the units is large ($> 1.0°$).

When the angular uncertainty is small ($< 1.0°$), acquisition can be simplified considerably. The beacons from both units can be made to cover the uncertainty angle so that both units are irradiated as soon as acquisition begins. In this case, a low bandwidth scanning or staring array of detectors search for the beacon and center it in its field-of-view. When centered, handover to the quadrant array for fine tracking occurs with an increased electronic bandwidth. The preliminary analysis indicates that fine tracking can be accomplished with the expanded beacons; however, if more signal-to-noise ratio is required for fine tracking, the beam divergence angle can be reduced to provide more power.

3.2.3 Propulsion Unit Synthesis

Before a matrix of PUs could be synthesized to determine the weights and performances, several analyses and assumptions were required to establish internal parameters.

- <u>Propellant Selection Analysis</u>

The LASERP computer program designs the tanks, insulation (if required), and propellant management subsystem based on the propellant density, temperature,

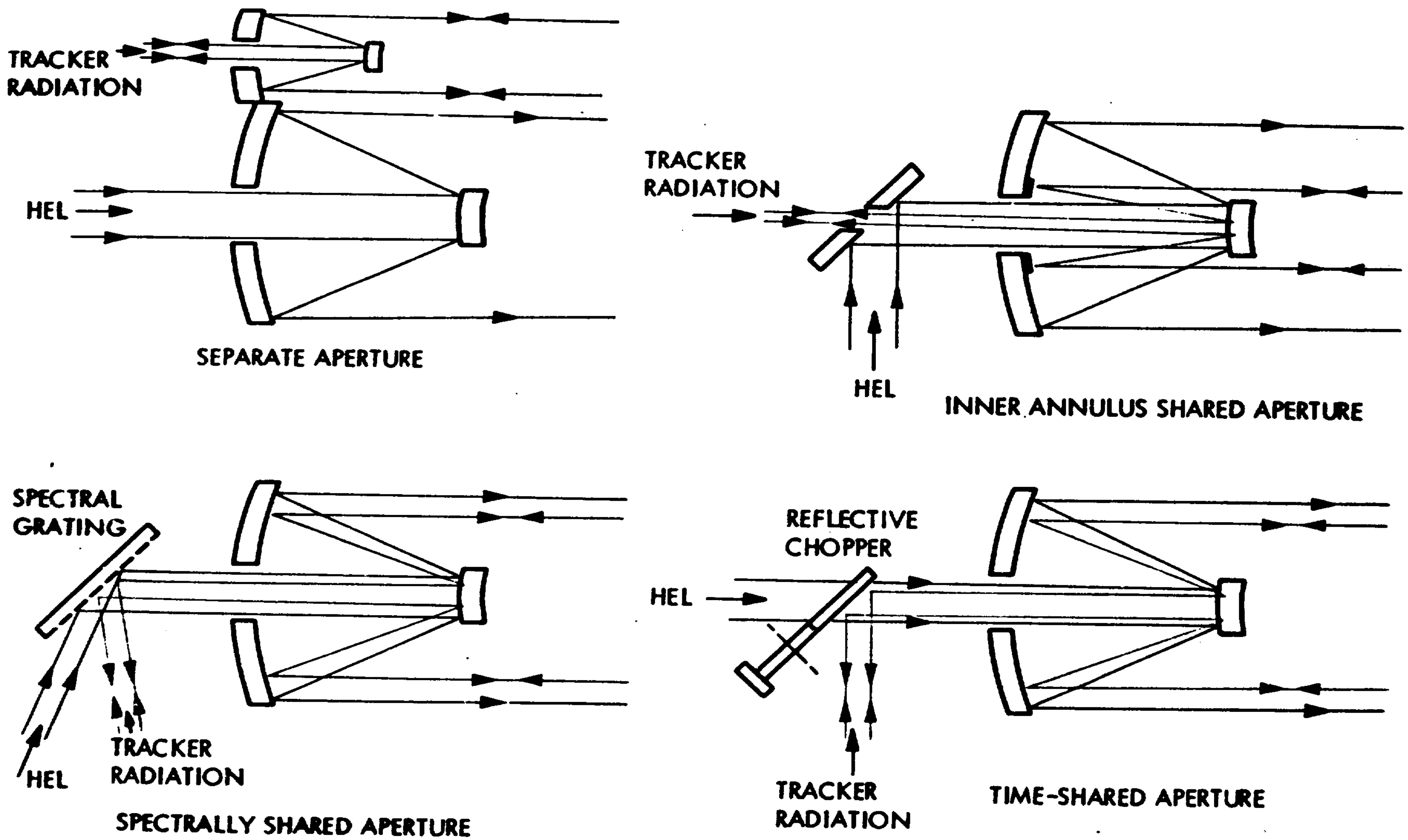

Figure 8. Tracker optical configurations

insulation densities of 32 kg/m^3 (2 lb/ft^3), 5% ullage, minimum tank wall thicknesses based on the diameter, and maximum diameter constraints to assure launch vehicle compatibility. The amount of propellant required is based on the specific impulse, input laser power, and thruster efficiency. Therefore, the selection of a propellant was required so that the associated subsystems could be established and weights derived.

Included as propellant candidates were hydrogen, ammonia, hydrazine, helium, and water. The analysis showed that the only realistic fluid for specific impulses in the range of 9800 to 19,600 N-s/kg (1000 to 2000 lbf-s/lbm) was hydrogen based on data generated from fluid thermodynamic properties, a search of available literature and data made available by thruster manufacturers. The theoretical performance of ammonia and hydrazine will exceed 9800 N-s/kg (1000 sec) at 100% nozzle efficiencies for average gas temperatures exceeding 8000 and 10,000 K, respectively. While detailed laser rocket engine design data for these fluids are not available, it is expected that losses due to realistic efficiencies and other factors such as cooling would reduce performance below 9800 N-s/kg unless average gas temperature in the range of 15,000 to 20,000 K were generated. These high temperatures could result in a laser rocket engine design which would require extensive advances in material technology. Helium is considered to be impractical due to the quantities required and the storage temperature to provide an acceptable density. Water was eliminated due to its performance which was less than ammonia and hydrazine. The NASA Lewis Research Center has been investigating the thruster and tests with a subscale thruster are currently being planned. Plasma's have been formed and maintained in air, argon, and nitrogen. Control of these plasmas has been demonstrated by moving the plasma by changing the focus of the beam. Plasmas with hydrogen have not been sustained as of this writing; however, ongoing tests could accomplish this. If hydrogen plasmas can be formed and controlled as well as with the other gases, then the feasibility of the thruster concept and the use of hydrogen as the working fluid will have been accomplished. Figure 9 shows the specific impulse attainable with hydrogen relative to the temperature. Dissociation of H starts at about 3000 K (5400°R) and increases until about 5000 K (9000°R), at which temperature dissociation is complete. To attain 19,600 N-s/kg (2000 lbf-s/lbm) specific impulse, about 8000 K (14,400°R) average gas temperature is required. This average temperature should not be difficult to get as the plasma is about 18,000 to 20,000 K (32,400 to 36,000°R). The primary problem is one of detail to maintain the thruster walls at an acceptable temperature.

- Engine

One of the more important technologies is the development of a window material that will withstand the required flux levels for relative long periods of time. Considerable work is currently being done, but not at the lower wavelengths. However, even if a material window is not developed, an aerodynamic window may possibly be used with degradation of engine efficiency. The analysis was performed assuming a material window for the engine with an overall engine efficiency of 75%. That is, the loss of thrust relative to the incoming energy would be 25%, due to cooling requirements, nozzle design, etc.

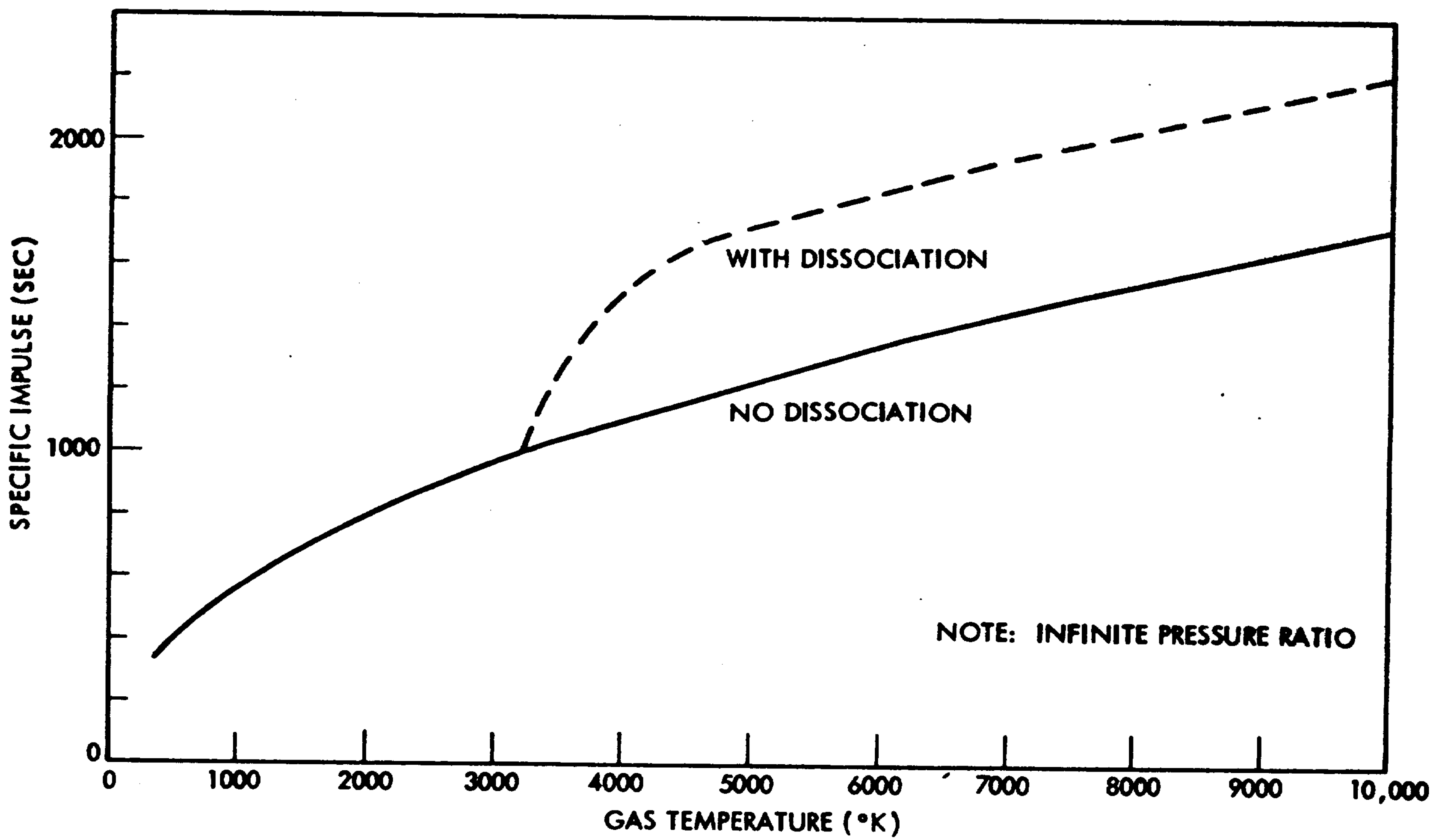

Figure 9. Specific impulse for hydrogen working fluid

- <u>Primary and Secondary Aperture, Optical Train, Mirror Cooling and Structure and Control Moment Gyroscopes (CMGs)</u>

The design of the PU receiving optical system directly impacts the PU capability; therefore, the concept design must be relatively sophisticated to assure that the PU capability can be determined with confidence. The MIRROR computer program models optical systems based on specific design parameters rather than scaling laws and corresponds relatively closely with specific point designs. The MIRROR program is used as a module of the LASERP program which models both PUs and ERUs. Some of the more important parameters are:

- APPLICATION (Ground or Space), affects reflectivity and "g" force acting on system
- PRIMARY MIRROR DIAMETER, affects flux density and thereby thermal capacitance of mirror plate
- MIRROR DELTA TEMPERATURE, the amount of temperature rise permitted during operation which affects cooling system
- MIRROR PLATE GRADIENT, permissible temperature gradient across and through mirror plate which affects cooling rate and mirror figure
- LASER POWER, affects flux density on primary, size of secondary and transfer mirrors, and amount of cooling
- WAVELENGTH, affects reflectivity or absorptivity
- OPERATING TIME, affects total energy absorbed
- MIRROR MATERIAL, affects strength, thermal capacitance, and heat transfer rate
- SLEW RATE, affects structural components and CMG size
- JITTER, affects structural components

With these and other inputs, MIRROR models the optical system by first calculating the primary plate thickness required to contain all the absorbed energy and checks this against the maximum thickness to maintain gradient. If the capacitance thickness is larger than the gradient thickness, a cooled mirror concept is selected. If the capacitance thickness is less than gradient thickness, then the plate weight is checked against the weight of a cooled system with the lighter of the two being selected as the basic design. The number of actuators to maintain figure control is calculated and the reaction structure designed for maximum deflection of 1.5 wavelengths. The reaction structure is an aluminum-honeycomb, beryllium-faceplate composite. Truss weights to hold the reaction structure are then calculated. The secondary weights are calculated similar to the primary except the size is governed by the specified flux density. The secondary support structure is calculated based on an optical f/no. of 1.5, the primary diameter, secondary weight, "g" load, jitter, and slew rate. The cooling system is based on the coolant density, permissible temperature rise, and radiator size. The radiator size is calculated to radiate energy (to average space conditions) in excess of the thermal capacitance of the structural members and coolant. The CMGs are sized based on the mass of that portion of the optical system to be moved during operation and its moment of inertia.

- **Mission Selection for Vehicle Design**

A review of the mission model (Table IV) shows missions with a wide variety of velocity and payload requirements for which individual propulsion vehicles could be optimally designed; however, standardization of propulsion units is required for space programs represented by the mission model to be economically feasible. With the range of payload weights and velocity requirements, standardizing to a single vehicle concept would require a vehicle large enough to carry the 148,000-kg (326,000-lbm) payload to GEO and return empty, which would be extremely inefficient for the smaller payloads with less demanding requirements. A review of the smaller payloads shows the 2268-kg (5000-lbm) payload to be predominant in activity and, when compared to the less stringent missions, has excess capability except for the two advanced interplanetary missions. For example, using the basic rocket equation and derivations

$$V = gI \ln\left(\frac{Wo}{W_{bo}}\right) \qquad \text{or} \qquad \frac{Wo}{W_{bo}} = e^{(V/gI)}$$

where

V = velocity

g = gravity constant = 9.8066 m/s^2

W_o = initial weight

W_{bo} = burnout weight

I = specific impulse

The characteristic velocity for a round trip to GEO is 8,596 m/s (28,200 ft/s) and a specific impulse of 2000 s gives

$$\text{Mass ratio } (r) = \frac{Wo}{W_{bo}} = e^{(V/gI)} = 1.55$$

Using a structural ratio (σ) of 0.44 = $\left(\frac{W_{bo} - W_{PL}}{W_o - W_{PL}}\right)$

$$\text{Payload fraction} = \frac{W_{PL}}{W_o} \quad \frac{1 - r\sigma}{r(1 - \sigma)} = 0.3664$$

where W_{PL} = payload weight

Using the 2268-kg payload

$$W_o = W_{PL}/0.3664 = 6191 \text{ kg}$$

As may be noted, the Neptune Jupiter flyby and Uranus Orbiter missions are beyond the capability of a system designed to carry 2268 kg round trip to GEO. The two mission capabilities can be met in two different ways. One is to develop a larger tank to contain sufficient propellant and the second is to put in series enough of the already developed tanks to provide enough capability. Because there are only a few of these missions, development of a different tank would not be warranted even though the structural efficiency would be better. By putting three tanks together with plumbing for the propellant to transfer, the propulsion system has excess capability of 29 and 11%, respectively, for the two missions.

The vehicle designed to carry the 148,000-kg (326,300-lbm) payload to GEO and return empty could perform the GEOSYNCH Space Station mission (25,000 kg up and 11,340 kg return) with about 45% of the propellant off-loaded. In the case of the Nuclear Waste Disposal, propellant could be off-loaded (~ 50%), but more effective, as the mission requires an expendable vehicle, is to increase the payload weight to capacity of the fully loaded vehicle.

As a result of these analyses based on characteristic velocity requirements, vehicles for the 2268-kg (5000-lbm) payload round trip to GEO and for the 148,000-kg (326,300-lbm) payload to GEO returning empty were selected to synthesize propulsion units over a range of variable parameters to select the most likely candidate for the space-based laser rocket system.

- Propulsion Unit Synthesis

A matrix of propulsion units (PUs) for both the small and large payloads was synthesized using the LASERP computer program and varying parameters for receiver diameter, maximum continuous burn time, specific impulse, thrust, and engine efficiency. All PUs synthesized were sized to perform the true velocity requirement based on the thrust-to-weight ratio. From this matrix, several PUs appeared promising and these were analyzed to determine the best PU subsystem relationship. Subtracting the payload weight, the propulsion system weighs 3923 kg (8648 lbm) and multiplying by the structural efficiency (λ'), the propellant weighs 2197 kg (4844 lbm).

The Mercury Orbiter mission requires 5180 m/s (17,000 ft/s) ΔV for a payload of 4175 kg (7000 lbm). To check the capability of the 2268-kg payload vehicle previously calculated, the resulting propulsion unit weight is added to the Mercury Orbiter weight to get the initial weight.

$$W_o = 3923 + 4175 = 8098 \text{ kg}$$

The burnout weight is obtained by subtracting the weight of propellant (2197 kg).

$$W_{bo} = 8098 - 2197 = 5901 \text{ kg}$$

Using the rocket equation, the velocity increment is obtained.

$$V = gI \ln\left(\frac{Wo}{W_{bo}}\right) = (2000)(9.8066) \ln\left(\frac{8098}{5901}\right) = 6207 \text{ m/s}$$

$$= 20{,}367 \text{ ft/s}$$

This approximation shows a 20% excess capability for a vehicle designed to carry 2268 kg round trip to GEO based on characteristic velocities for both missions. When a vehicle is designed for an ideal velocity (includes losses), the excess capability will remain about the same using true velocity requirements for the Mercury Orbiter mission. Relative capabilities for other missions are:

- Orbit Maintenance of Large Structures (W_{PL} = 317,500 kg) - 47% excess
- Pioneer Saturn/Uranus Probe (W_{PL} = 500 kg) - 10% excess

from a system standpoint. This set of parameters was used to generate the curves in Figure 10 for the small payloads which shows that the total weight and propellant weight are only slightly sensitive to the thrust-to-weight ratio. The driving parameters are the input power required and the burn time which have a large effect on the laser transmitter unit (LTU) and the fleet size required to perform a mission model. The LTU weight (Figure 11) is extremely sensitive to output laser power. The burn time can affect the system in two ways: (1) if the PU takes too long to complete a mission, then extra PUs will be required to perform the entire mission model; and (2) a long burn time per mission with overlapping missions will exceed the total LTU available time requiring an additional LTU. An analysis showed that a trip time (up and back) of 1 week would not require additional PUs or transmitter; therefore, a total burn time of 60,000 sec (~ 17 hr) would be acceptable with the input laser power requirement of 13.4 MW. This results in an initial thrust-to-weight ratio of 0.01. For the small payloads, the thrust is 1000 N (225 lbf) and the large payload thrust is 31,700 N (7000 lbf). Because of the much larger propellant weight of large payload PU and the fact that it returns without payload, the burn time is less since the final thrust-to-weight ratio is higher (large PU final T/W = 0.074, small PU final T/W = 0.021). Based on these analyses, preliminary specifications for the small and large PUs were established as shown in Table XI.

3.2.4 Laser Transmitter-Unit Synthesis

With the input power (P_I) of the PU thruster established at 13.4 MW for the small payload, the losses incurred between the laser and the PU thruster must be calculated to determine the laser output power (P_L). These losses include absorption by the mirrors in the LTU and PU optical trains, secondaries, and primaries as well as the transmission losses due to diffraction and wavefront errors. As previously discussed,

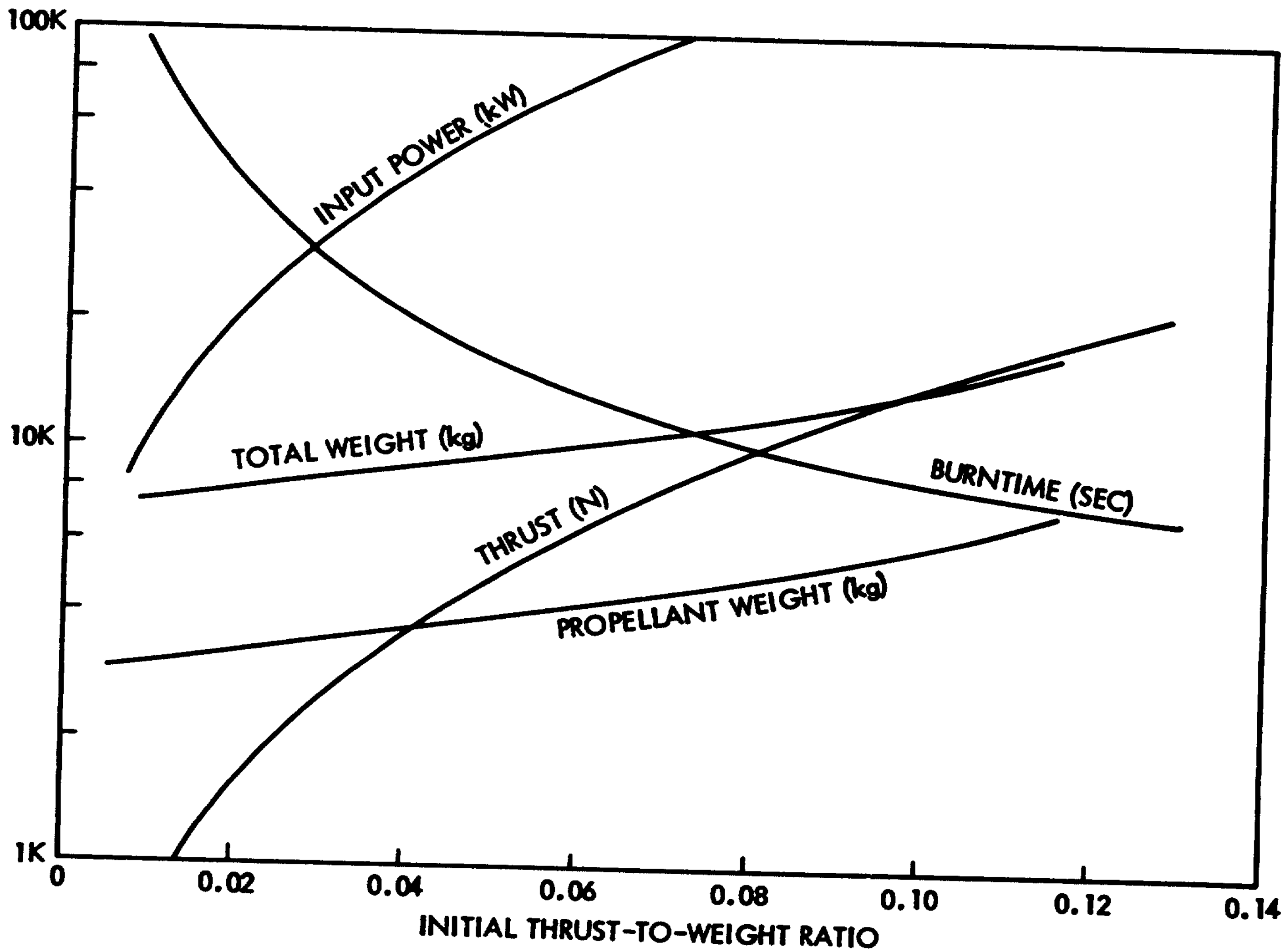

Figure 10. Propulsion unit driving parameter

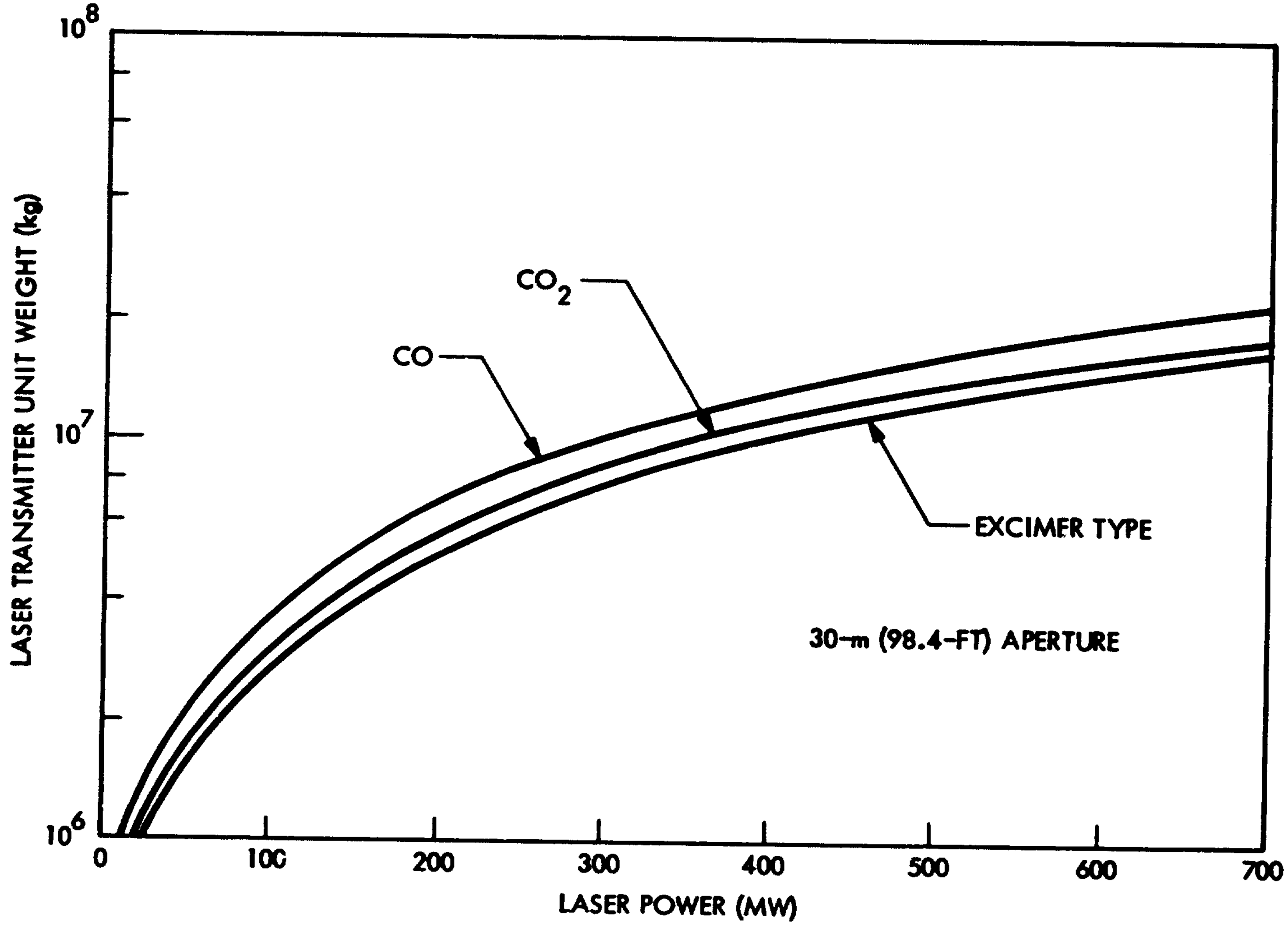

Figure 11. Laser transmitter unit weights

TABLE XI. PRELIMINARY PROPULSION UNIT SPECIFICATIONS

Subsystem	Small PU		Large PU	
	SI Units	English	SI Units	English
Payload (kg/lbm)	2,268	5,000	148,000	326,300
Input Power (MW)	13.4	13.4	418	418
Receiver Aperture Diameter (m/ft)	4.25	13.94	4.5	14.76
Obscuration (ID/OD)	0	0	0	0
Number of Gimbals	1	1	1	1
Ideal Velocity (m/s - ft/s)	10,500	34,450	10,000	32,810
Thrust (N/lbf)	1,000	225	31,100	7,000
Initial Thrust/Weight	0.0135	0.0135	0.0121	0.0121
Final Thrust/Weight	0.0230	0.0230	0.0740	0.0740

a 0.5-μm wavelength was selected for the system because of the system advantage in transferring energy over long ranges. Current projections of coatings for the 0.5-μm wavelength show the expected reflectivity (R) be 99 to 99.7%. The total number (N) of mirrors in the LTU and PU is expected to be 10 and as previously discussed, about 92 to 93% of the energy transmitted must be collected by the receiver to avoid detrimental effects to the PU structure. Therefore:

$$P_L = \frac{13.4 \text{ MW}}{(0.99^{10})(0.925)} = 16.0 \text{ MW}$$

If the coating reflectivity reaches the projected high of 99.7%, then the laser output power could be reduced to about 15 MW. The PU thruster for the large payload requires a laser power input of 418 MW; however, because of the higher power, the diffraction rings have a higher flux density and just over 94% of the energy transmitted has to impinge on the PU receiver. The laser output power for the large payload PU is 490 MW.

Since EXCIMER and other short wavelength laser devices are in their early development stage in laboratories, no factual data exist for high powers. However, indications are that electrical/optical efficiencies and specific powers will be higher than projected for CO and CO_2 devices. For the purpose of this study, the electrical/optical efficiency was assumed to be 20% (as CO_2) and the specific power 80 kJ/kg which is similar to CO devices. The lasing gases are assumed to be circulated by an electric motor driven compressor and cooled by freon in a closed refrigeration loop. The refrigeration horsepower is based upon removing the heat absorbed by the laser cavity and assumes the refrigeration system operates at approximately 300 K temperature with a 35% efficiency.

Silicon solar cells with a solar concentration of 2:1 were selected as the electrical power supply concept for this analysis based on the results of studies by the NASA

Johnson Space Center, Boeing Company and Rockwell International (Refs. 7, 8, 9, and 10). Open cycle electrical power supplies were eliminated because of the need to resupply fuel, and the various types of nuclear power supplies were heavier. Gallium arsenide cells with high solar concentration also appear to be a likely candidate for this application and worthy of a more detailed investigation. They were eliminated from the Space Power Satellite studies because of the limited amount of gallium available.

The optical system for the LTU as discussed earlier should be the maximum diameter within the limits of the study, provided the penalty to the LTU was acceptable and, in fact, increased the total system efficiency. The optical system analysis for the LTU was performed with the MIRROR computer program previously described in section 3.2.3, the primary differences being that the aperture is built up in segments and is Cassegrainian with an obscuration ratio of 0.2. Each segment mirror plate is adaptive and controlled by actuators reacting against an aluminum-honeycomb/beryllium-faceplate structure that is designed for a maximum flexure of 1.5 wavelengths under the most severe loading conditions. Each segment of the aperture is controlled by three larger actuators for positioning relative to one another. A beryllium truss structure support the segments and holds them together. Cooling systems for the 30-m-diameter apertures are not required for either the 16- or 490-MW LTU. The secondaries and optical train mirrors do require cooling as they are sized for a flux density of 15,000 W/cm^2. The cooling system is the primary difference between the optical systems of the 16- and 490-MW LTU.

Based on these analyses, the specifications for the 16 MW and 490 MW LTUs were established and are shown in Table XII.

TABLE XII. LASER TRANSMITTER UNIT SPECIFICATIONS

	Small Payloads		Large Payloads	
	SI Units	English	SI Units	English
Laser Device Type	Closed-Cycle	EXCIMER CW	Closed-Cycle	EXCIMER CW
Laser Power (MW)	16	16	490	490
Transmitting Aperture Diameter (m-ft)	30	98.4	30	98.4
Obscuration (ID/OD)	0.2	0.2	0.2	0.2
Electrical Power Supply (MW)	131	131	4,000	4,000
Orbit Type	Circular	Circular	Circular	Circular
Altitude (km/nmi)	500	270	500	270
Orbit Inclination (deg)	28.5	28.5	28.5	28.5

3.2.5 Energy Relay Unit Synthesis

The energy relay unit (ERU) is placed at GEO to relay the energy from the LTU (in low-earth orbit) to the PU for maneuvers at or near synchronous altitudes. The ERU has a relatively large receiving aperture because of the long range. The ERU reduces the beam size, corrects wavefront errors, refocuses the beam and relays the energy to the PU at relatively short ranges (thereby eliminating the necessity of large receiving apertures on the PUs or the alternate of performing the maneuvers at low specific impulse of an advanced chemical system), and over a 10-year life cycle will substantially reduce the amount of propellant required to be transported to space.

The ERU receiver is sized based on a maximum transmission range of 40,000 km (21,620 nmi), 0.5-μm wavelength, 30-m (98.4-ft) diameter laser transmitter, 0.05-μrad jitter, and 1.3 beam quality factor. Around the ERU receiver, also acting as shields, are photovoltaic arrays that convert the energy of the outer rings of the diffraction pattern to electrical energy for spacecraft functions. This requires an ERU receiver diameter of 8 m (26.25 ft). For the 490-MW transmitter, the flux density in the outer rings is higher and the photovoltaic arrays are extended to pick up additional rings to avoid spillover of critical flux densities.

The ERU transmitting aperture is adaptive and corrects wavefront errors and refocuses the beam. The aperture is sized for 6000-km (3243-nmi) range to provide enough time to complete the synchronous equatorial maneuvers during one pass. This is required to avoid having to wait until the ERU and PU orbit nodes are in phase within the range constraints. This phasing requires days because the orbit periods are close to one another.

An integral propulsion system is included as a part of the ERU so that placement at GEO can be accomplished with the advantage of laser propulsion. The receiver is sized for the synchronous range and the beam is transferred and focused into the thruster. The thruster is identical to the PU thruster; that is, an ERU for a small payload system will have a 1000-N (225-lbf) thruster and for large payloads or a mixture of large and small payloads, the ERU will have a 31,150-N (7000-lbf) thruster. The mode of operation for placement of the ERU is the same as the PU.

The ERU has two cooperative pointing and tracking subsystems. One to interact with the LTU and one to interact with the PU.

The preliminary specifications for the small payload ERU and the large payload ERU are shown in Table XIII.

TABLE XIII. ENERGY RELAY UNIT SPECIFICATIONS

	Small and Large Payloads	
	SI Units	English
Receiver Diameter (m/ft)	8	26.25
Obscuration Ratio	0	0
Number of Gimbals	2	2
Transmitter Diameter (m/ft)	3	9.84
Obscuration Ratio	0.2	0.2
Number of Gimbals	2	2
Propulsion (m/s ft/s)	5250	17,225

3.3 TASK III: CONCEPTUAL DESIGN

The results of the parametric analysis, Task II, were used to further definitize laser rocket system concepts for one system capable of carrying a 2268-kg (5000-lbm) payload from low earth orbit (LEO) to geosynchronous equatorial orbit (GEO) and return to LEO with an equal payload, and the second system with a capability of carrying a 148,000-kg (326,300-lbm) payload from LEO to GEO and return empty. Volume/space allocation analyses were performed, inboard profiles prepared, and preliminary weight estimates established. From these data the technology requirements were assessed and critical technologies identified.

The laser rocket system concepts consist of three units, the laser transmitter unit (LTU), the propulsion unit (PU), and an energy relay unit (ERU), and are discussed in that order.

3.3.1 Laser Transmitter Units

The LTU is space-based in a circular orbit of 500-km (270-nmi) altitude with an inclination of 28.5°. The small payload concept requires a 16-MW laser and a supporting 131-MW electrical power system as shown in Figure 12. The large payload concept requires a 490-MW laser with a supporting 4-GW electrical power supply and is a scaled-up version of the concept shown in Figure 12. Both systems have 30-m (98.4-ft) diameter transmitter apertures. The electrical power supply concept is based on the studies by NASA Johnson Space Center and the Boeing Company (Refs. 7, 8, and 9). The concept uses silicon solar cells with a 2:1 solar concentration and requires 0.972-km^2 (0.375-mi^2) area for 131 MW_e and 29.68-km^2 (11.45-mi^2) area for 4 GW_e. A laser efficiency of 20% plus the electrical power requirement of the cooling and lasing medium compressors were used to size the electrical power output requirements. The power is assumed to be generated at 20,000 V and is inverted to ac current and transformed to the desired voltages for the motors, E-Guns and sustainers. The motors will use the current as ac, and the current will be rectified to dc for the E-Guns and sustainers. The efficiency for conversion to ac and transformation is assumed to be 85%. The efficiency for conversion-transformation-rectification is assumed to be 80%.

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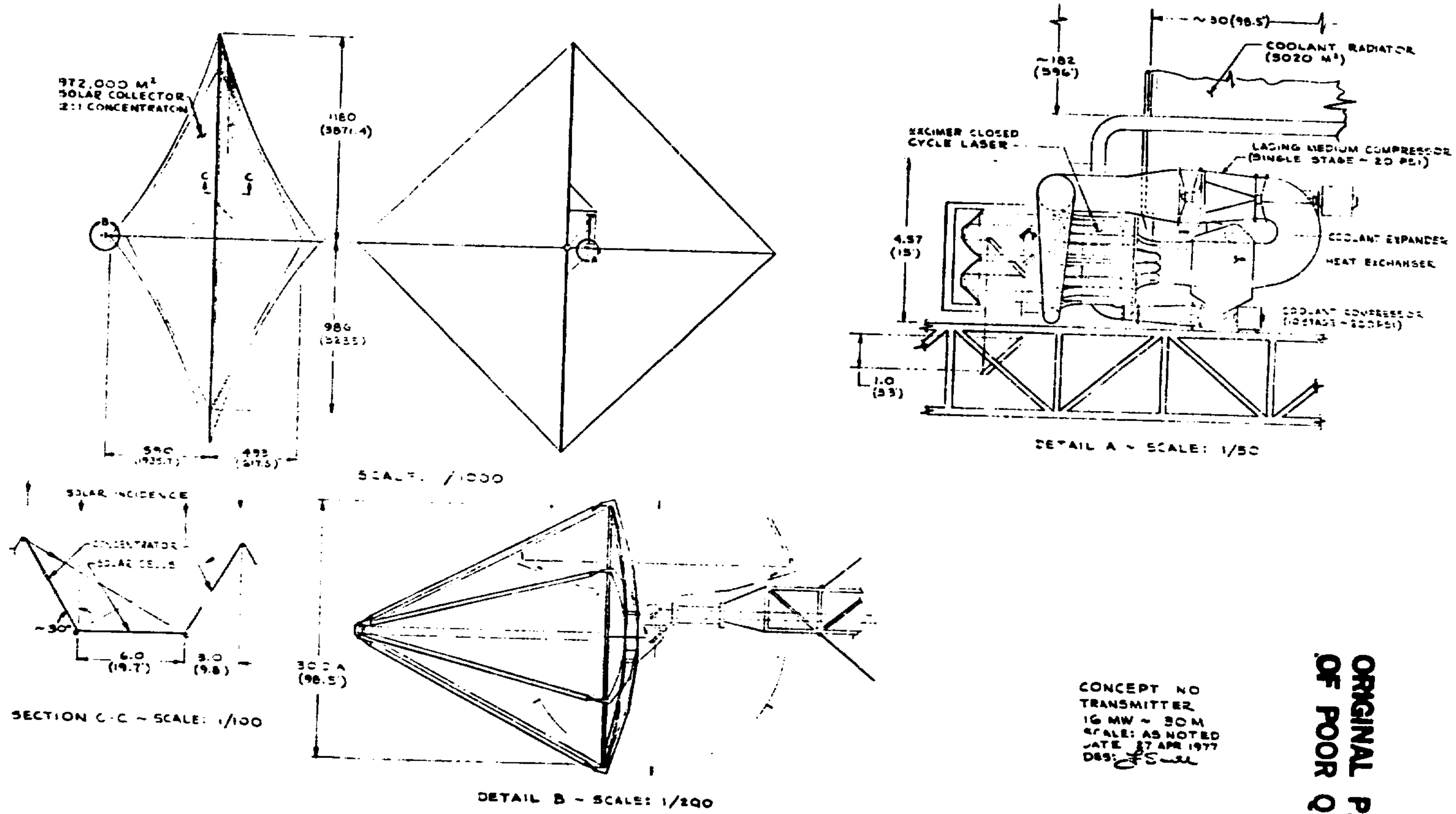

Figure 12. Space-based laser transmitter concept

The laser specific power is assumed to be 175 kJ/kg (80 kJ/lbm) similar to the CO_2 EDL and the electrical/optical efficiency is assumed to be 20%. The cavity flow is subsonic and circulated by an electric motor driven compressor and cooled by freon in a closed refrigeration loop. The refrigeration system horsepower is based upon removing the heat absorbed in the laser cavity with the system operating at 300 K temperature at 35% efficiency.

The primary aperture is 30-m (98.4-ft) diameter made up of segments which are smaller than 4.5 m (14.77 ft) across the longest dimension. This is required for compatibility with the shuttle bay dimensions in the case of the small payload system, plus the 4.5-m dimension is presently being projected as the largest diameter that can be manufactured with the optical tolerances required. The solid mirror plate for the 16-MW system is 0.64-cm (0.25-in.) thick which has sufficient thermal capacity for the total energy absorbed during a continuous operation of 10,000 s. During this period, the mirror plate temperature is permitted to rise some 110°C (200°F). The mirror plate for the 490-MW system is a cooled system with coolant passages through which the coolant flows to maintain the specified gradient (1.1°C or 2°F) through and across the mirror plate while permitting the overall temperature to rise 110°C. The cooling requires 4465 liter (1180 gal) of coolant and a radiator area of 525 m^2 (5660 ft^2) to radiate the excess energy to space. The mirror plates are supported on a reaction structure by close tolerance actuators which maintain the mirror figure so that minimum aberrations are present in the beam as it leaves the aperture. The reaction structure is an aluminum honeycomb/beryllium-faceplate sandwich structure 40.64-cm (16-in.) thick designed for a maximum flexure of 1.5 wavelengths (0.75 μm) under the most severe loading conditions. The mirror segments and their reaction structures are mounted on a beryllium or composite truss structure with actuators to maintain the segments in proper relationship to one another. Figure 13 illustrates the primary aperture concept. Two primary apertures are required to provide 4-π sr pointing capability. The outgoing beam from the primary aperture is sampled for three reasons. One is to measure the amount and direction of jitter so that jitter perturbations can be minimized by actuation of one or more agile mirrors in the mirror train. The second reason is to measure the wavefront aberrations so that corrections can be made by adjusting the fine actuators. The third reason for sampling the outgoing beam is to determine the amount of, and to correct, the boresight errors. A Tracker/Beam Controller subsystem interprets the beam aberrations with algorithms to determine amount and time of corrections and provide commands to the agile mirrors, fine actuators, and tracker subsystem.

The optical train and secondary mirrors are sized for a flux density of 15 kW/cm^2 which requires a cooling system. The secondary diameters for the 16- and 490-MW systems are 36.85 cm (14.5 in.) and 204 cm (80.3 in.), respectively. The optical train mirrors, including those that are agile, are flat elliptical shapes with the major axis larger than the secondary diameters by a factor of 1.414 because of the angle at which the beam hits the mirror. The cooling requirements for all these mirrors is considered to be the same even though the flux level is reduced slightly at each succeeding mirror by the absorption of the previous mirror. Each mirror requires about 86 liters (23 gal) of coolant and 20 m (215 ft^2) of radiator surface.

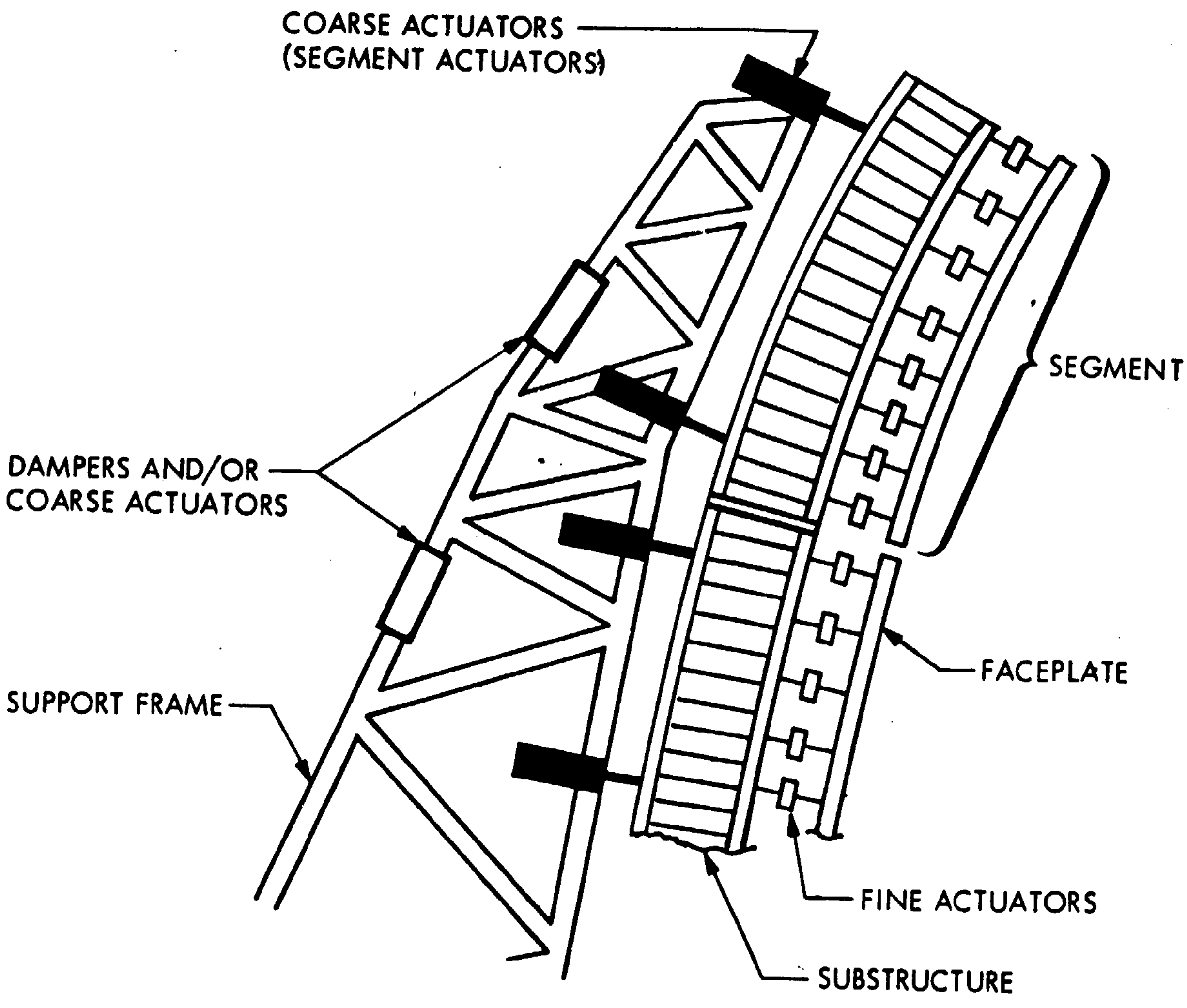

Figure 13. Segmented mirror concept

As may be noted in Figure 12, the gimbals have been designed to minimize the number of transfer mirrors required. The two gimbals provide 2-π sr pointing capability for each primary which together permits transfer of the energy to any direction without regard to altitude of the solar collector.

Both the 16- and 490-MW laser transmitter units will require on-orbit assembly as the total weight and volume exceeds the shuttle capability in the case of the 16-MW system and the heavy lift launch vehicle for the 490-MW system. Table XIV presents the estimated weights by subsystem for the two systems. Each subsystem weight includes the structure to tie it together with the other subsystems. The secondary and support structure are included in the beam expander weight. Cooling system weights are included in the beam expander and optical train as applicable.

3.3.2 Propulsion Units

- <u>Propulsion Unit for 16-MW System</u>

The propulsion unit concept for the 16-MW laser rocket system is sized to carry a 2268-kg (5000-lbm) payload round trip from LEO to GEO. The specific impulse for the perigee burns to raise the apogee going up and to circularize at LEO is 19,600 N-s/kg (2000 lbf-s/lbm). The maneuvers at or near synchronous orbit to change plane and circularize going up and change planes and lower perigee going down are accomplished with a specific impulse of 16,500 N-s/kg (1680 lbf-s/lbm). This reduction in specific impulse is a result of energy loss due to beam diffraction and the absorption of energy by the relay mirrors

Figure 14 shows an inboard profile of the propulsion unit. The dimensions, with the receiver stowed for transport to LEO, are 8.88 m (29.14 ft) in length and 4.57-m (15.0-ft) diameter which provides for the accommodation of two units per shuttle flight from earth to LEO.

The propellant (LH_2) tank is aluminum with 10 cm (4 in.) of multilayer super insulation. The tank is thermally isolated by filament-wound, low-thermal conductance struts with penetration plumbing being stainless steel to inside the insulation. With proper chill-down during propellant loading and the possible use of slush hydrogen, propellant losses for the 1-week round-trip time are negligible. A propellant management device to maintain liquid for restart in zero gravity is not required because the laser energy will react with the hydrogen liquid or gas in the thruster and orient the liquid immediately.

The thruster or engine is designed for 1000-N (225-lbf) thrust with engine cooling accomplished by using the propellant as a coolant and controlling the flow of propellant inside the engine to maintain acceptable temperatures on the engine walls. The beam entrance into the engine is through a material window which could be used to establish a relatively short focal length so that flux intensities high enough to form a plasma are limited to very near the focal point. The propulsion unit concept shows the laser beam

TABLE XIV. WEIGHT STATEMENT FOR LASER TRANSMITTER UNITS

	16-MW Unit		490-MW Unit	
	kg	lbm	kg	lbm
Acquisition (2)	58	128	58	128
Tracker (2)	298	657	298	657
Ranger (2)	70	154	70	154
Beam Expander (2)	39,274	86,583	62,034	136,760
Optical Train (2)	1,376	3,034	27,040	59,612
Gimbals and CMGs	15,594	34,379	24,630	54,299
Astrionics	272	600	1,059	2,319
Fire Control Computer	194	428	194	428
Spacecraft Electrical Power	2,124	4,683	6,866	15,127
Laser Device	3,204	7,064	59,183	130,425
Compressor and Motor	2,238	4,934	6,291	13,869
Refrigeration	9,238	20,366	177,650	391,647
Laser Electrical Power Supply	601,886	1,326,918	11,348,300	25,018,462
Power Conditioning	6,668	14,700	188,643	415,882
Stabilization and Control	2,965	6,537	34,317	75,655
Totals	685,459	1,511,165	11,936,633	26,315,500

entering the side of the thruster. This eliminates the requirement for one transfer mirror and its associated cooling; however, should this requirement penalize the thruster design too severely, another transfer mirror would be added.

The receiver aperture is an off-axis, adaptive optical system. An off-axis system is required so that the incoming beam is unobscured. The adaptive optical surface is primarily to control the mirror figure as no wavefront correction is required. Wavefront errors affect the beam spread and as the beam transmission is limited to a few meters; the affect of additional beam spread is negligible. The primary mirror faceplate is a single unit backed by coolant passages. Fine adjust actuators maintain the figure by reacting against an aluminum honeycomb/beryllium-faceplate sandwich structure. This reaction structure is designed for a maximum flexure of 1.5 wavelengths under the most severe loading conditions. The mirror plate and supporting structure folds toward the secondary structure which folds behind the engine during transportation in the shuttle. The secondary mirror is sized for a flux density of 15 kW/cm^2 and requires cooling. The single transfer mirror also focuses the beam and is agile to minimize jitter components.

The cooling system maintains the specified gradient (1.1°C or 2°F) both through and across the mirror. Because of the long irradiation times, a radiator is required to expell the excess energy to space. The radiator, plumbing, coolant, and pump weights are included in the weights for the beam expander and optical train.

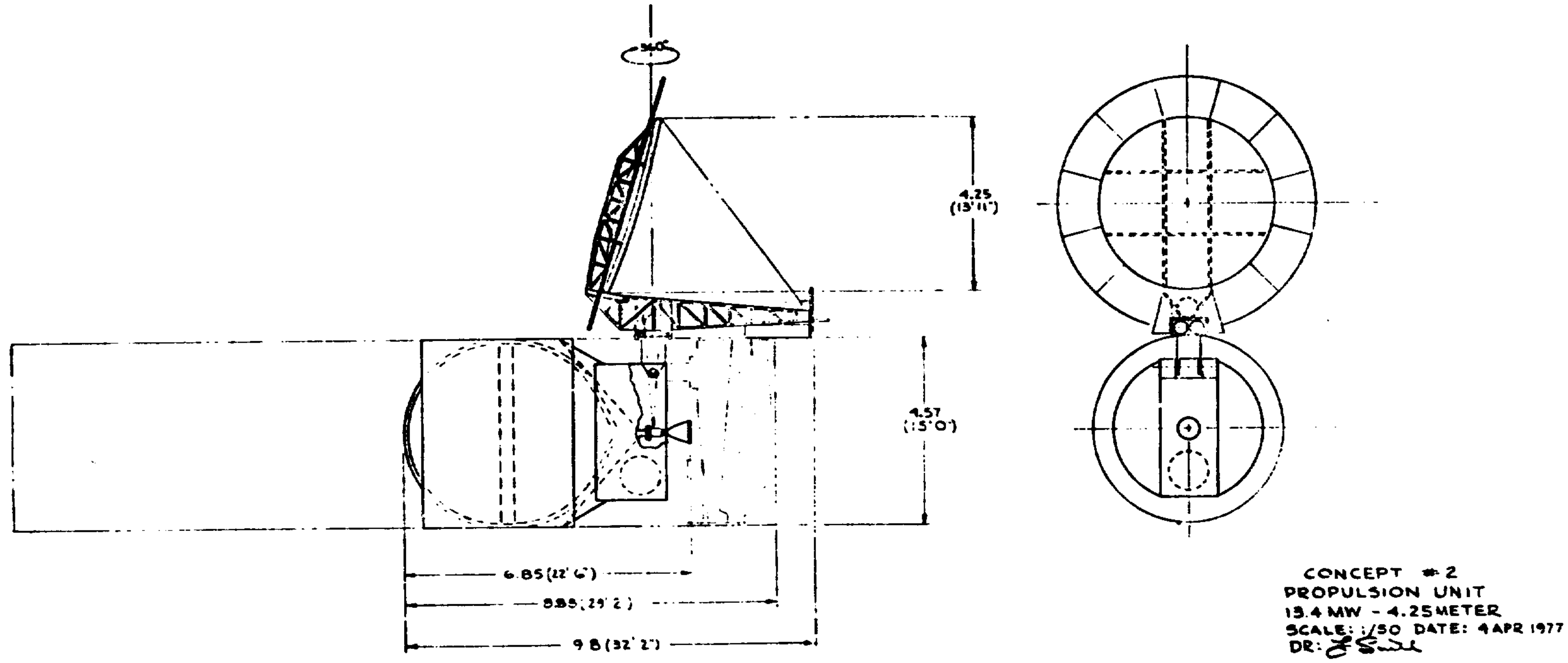

Figure 14. Propulsion unit concept for 16-MW system

Photovoltaic arrays are placed around the receiving aperture to pick up the outer diffraction pattern rings with flux levels that may be detrimental to the propulsion unit structure. This laser energy is converted to electrical energy for spacecraft functions. Additional protection for accidental energy spillover is accomplished by a highly reflective coating (at laser wavelength) applied to the structure as well as detectors to warn of detrimental flux levels. However, the additional safeguards should be seldom if ever required because of the pointing and tracking accuracies attainable with the cooperative system.

The spacecraft function hardware required for the propulsion unit are state-of-the-art and require no technology development. The only exception might be the photovoltaic devices to provide electrical power. As a result, minimum effort was expended on this subsystem.

The propulsion unit weight, by subsystem, was estimated and is presented in Table XV. These weights with the 1000-N (225-lbf) thrust engine, provide an initial thrust-to-weight ratio of 0.0193 and a final thrust-to-weight of 0.047. These ratios are higher than established in the parametric analysis which provides some measure of conservatism.

TABLE XV. WEIGHT STATEMENT FOR 16-MW PROPULSION UNIT

Subsystem	kg	lb
Receiver Beam Expander	614	1,354
Optical Train	142	313
Gimbals and CMGs	149	328
Tracker	77	170
Astrionics	123	271
Electrical Power	78	172
Propulsion System, Dry	476	1,049
Propellant	3,131	6,903
Stabilization and Attitude Control	123	272
Structure	378	833
Total	5,291	11,665

- Propulsion Unit for 490-MW System

The propulsion unit (PU) concept for the 490-MW laser rocket system is sized to carry a 148,000-kg (326,000-lbm) payload to GEO and return empty in support of projected programs such as the Space Power Satellite. As for the 16-MW system propulsion unit, the specific impulse for LEO maneuvers is 19,600 N-s/kg (2000 lbf-s/lbm) and 16,500 N-S/kg (1680 lbf-S/lbm) for GEO maneuvers.

Figure 15 shows the inboard profile of the propulsion unit concept. This concept assumes a heavy lift launch vehicle (HLLV) (Ref. 8) is available for transportation to

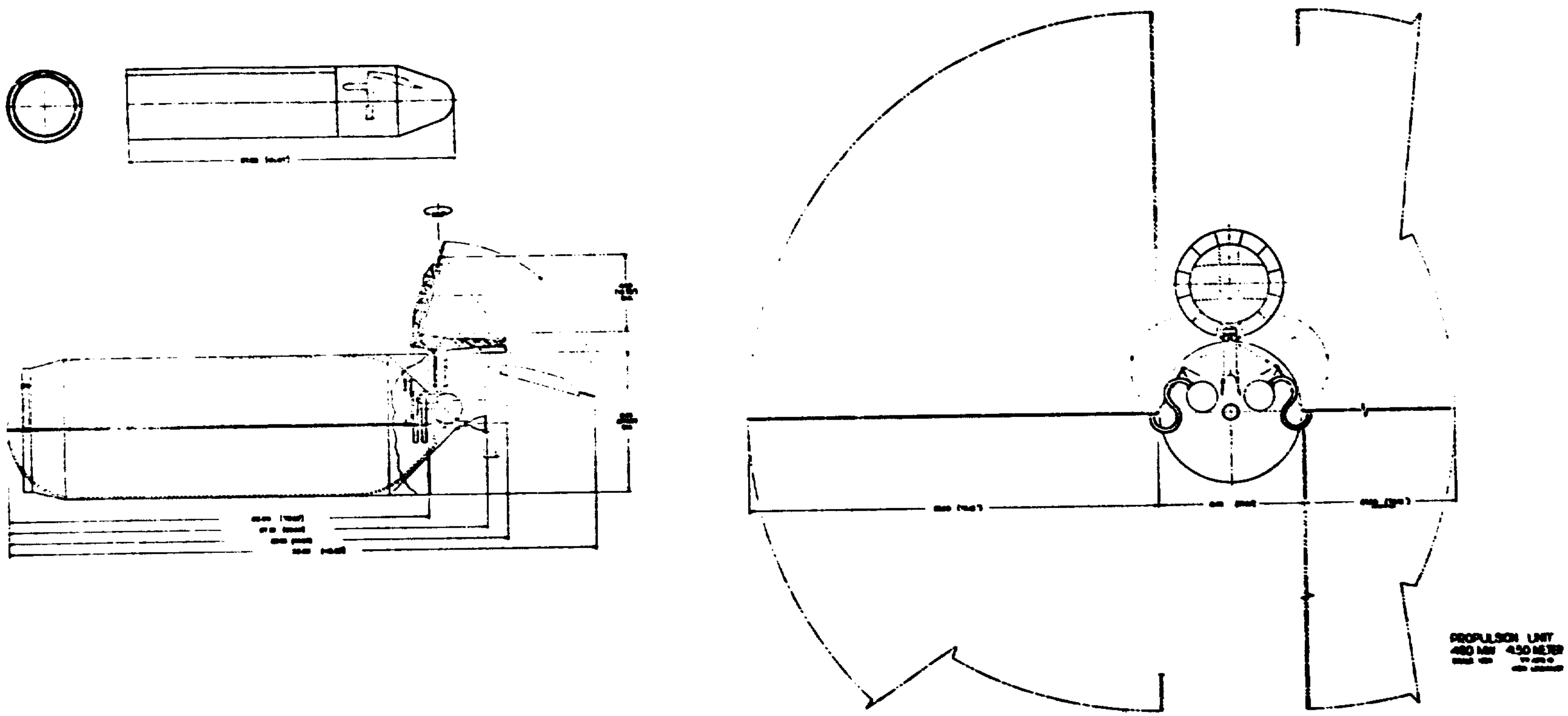

Figure 15. Propulsion unit concept for 490-MW system

LEO and is not restrained by shuttle capabilities. The PU is 8.23-m (27.0-ft) diameter and 33.63-m (110.29-ft) long in the stowed position for transport. The addition of a nose fairing for protection during ascent extends the length to 37.08 m (121.67 ft). In the deployed position, the receiver aperture extends outside the vehicle diameter to provide 360° rotation. Because of the high laser power required, the mirror cooling radiator size is predominate and is designed to roll out upon loading with the coolant stored in tanks in the aft portion of the vehicle. These radiators deploy to both sides of the vehicle and each can be rotated 180° for best emission toward black space.

The propellant tank is designed around the material properties of aluminum with 10 cm (4 in.) of multilayer super insulation for thermal protection. Additional thermal isolation is provided with filament-wound struts with low thermal conductivity and material changes from aluminum to low conductive materials for all penetrations. A propellant management device for maintaining liquid for restart in zero gravity is not required because the laser energy will react with liquid or gaseous hydrogen to produce thrust which will orient the propellant.

The engine is designed for 31,150-N (7000-lbf) thrust with engine cooling accomplished by using the propellant as a coolant and controlling the flow of propellant inside the engine to maintain acceptable temperatures on the engine walls. The beam entrance is from the rear of the engine through a material window.

The receiver aperture is an off-axis system so that the incoming laser beam is unobscured. Adaptive optics are used to control the mirror figure. No wavefront error correction is required. Again, wavefront error causes beam spread which will negligibly effect the beam diameter over the short transmission path to the engine. The primary mirror plate is a single unit backed by coolant passages. Fine adjust actuators maintain the mirror figure by reacting against an aluminum honeycomb/beryllium-faceplate sandwich structure. This reaction structure is designed for a maximum flexure of 1.5 wavelengths under the most severe loading conditions. The entire optical system folds down for transporting to LEO. The secondary is sized for 15 kW/cm^2 and requires cooling as do the primary and transfer optics. Two transfer mirrors are required for this concept of which one or both may be agile for jitter correction and alignment.

As in the 16-MW concept, photovoltaic arrays surround the receiver mirror to collect the energy in outer rings of the diffraction pattern and convert this laser energy to electrical energy for use in spacecraft functions. One major difference between the two PUs is that the reflective coating on the structure is required for the 490-MW-system PU. This coating is required because the much higher laser power results in higher flux intensities on all outer rings, and to provide photovoltaic arrays or other shielding beside the coating would be impractical. The flux density is sufficiently high to cause concern out to the fourteenth and fifteenth rings.

The propulsion unit weight for the 490-MW laser rocket system was estimated by subsystem and is presented in Table XVI. The initial and final thrust-to-weight ratios are 0.028 and 0.074, respectively, which again is some measure of conservatism.

TABLE XVI. WEIGHT STATEMENT FOR 490-MW PROPULSION UNIT

Subsystem	kg	lb
Receiver Beam Expander	3,662	8,073
Optical Train	6,992	15,415
Gimbals and CMGs	887	1,955
Tracker	77	170
Astrionics	125	276
Electrical Power	74	163
Propulsion System, Dry	10,688	23,563
Propellant	71,726	158,127
Stabilization and Attitude Control	4,451	9,813
Structure	15,937	35,135
Total	114,619	252,689

3.3.3 Energy Relay Units

The energy relay units (ERUs) for both the 16- and 490-MW laser rocket systems operate at geosynchronous equatorial orbit (GEO) to relay the beamed energy from the laser transmitter at low-earth orbit (LEO) to the propulsion unit near synchronous altitude. This relieves the propulsion units of the requirement to have large receiving apertures required for the long range from LEO to GEO. The ERU receives the laser beam, corrects wavefront errors, focuses the beam, and directs it to the propulsion units.

Figure 16 shows the inboard profile of the ERU for the 16-MW laser rocket system. The difference between the 16-MW system ERU and the 490-MW system ERU is the propellant tank size and radiators. The predominant radiators are the result of the higher laser power and the amount of energy to be expelled. The larger propellant tank is the result of the extra weight of the mirror cooling system. The space radiator has been omitted in Figure 16 for clarity, but is mounted along the sides of the propellant tank as shown for the 490-MW system propulsion unit in Figure 15. For both ERUs, the coolant is carried to LEO in a container and upon deployment is pumped into the cooling system. The integral propulsion system requires deployment at LEO so that the receiving aperture can receive and direct the energy into the thruster for self-propulsion to GEO. The integral propulsion provides a means of transport to GEO at laser propulsion efficiencies with minimum cost when compared to the alternative of a chemical system. The thrusters are the same as used in the respective propulsion units.

The receiving apertures are segmented off-axis optical systems to receive an unobscured incoming beam. The aperture is adaptive to maintain figure control. This control for figure must be near-diffraction-limited performance to avoid inducing additional wavefront errors which the transmitter optics must correct. The receiving

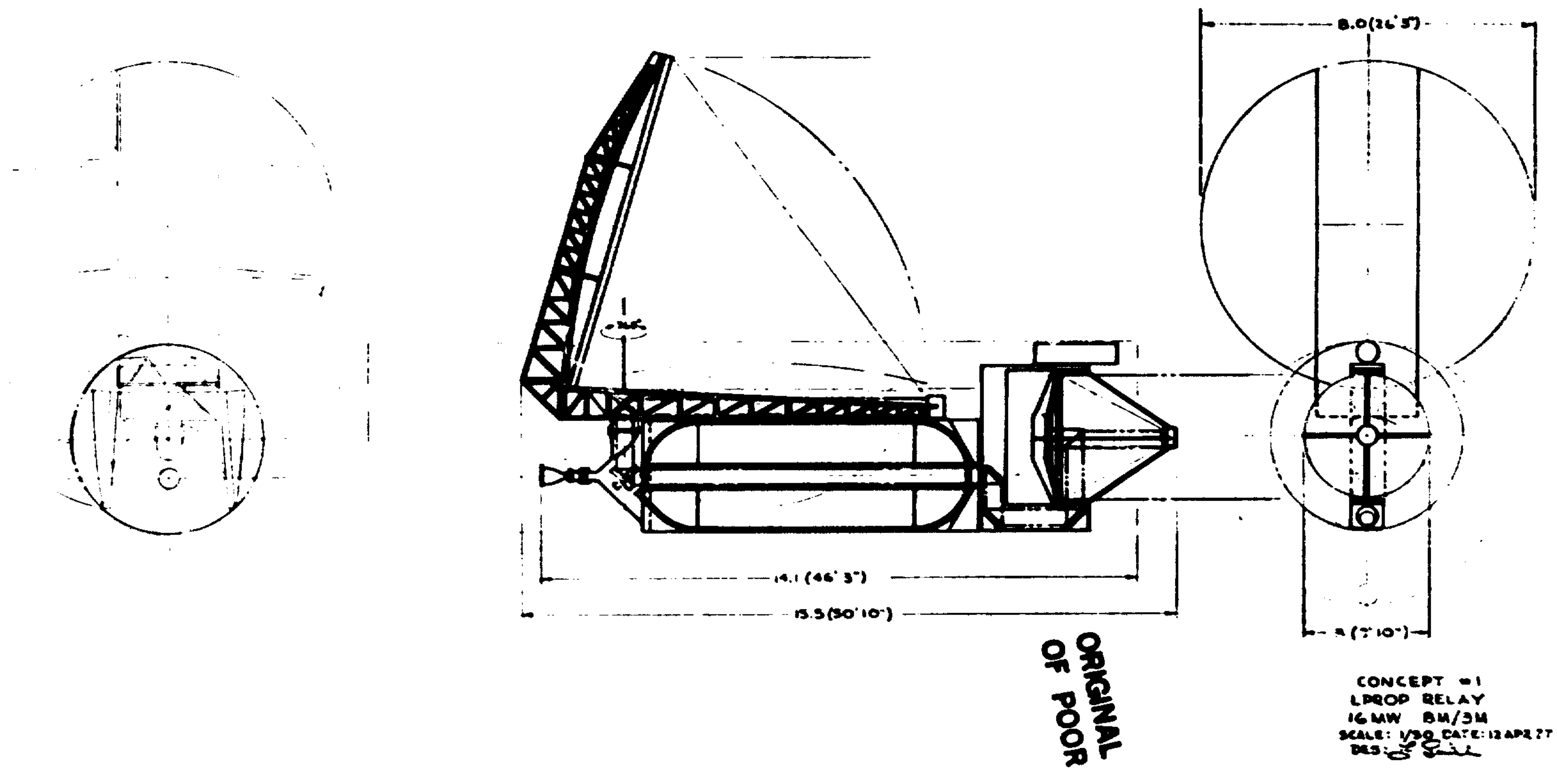

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Figure 16. 16-MW energy relay unit

aperture reduces the beam diameter and directs it to the secondary which redirects the energy through the optical train. The secondary is sized for 15 kW/cm^2 (96.8 kW/in.^{2}). The second transfer mirror as shown in Figure 16 is reflective on both sides and can direct the energy through the normal train to the transmitting aperture, or, by relating 90°, can focus the beam and direct it into the engine during propulsive maneuvers.

The transmitter aperture for both the 16- and 490-MW systems is an adaptive, monolithic, Cassegrainian system. Cooling is required for both systems. The transmitter aperture is double gimbaled to provide greater than 2-π sr pointing capability and when coupled with the receiver and vehicle attitude provides the capability of receiving and pointing from and to any direction.

Table XVII shows the weights by subsystem of both the 16- and 490-MW system energy relay units.

TABLE XVII. WEIGHT STATEMENTS FOR ENERGY RELAY UNITS

	16-MW System		490-MW System	
	kg	lbm	kg	lbm
Transmitter				
Beam Expander	561	1,237	5,597	12,339
Optical Train	675	1,488	11,834	26,089
Gimbals and CMGs	494	1,089	4,921	10,849
Acquisition	136	300	136	300
Tracker	156	344	156	344
Ranger	36	79	36	79
Receiver				
Beam Expander	1,478	3,258	5,084	11,208
Optical Train	801	1,766	13,974	30,807
Gimbals and CMGs	517	1,140	1,780	3,924
Tracker	77	170	77	170
Astrionics	343	756	1,065	2,348
Electrical Power	101	223	110	243
Propulsion Inert	351	774	2,505	5,523
Propellant	2,127	4,689	16,764	36,958
Stabilization and Attitude Control	159	351	1,079	2,379
Structure	453	999	3,569	7,868
Total	8,465	18,662	68,687	151,427

3.4 TASK IV: CONCEPT EVALUATION AND COST, SPACE-BASED LASER

Cost analysis was performed to develop basic cost data for the space-based laser rocket system candidates and compare them to similarly performing chemical Orbital Transfer Vehicle (OTVs).

The following four concepts were costed:

(1) Basic laser rocket system – 16-MW, 2268-kg (5000-lbm) round trip to geosynchronous orbit
(2) SPS-type laser rocket system – 490-MW, 148,000-kg (326,000-lbm) one way to geosynchronous orbit
(3) Small two-stage chemical OTV – 2268-kg (5000-lbm) round trip to geosynchronous orbit
(4) Large two-stage chemical OTV – 148,000-kg (326,000-lbm) one way to geosynchronous orbit

The 16-MW laser rocket system and the small OTV were assumed to represent the shuttle era systems supported by shuttle as the low earth orbit (LEO) transportation system. The 490-MW laser rocket system and the large OTV represent later systems which require the support of heavy lift launch vehicles for transportation to LEO.

The space-based laser rocket system consists of the following elements:

- Single laser transmitter unit
- Single energy relay unit in geosynchronous orbit
- Multiple propulsion units (fleet size as required by the mission model)

The OTV system consists of a fleet of common two-stage chemical OTVs sized as indicated above.

The system life-cycle cost was defined to include DDT&E costs, investment costs, and 10-year operating costs for a given concept and a given mission model.

3.4.1 Groundrules and Assumptions

The following groundrules and assumptions were applied in this cost analysis:

- All costs in FY '77 dollars
- Shuttle fee at \$13.5 million (\$208/lb to LEO)
- Shuttle load factor at 90% (58,500 lb to LEO)
- Existing HLLV fee at \$6.3 million (\$14/lb to LEO)
- HLLV load factor at 95% (427,500 lb to LEO)
- LH_2 cost in the shuttle era of 40¢/lb and in the HLLV era at 30¢/lb
- LO_2 cost in the shuttle era at 3¢/lb and in the HLLV era at 2.2¢/lb
- On-orbit assembly, refurbishment, and refueling techniques assumed to be developed. Unmanned operations assumed with the exception of the shuttle crew

- Unmanned on-orbit refurbishment assumed at 19% of the laser propulsion unit and OTV cost. (This factor was derived from the Payload Effects Studies performed for NASA by LMSC)
- The laser transmitter unit and the energy relay unit are assumed to require a single on-orbit refurbishment each at 19% of their unit cost during the 10-year life of the system
- 90% leaving curve was applied to propulsion units and OTV's

3.4.2 Laser Rocket System Costs

The laser rocket system costs were estimated by means of LMSC-developed CERs (Cost Estimating Relationships) for the major subsystems and elements of the system. These CERs are based on actual as well as study cost data collected and generated during the past several years of laser systems cost analysis and the basic historical spacecraft costs.

The solar power supply costs were extracted from the NASA SPS (Solar Power Systems) studies; $500/kW was used for the earlier 16-MW system and $300/kW for the later 490-MW system.

New engine development was assumed and its costs charged to the propulsion unit. The same engine is used by the relay unit. New spacecraft development was assumed for each of the three laser rocket system elements. The use of an existing spacecraft bus was not investigated. Complete development costs for an EXCIMER-type closed cycle laser are charged to the laser transmitter.

In general, the laser rocket system development costs include one engineering model and one prototype unit which becomes the first operational unit after on-orbit deployment. In case of the laser propulsion units, the prototype is an on-orbit spare and additional units are bought for the fleet as required by a given mission model and assumed reuse frequency. In case of common equipment used by more than one element of the laser rocket system (i.e., ranger, tracker), the development of an engineering model is charged only once, but each system element is charged with the costs of the prototype equipment.

Technology development costs were estimated but are not included in the laser rocket system costs, since it is felt that these funds should not be charged to any given system.

The laser rocket 16- and 490-MW DDT&E and initial system deployment costs (except the propulsion unit fleet) are presented in Tables XVIII and XIX, respectively. The 16-MW system initial costs are estimated at a little under $2 billion and the 490-MW laser rocket system is slightly over $5 billion.

The laser propulsion unit fleet investment costs, including on-orbit deployment costs, were estimated as a function of the fleet size as shown in Figure 17 for both the 16- and 490-MW systems.

TABLE XVIII. 16-MW SPACE SYSTEM DDT&E AND INITIAL DEPLOYMENT COSTS ($M)

	Tech.	Space Laser Transmitter	Propulsion Unit		Relay		Total[a]
			DDT&E	1st Unit	Transmitter	Receiver	
Acquisition	$ 1.3	$ 14.037			$ 5.904		
Tracker		19.933	$ 7.034	$ 1.657	5.873	$ 2.485	
Ranger		4.090			1.079		
Beam Control	2.0	26.672			10.284		
Gimbals & CMGs		8.425	1.799	0.671	2.733	2.852	
Fire Control		14.856					
Beam Expander	26.3	163.629	22.559	4.584	15.588	52.424	
Adaptive Mech.	4.3	44.643	6.344	2.072	4.491	11.900	
Optical Train	3.8	12.353	6.016	1.004	8.485	3.448	
Subtotal	$37.7	$ 308.638	$ 43.752	$ 9.988	$ 54.437	$ 73.109	
Spacecraft		$ 68.139	$ 39.962	$13.923	$ 49.263		
Propulsion			101.269	1.699	3.747		
Laser	8.6	146.296					
Power Supply		110.356					
Subtotal	$46.3	$ 633.429	$184.983	$25.610	$180.556		$ 998.968
Facilities		22.500					22.500
Syst. Eng. & Intg.		95.014	27.747	4.098	27.083		149.844
Syst. Test		25.337	7.399	1.152	7.222		39.958
Syst. GSE		12.669	1.675		3.536		17.880
Launch OPS		49.471	4.264		9.974		63.709
Flight OPS		41.037	3.940		11.050		56.027
C3 Mods		25.000					25.000
Shuttle Fee[b]		341.600	6.750		13.500		361.35[illegible]
Data		35.178	9.200	0.926	9.577		53.955
Prog. Mgm't.		45.732	11.960	1.589	12.450		70.142
Total	$46.3	$1,326.967	$257.918	$33.375	$274.948		$1,859.833

(a) Total excludes technology and propulsion unit fleet deployment, but includes first prototype on-orbit deployment.
(b) Based on shuttle net payload to LEO of 58,500 lb (65,000 × 0.9 load factor) and $13.5 M/flight fee in 1976 dollars (65,000 × $208/lb).

TABLE XIX. 490-MW SPACE SYSTEM DDT&E AND INITIAL DEPLOYMENT COSTS ($M)

	Tech.	Space Laser Transmitter	Propulsion Unit		Relay		Total[a]
			DDT&E	1st Unit	Transmitter	Receiver	
Acquisition	$ 1.3	$ 14.037			$ 5.904		
Tracker		19.933	$ 7.034	$ 1.657	5.873	$ 2.485	
Ranger		4.090			1.079		
Beam Control	2.0	26.672			10.284		
Gimbals & CMGs		12.915	4.197	0.823	7.416	5.595	
Fire Control		14.856					
Beam Expander	26.3	171.105	30.014	5.464	23.029	59.878	
Adaptive Mech.	4.3	44.643	6.715	2.206	4.491	11.900	
Optical Train	3.8	76.621	56.952	15.268	42.755	30.078	
Subtotal	$37.7	$ 384.872	$104.912	$25.438	$210.767		
Spacecraft		91.378	68.924	19.280	60.452		
Propulsion			200.442	6.366	10.264		
Laser	8.6	714.927					
Power Supply		1,261.229					
Subsystems	$46.3	$2,452.406	$374.278	$51.084	$281.483		$3,108.167
Facilities		22.500					22.500
Syst. Eng. & Intg.		367.861	56.142	7.663	42.222		466.225
Syst. Test		98.096	14.971	2.043	11.259		124.326
Syst. GSE		49.048	3.477		5.424		57.949
Launch OPS		253.609	7.624		15.647		276.880
Flight OPS		272.977	7.975		19.082		300.034
C3 Mods		25.000					25.000
HLLV Fee[b]		398.582	3.724		2.232		404.538
Data		140.660	18.579	1.824	15.005		174.244
Prog. Mgm't.		182.858	24.152	3.131	19.506		226.516
Total	$46.3	$4,263.597	$510.922	$65.745	$411.860		$5,186.379

(a) Total excludes technology and propulsion unit fleet deployment, but includes first prototype on-orbit deployment.

(b) Based on use of existing HLLV with net payload to LEO of 427,500 lb (450,000 × 0.95 load factor) and $6.3 M/flight fee in 1976 dollars (450,000 × $14/lb).

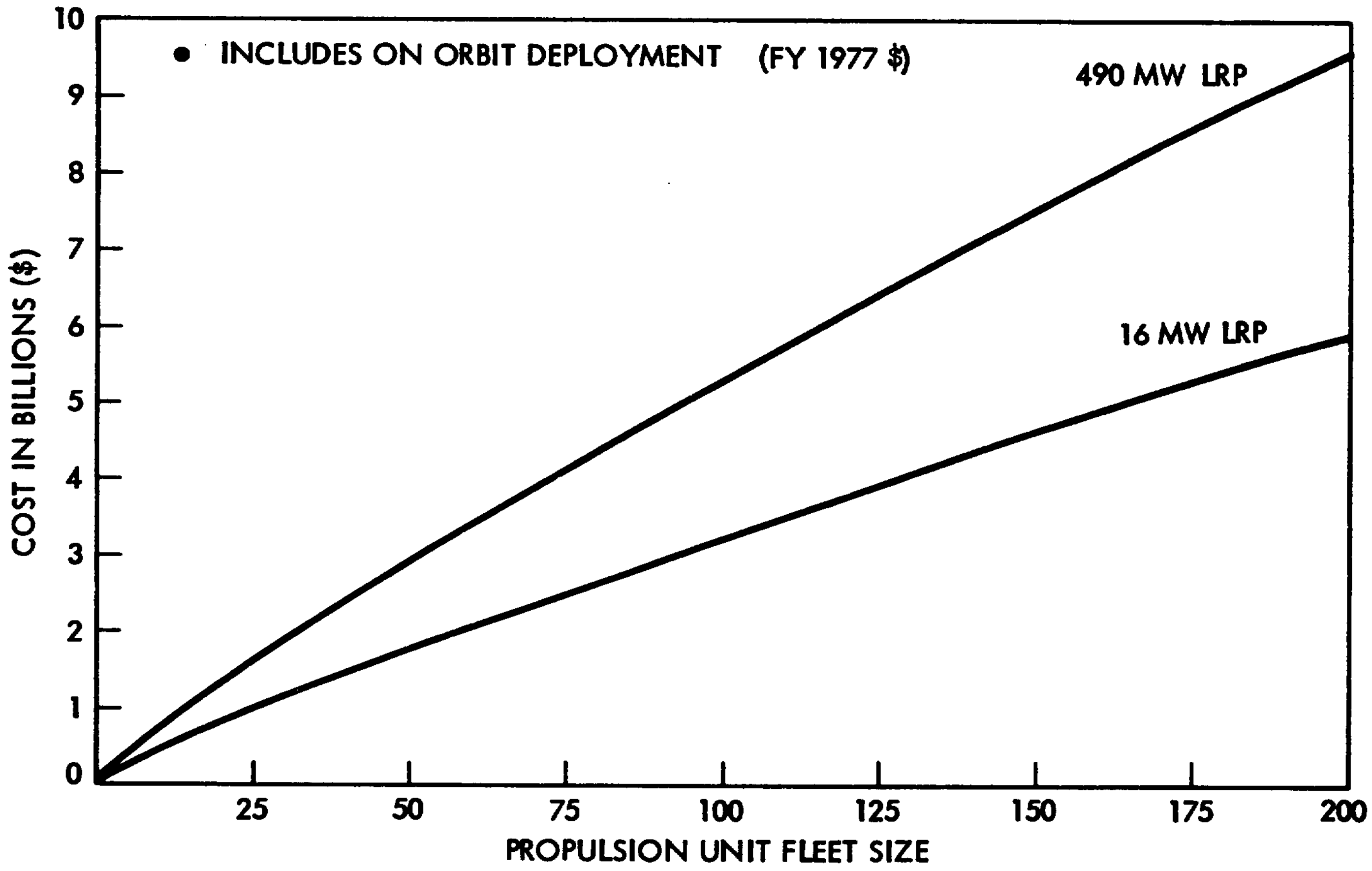

Figure 17. LRP system propulsion unit fleet investment costs

The operations costs over the 10-year system life are dependent on the fleet size, the frequency of propulsion unit refurbishment, and the number of missions performed for which fuel has to be supplied. The fuel resupply costs for the propulsion unit were costed on per mission basis as follows:

16-MW System	490-MW System
$3.537 M/Mission	$4.138 M/Mission

Simultaneous resupply of over four 16-MW and almost two 490-MW propulsion units was assumed with a single fuel resupply flight to LEO by the shuttle or the HLLV, respectively.

The fleet size dependent operations costs are plotted for the two LRP systems as shown in Figure 18. The basic 10-year system operations cost can be read off the left-hand scale for a given fleet size. The refurbishment cost for a single propulsion unit refurbishment is also plotted in Figure 18 and can be read off the right-hand scale.

The refurbishment cost of the laser transmitter and the relay unit is a constant operating cost regardless of the fleet size. It was assumed that after 5-year operating period, both of these system elements will be refurbished once in the LEO orbit. The respective refurbishment costs are as follows:

	16-MW System	490-MW System
	$139.282 M	$517.589 M
	26.748	41.347
ERU Refurbishment Total	$166.030 M	$558.886 M

These constitute the basic cost data blocks developed by LMSC for the laser rocket system and submitted to ECON, Inc., for their breakeven and further mission model analyses.

3.4.3 LO_2/LH_2 OTV System Costs

The LO_2/LH_2 OTV costs were developed in similar cost data block fashion. The basic OTV DDT&E and investiment costs were extracted from the Boeing "Future Space Transportation Systems Analysis" study report No. D180-20242-3. These costs were modified to assure that comparable cost elements are included and escalated to current dollars as follows:

	Small OTV	Large OTV
DDT&E Cost:		
New Engine 1st Stage	$104 M	$180 M (common)
New Engine 2nd Stage	87	
1st Stage	213	650
2nd Stage	69	162
Total	$473 M	$992 M

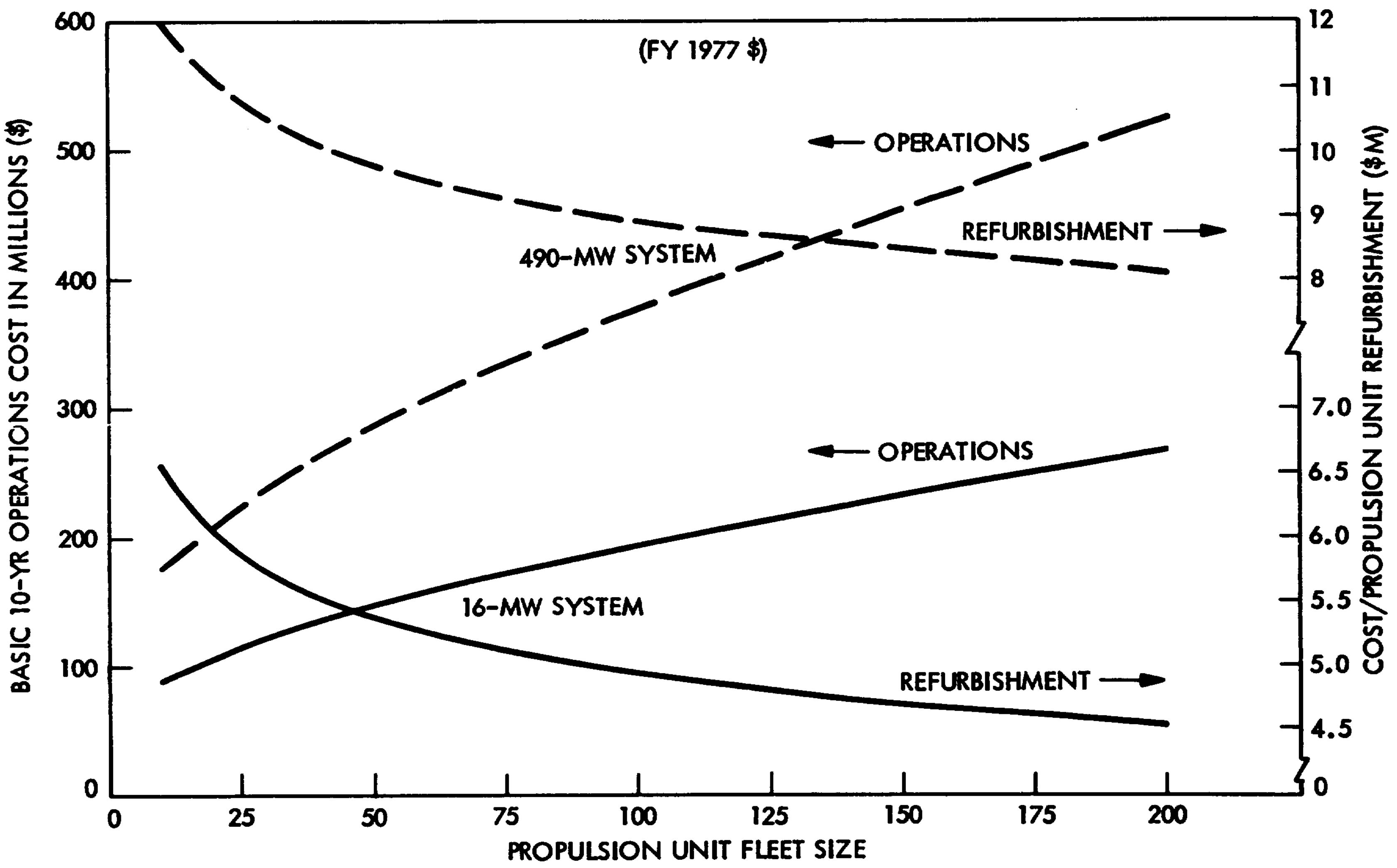

Figure 18. LRP system operations costs

First Unit Cost:		
1st Stage	$ 21.0 M	$ 44.5 M
2nd Stage	19.5	35.5
Total	$ 40.5 M	$ 80.0 M

The chemical OTV fleet investiment and deployment costs are shown plotted in Figure 19 as a function of fleet size and the OTV configuration. The OTV system operations costs are displayed in Figure 20 in a similar manner to the laser rocket system operations costs shown previously in Figure 18.

The costs of OTV fuel resupply to LEO (by far the most significant cost contributor) were estimated including losses as follows:

	Small OTV		Large OTV	
Fuel Cost	$ 0.009 M		$ 0.123 M	
Transportation to LEO	25.127	(1.9 shuttle flights)	37.800	(6 HLLV Flts.)
Launch Operations	2.280		7.200	
Cost/Mission	$27.416 M		$45.123 M	

Weight of Fuel/Mission Excluding Losses:				
LH_2	6,495 kg	(14,314 lbm)	131,686 kg	(290,314 lbm)
LO_2	32,473 kg	(71,570 lbm)	658,427 kg	(1,451,569 lbm)
Total 2 Stages	38,968 kg	(85,884 lbm)	790,113 kg	(1,741,883 lbm)

The above chemical OTV cost data were derived by LMSC based on basic Boeing costs and turned over to ECON, Inc., for mission model cost sensitivity and further economic analyses.

3.4.4 Sample Case Cost Comparison

An arbitrary sample case of a mission model was generated to illustrate the aggregation of system life cycle costs and a cost comparison of the laser rocket system to the OTV chemical system. In addition to the basic assumptions listed in section 3.4.1, the following assumptions were made with respect to the propulsion units and chemical OTV's for this sample case:

	Laser Propulsion Unit	Chemical OTV
Nominal Reuse Frequency	60 Reuses	30 Reuses
Nominal Refurbishment Frequency	After 20th Reuse	After 10th Reuse
No. of Missions Prior to Expending Unit	10	10
Refurbishment Required Prior to Expendable Mission	Yes	Yes

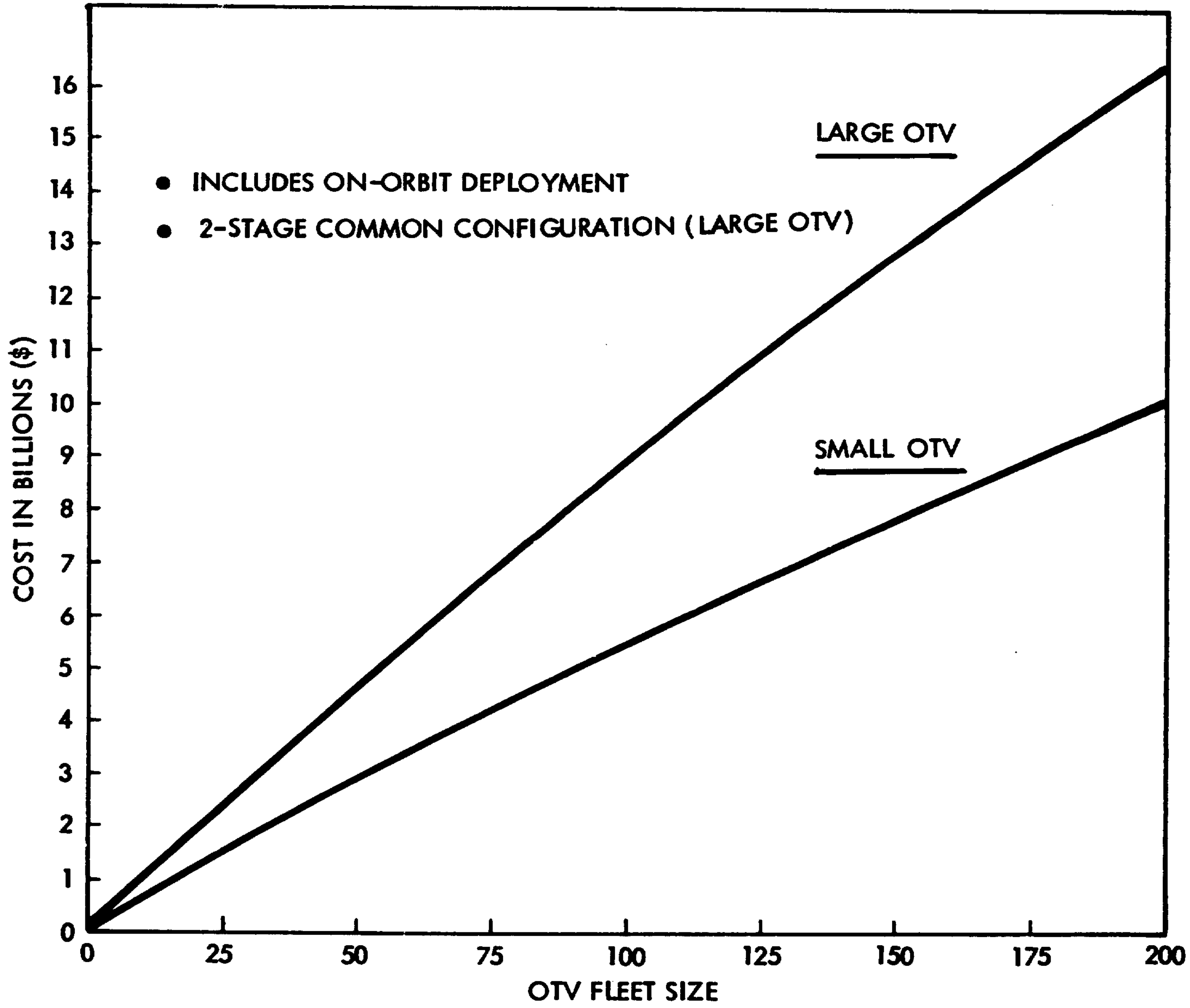

Figure 19. LO_2/LH_2 OTV fleet investment costs

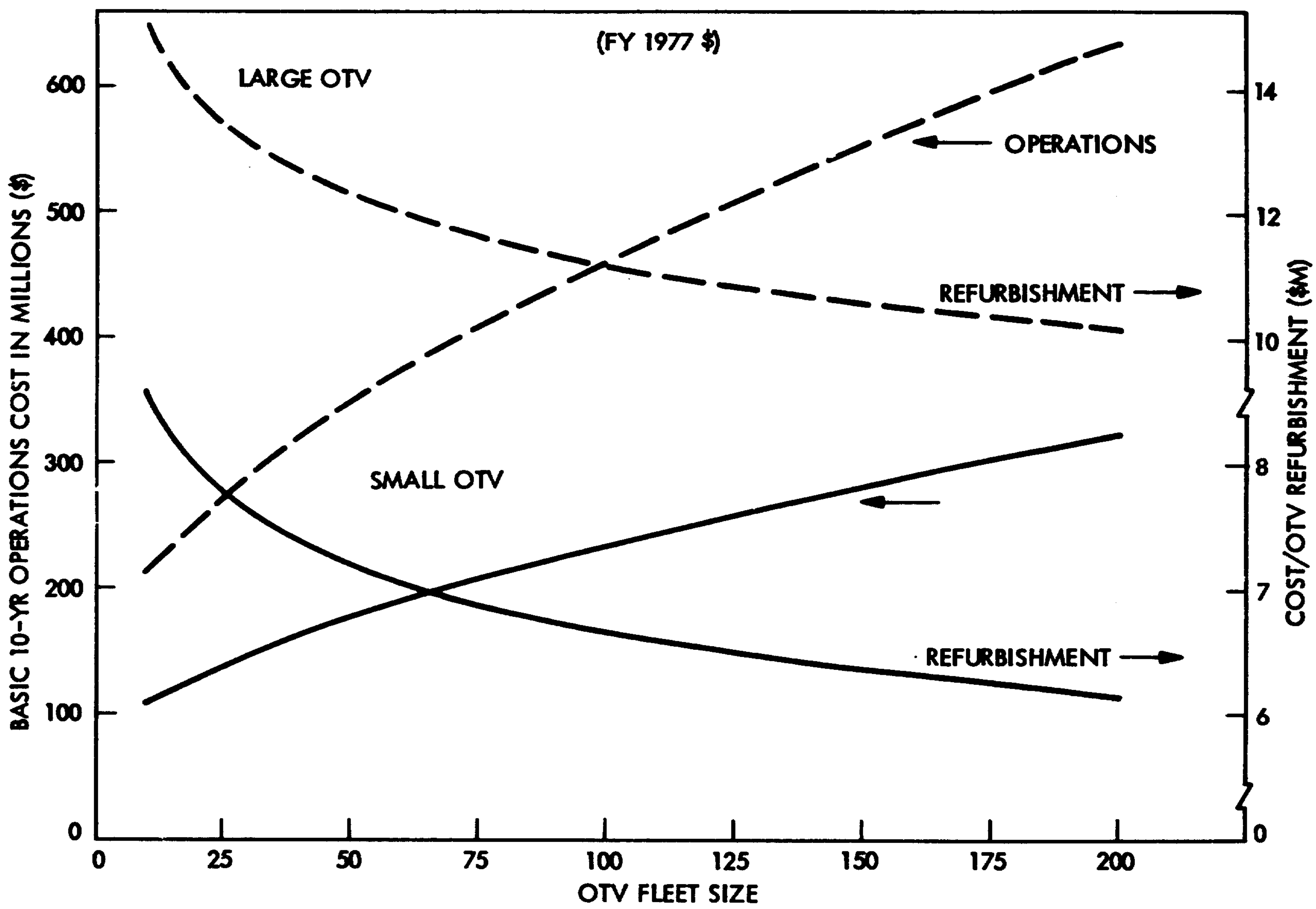

Figure 20. LO_2/LH_2 OTV system operations cost

These assumptions were made after reviewing the inherent operational modes of the LRP unit as opposed to the chemical OTV. The OTV engine is subject to hard starts. It was also felt, that in case of an expendable mission, a refurbishment would be required prior to a long flight, such as in the case of the interplanetary missions.

Fuel loss allowances of 4 to 14% have been accounted for in the life cycle costs. The percent of allowance varies with the amount of fuel transferred to an OTV. The higher allowance is for a smaller fuel capacity OTV.

The sample case mission model consists of 10 expendable and 1300 reusable missions in a 10-year time span. This results in a requirement of 30 LRP units and 50 chemical OTVs to accomplish the mission model.

Table XX shows the costs for the above case. For the 16-MW LRP and the small chemical two-stage OTV, the 10-year cost is $8.0 and $38.7 billion, respectively. For this mission model, costs were per mission by a factor of almost 5 over the Laser system.

TABLE XX. 1310 MISSION SAMPLE CASE COST COMPARISON

(FY '77 $M)

	Small		Large	
	Laser PU (16 MW)	Chemical OTV	Laser PU (490 MW)	Chemical OTV
No. of Vehicles	30	50	30	50
No. of Refurbs.	50	90	50	90
DDT&E and Initial Deployment	$1,859.883	$ 473.0	$ 5,186.379	$ 992.0
Fleet Investment	1,095.77	2,828.18	1,860.200	4,736.36
Subtotal	$2,955.65	$ 3,301.18	$ 7,046.58	$ 5,728.36
Basic 10 yr OPS	$ 123.58	$ 176.76	$ 239.00	$ 349.57
Relay and Transmitter Refurbishment	166.03		558.89	
PU Refurbishment	287.11	647.55	521.00	1,102.59
Fuel Resupply	4,456.62	34,544.16	5,213.83	56,854.98
Total LCC	$7,988.99	$38,669.65	$13,579.35	$64,035.50
Cost Ratio = (Chemical LCC/Laser LCC)		4.84		4.72
Cost/Mission:	$ 6.10 M	$ 29.52 M	$ 10.37 M	$ 48.88 M
Cost/Lb:	$1,220	$ 5,924	$ 31.81	$ 149.94
Payload Wt/Mission	2268 kg (5000 lb) Roundtrip		148,000 kg (326,000 lb) One Way	

In terms of \$/lb of payload delivered from LEO to GEO and back to LEO, the LRP system costs ≈ \$1200/lb and the chemical OTV system ≈ \$5900/lb.

Comparing the larger systems, the 490-MW LRP and the large chemical OTV, the 10-year life cycle costs for the sample case are \$13.6 and \$64.0 million, respectively. Their respective costs per mission, as shown in Table XX, are \$10.4 and \$48.9 million or again a factor of almost 5 higher in the case of the OTV. Corresponding \$/lb, one-way to GEO, are \$32/lb for the LRP and \$150/lb for the OTV.

As can be seen from Table XX, the fuel resupply represents 40% to 60% of the LRP system cost, and in the chemical OTV case it is almost 90% of the system life cycle costs for a case of 1310 missions.

3.4.5 Cost Analysis-Space-Based System (ECON Inc.)

With the selection of the LO_2/LH_2 system as the most cost-effective alternative system, the fleet sizing was determined for both the baseline case (cryogenic system) and the laser rocket propulsion case for each variation of the mission model. (Variations, or scenarios, were created by exercising the activity level multipliers.) The most recent state-of-the-art assumptions were utilized to determine the number of reuses and the refurbishment needs as indicated in Table XXI. With the fleet size and the number of flights in the mission model obtained, the total life-cycle costs were calculated.

TABLE XXI. REUSE AND REFURBISHMENT CAPABILITIES OF LO_2/LH_2 AND LASER PROPELLED OTVs

Attribute	~ 5,000-lb Payload Capability		~ 326,000-lb Payload Capability	
	Laser	Cryogenic	Laser	Cryogenic
Number of Reuses	40	30	60	30
Number of Missions Before Refurbishment	20	10	20	10
Number of Reuse Missions Prior to Use on an Expendable Mission (Refurbishment Prior to Expending)	10	10	10	10

All analyses in this study use FY '77 dollars for cost calculations. In many instances, the life-cycle costs are discounted back to the first year in which costs are incurred, but it should be kept in mind that cost figures were not inflated to reflect 1980 or 1995 dollars. Spread functions were used to distribute DDT&E and investment costs in a manner that distributed 40% of the costs over the first 50% of the time period. Life-cycle cost calculations were performed using 10 years of flight operations in addition to the time prior to flight operations needed for deployment (in the case of the laser powered system) and DDT&E.

Cost expenditures are grouped in the following categories:

- DDT&E (Design, Development, Test and Engineering). This includes all research, design, and testing efforts. In the case of the laser system, this category also includes the cost of the laser unit, and the necessary relay units required for initial system deployment.
- Investment and Spares. This category covers the purchase costs of the cryogenic orbit transfer vehicles or the laser propulsion units. The cost per unit is a function of the total number of units manufactured. This category also includes the spare parts for the OTVs.
- Laser System Deployment. This cost occurs only in the laser rocket system case and covers the cost of placing the laser and/or relay units into suitable orbit prior to the initiation of operations.
- OTV Deployment and Operations. This category covers all deployment and operations costs of the OTVs except for refurbishment and fuel resupply. It includes ground operations, flight operations, and miscellaneous support costs such as data management and training.
- Refurbishment. This category covers refurbishment costs for the OTVs. This refurbishment is assumed to take place in low earth orbit.
- Fuel Resupply. This category covers the costs of the propellant and the costs to transport the propellant to low-earth orbit.

The costs for each category are distributed over the appropriate time span as illustrated in Table XXII. Once the fleet size and number of missions are determined and the total per-year costs by case are determined, discounting is performed.

The purpose of discounting is to convert life-cycle costs of alternative projects into common dollar terms. Figure 21 indicates how discounting allows comparison of expenditures varying in magnitude and time period. Discounting the cost streams of alternative projects provides an objective criterion for comparing costs when one project has large expenditures later in the project time frame. In the cases shown in this report, the cost streams were discounted back to the first year in which costs were incurred by either propulsion system; both alternatives are discounted to that same year.

Four cases have been selected from the twelve which were computed. These four cases represent reasonable possible future missions. The life-cycle cost comparisons for Cases 3, 6, 8, and 11 are given in Tables XXIII through XXVI. They will be discussed in the same order.

Case 3 represents a mission model composed of all payloads in the 2268 kg or 5000-lb payload range. There are a total of 450 missions where the vehicle can be reused, and

TABLE XXII. YEARLY COST DISTRIBUTION
CASE 8
(IN MILLIONS OF FY '77 DOLLARS)

LASER PROPULSION SYSTEM

	1987	1988	1989	1990	1991	1992	1993	1994	1995
RDT&E	116.506	610.532	1123.037	1317.105	1066.900	402.104			
INVESTMENT & SPARES								28.976	172.093
LASER SYSTEM DEPLOYMENT						327.166	327.166	327.166	
OTV DEPLOYMENT & OPERATIONS							49.315	49.305	197.020
REFURBS									226.649
FUEL RESUPPLY									1660.653
YEARLY TOTAL	116.506	610.532	1123.037	1317.105	1066.900	729.271	376.482	405.458	2210.933

	1996	1997	1998	1999	2000	2001	2002	2003	2004
RDT&E									
INVESTMENT & SPARES	384.577	597.044	758.822	830.963	886.792	672.010	644.580	154.025	
LASER SYSTEM DEPLOYMENT									
OTV DEPLOYMENT & OPERATIONS	197.020	197.020	197.020	197.020	197.020	197.020	197.020	197.020	197.020
REFURBS	226.649	226.649	226.649	226.649	226.649	226.649	226.649	226.649	226.649
FUEL RESUPPLY	1660.653	1660.653	1660.653	1660.653	1660.653	1660.653	1660.653	1660.653	1660.653
YEARLY TOTAL	2422.017	2636.204	2900.462	3174.193	2951.532	2710.250	2482.978	2193.066	2038.240

TOTAL CURRENT YEAR COSTS = $31,[illegible].10 MILLION

CRYOGENIC SYSTEM

	1987	1988	1989	1990	1991	1992	1993	1994	1995
RDT&E	-0-	-0-	27.985	144.034	264.942	310.705	249.112	94.063	
INVESTMENT & SPARES								52.720	314.247
OTV DEPLOYMENT & OPERATIONS							75.305	75.305	702.035
REFURBS									331.702
FUEL RESUPPLY									10220.745
YEARLY TOTALS	-0-	-0-	27.985	144.034	264.942	300.705	324.417	222.085	19640.880

	1996	1997	1998	1999	2000	2001	2002	2003	2004
RDT&E									
INVESTMENT & SPARES	699.801	1086.435	1380.820	1519.354	1468.111	1222.809	880.012	281.734	
OTV DEPLOYMENT & OPERATIONS	702.035	702.035	702.035	702.035	702.035	702.035	702.035	702.035	702.036
REFURBS	331.702	331.702	331.702	331.702	331.702	331.702	331.702	331.702	331.707
FUEL RESUPPLY	10220.745	10220.745	10220.745	10220.745	10220.745	10220.745	10220.745	10220.745	10220.745
YEARLY TOTALS	20834.373	20420.987	20715.382	20853.946	20802.673	20567.411	20043.574	19616.296	19330.962

TOTAL CURRENT YEAR COSTS = $3[illegible],482.33 MILLION

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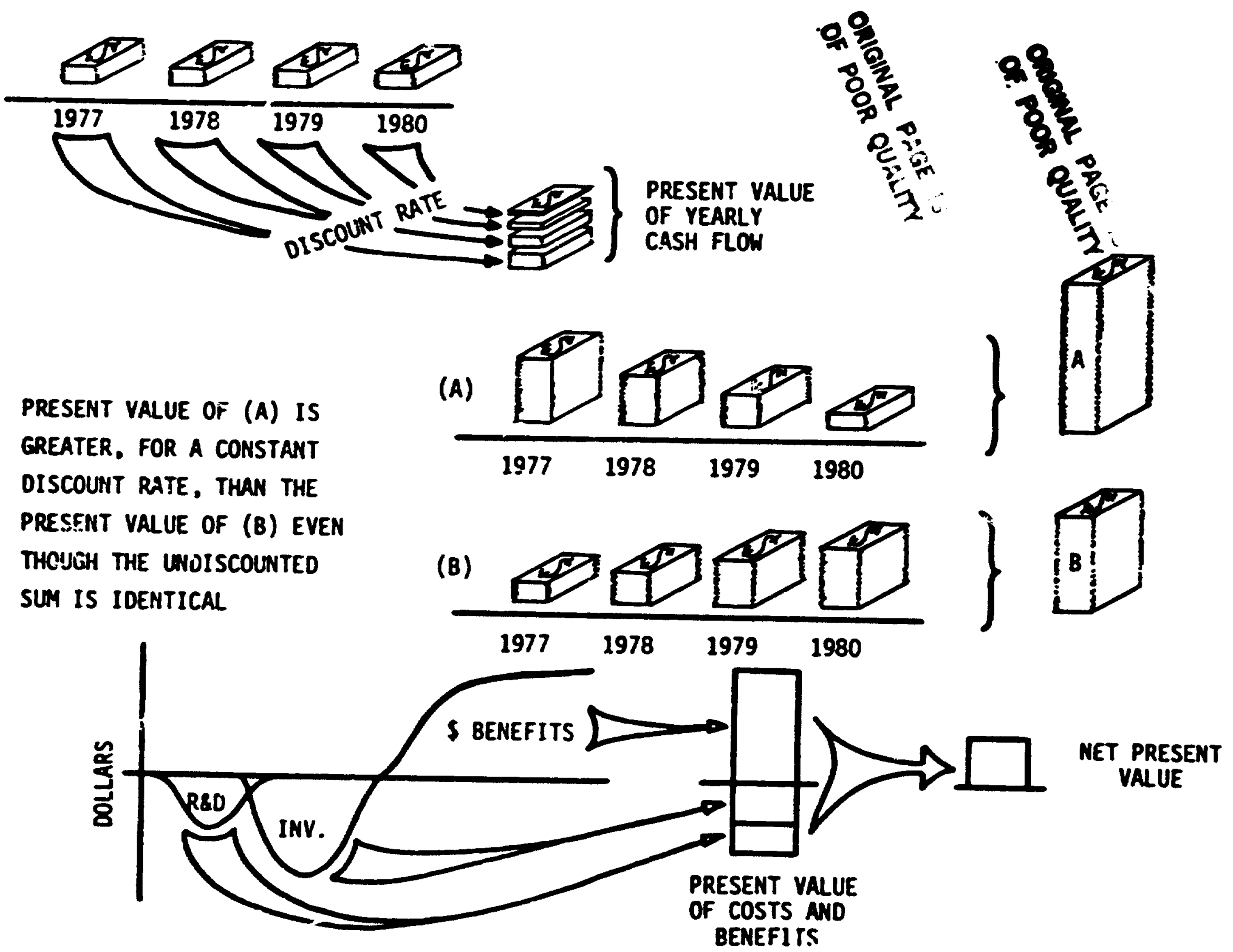


Figure 21. Money Now Versus Money Later

TABLE XXIII. LCC COST COMPARISON CASE 3 (SPACE)

Mission Composition: 450 5,000 lb P/Ls
10 5,000 lb Expendable P/Ls

Number of OTV's = Laser 16, Cryo 22

LCC Costs (In Millions of Dollars)

Cryogenic System	Category	Space-Based Laser Rocket System
473.00	DDT&E	1,377.80
694.43	Investment and Spares	442.99
0.0	Laser System Deployment	482.00
759.16	OTV Deployment and OPS	295.77
254.21	Refurbs.	302.54
12,611.36	Fuel Resupply	1,627.02
14,792.16	Total Consant Year LCC Costs (FY '77 $M)	4,528.12
5,821.70	Total LCC Costs Discounted to (1984)	2,456.40
	Discounted Cost Ratio (Chem/Laser)	2.37
	Discounted Cost Ratio w/o DDT&E	4.08

TABLE XXIV. LCC COST COMPARISON CASE 6 (SPACE)

Mission Composition: 4,500 326,000 lb P/Ls
14 326,000 lb Expendable P/Ls

Number of OTV's = Laser 87, Cryo 160

LCC Costs (In Millions of Dollars)

Cryogenic System	Category	Space-Based Laser Rocket System
992.0	DDT&E	4,204.90
7,787.86	Investment and Spares	3,736.53
0.0	Laser System Deployment	981.50
6,544.79	OTV Deployment and OPS	1,304.22
3,491.46	Refurbs.	2,015.38
203,685.22	Fuel Resupply	18,678.93
222,501.33	Total Constant Year LCC (FY '77 $M)	30,921.46
70,568.306	Total LCC Discounted to (1987)	11,954.83
	Discounted Cost Ratio	5.90
	Discounted Cost Ratio w/o DDT&E	8.02

TABLE XXV. LCC COST COMPARISON CASE 8 (SPACE)

Mission Composition:	4000	326,000 lb P/Ls		
	400	5,000 lb P/Ls		
	85	5,000 lb Expendable P/Ls		
Number of OTVs =	Laser		Cryo	
	67	Large	133	
	85	Small	85	

LCC Costs (In Millions of Dollars)

Cryogenic System	Category	Space-Based Laser Rocket System
1,091.2	DDT&E	4,625.39
8,835.1	Investment and Spares	4,855.28
0	Laser System Deployment	981.50
7,970.96	OTV Deployment and OPS	2,076.91
3,317.82	Refurbs.	2,266.49
182,207.45	Fuel Resupply	16,696.53
203,422.53	Total Constant Year LCC Costs (FY '77 $M)	31,502.10
64,599.43	Total LCC Costs Discounted to (1987)	12,347.29
	Discounted Cost Ratio	5.23
	Discounted Cost Ratio w/o DDT&E	7.27

TABLE XXVI. LCC COST COMPARISON CASE 11 (SPACE)

Mission Composition:	8000	326,000 lb P/Ls		
	425	5,000 lb P/Ls		
	14	5,000 lb Expendable P/Ls		
Number of OTVs =	Laser		Cryo	
	135	Large	270	
	19	Small	24	

LCC Costs (In Millions of Dollars)

Cryogenic System	Category	Space-Based Laser Rocket System
1,091.20	DDT&E	4,625.39
12,758.74	Investiment and Spares	5,510.44
0	Laser System Deployment	981.50
2,075.31	OTV Deployment and OPS	2,098.50
5,330.43	Refurbs.	3,013.09
362,059.55	Fuel Resupply	33,234.82
393,315.23	Total Constant Year LCC Costs (FY '77 $M)	49,463.74
124,523.49	Total LCC Costs Discounted to (1987)	18,016.71
	Discounted Cost Ratio	6.91
	Discounted Cost Ratio w/o DDT&E	8.57

10 outer planet missions which call for an expendable vehicle. The fleet was sized at 22 units for the cryogenic version and 16 for the laser rocket system. For this case, the smaller (16-MW) space-based laser would be utilized. Shuttle was used for fuel transportation to low-earth orbit. It was assumed that the initial operating capability (IOC) date for this system would be 1990 and DDT&E would begin in 1984. All costs are discounted back to 1984. The discounted costs are: $5821.7 M for the cryogenic system and $2456.4 M for the laser rocket system.

Case 6 is composed of missions with large payloads. This case may not represent a realistic mission model, but it is worthwhile to investigate the economics of a large-payload mission model. The large payloads (148,000 kg or 326,000 lbm) are sized to represent the solar power satellite segments, and due to the large mass that would be transported to space, 4,500 reusable missions are called for in addition to 14 expendable missions. The fleet size for this case is 160 vehicles for the cryogenic system and 87 for the laser system. This mission model would require the 490-MW space-based laser with an IOC date of 1995. A heavy-lift launch vehicle is assumed for full transportation to low-earth orbit. The discounted costs are: 70568.3 million dollars for the cryogenic case and 11954.8 million dollars for the laser system discounted back to 1987.

Cases 8 and 11 are mixed cases, that is, they encompass both large and small payloads. The IOC date is taken to be 1995 and the DDT&E costs are 10% larger than for strictly large-payload DDT&E to cover the additional costs of developing the two vehicle sizes (i.e., different mirror cooling systems, different tank sizes, etc.). The mixed case also assumes the full transportation to low-earth orbit is by the means of the heavy-lift launch vehicle.

Case 8 is a mission model calling for 4000 large payloads and 400 small payloads during the 10 years of operations in addition to 85 expendable missions. The discounted (to 1987) costs are: $64599.4 M for the cryogenic system and $12347.3 M for the laser rocket system.

Case 11 is a mission model calling for 8000 large payloads, 425 small ones and 14 small expendable payloads. The discounted costs are: $124523.5 M for the cryogenic system and 18016.7 M for the laser rocket system.

3.5 TASK V: PARAMETRIC ANALYSIS, GROUND-BASED LASER

The purpose of this parametric analysis is to define concepts of laser rocket systems with the ground-based laser transmitter. The ground-based concepts utilize the same propulsion units established in the space-based laser rocket system parametric analysis, section 3.2.

This analysis considered orbital propulsion missions using energy from a ground-based laser transmitter to heat a working fluid in the PU and provide the necessary thrust to accomplish the mission established in Task I. The ground-based laser rocket system resulted in four separate units that must interface and interact with one another as one coordinated system. The four units are:

(1) Ground Laser Transmitter Unit (GLTU)
(2) Propulsion Unit (PU)
(3) Geosynchronous Energy Relay Unit (GERU)
(4) Medium Earth Orbit Energy Relay Unit (MERU)

Figure 22 illustrates the relay system conceived for the ground-based laser rocket system in which the laser transmitter beams the energy to a relay satellite in medium earth orbit which in turn relays the beam to the LEO propulsion unit or the geosynchronous relay for further relay to the PU at GEO.

Because the GLTU is based on earth, MERUs are required to relay energy to the PU for the LEO maneuvers. The two system concepts derived in this analysis are equal in performance to the systems derived in the space-based analysis; that is, one system is conceptually designed around the projected geosynchronous mission with a 2268-kg (5000-lbm) payload round trip. The other system is conceived to perform the space power satellite (SPS) missions to GEO with a 148,000-kg (326,300-lbm) payload one way. Both systems have excess capability for the less demanding missions of the mission model. The SPS payload weight was selected because of the parametric bounds of 1000-MW laser power established in the statement of work. The SPS missions are transportation of SPS segments to GEO and do not include other support missions such as work crew quarters and supply.

The PU operation for the ground-based laser rocket system is identical to the space-based systems, i.e., the LEO maneuvers to raise apogee are performed near perigee and may require two or more propulsive burns in succeeding orbits. The GEO maneuvers (plane change and circularization) are performed with a single burn to avoid the long phasing periods that can occur with orbit periods near 24 hr each.

Of primary concern in this analysis of ground-based laser rocket systems is the atmospheric propagation of the laser beam, laser transmitter site location, and orbital parameters of the MERUs. All of these interact and effect the total system.

3.5.1 Atmospheric Propagation

In addition to diffraction effects, propagation of a laser beam through the atmosphere has two basic effects to the laser beam width and flux density. They are:

(1) Extinction – loss of energy by absorption and scattering
(2) Beam divergence – widening of the beam caused by atmospheric turbulence and possible thermal blooming

- <u>Extinction</u>

Extinction, a physical phenomenon which may be lessened but not avoided, is a function of the laser wavelength, beam zenith angle, and transmitter altitude; it is linear with

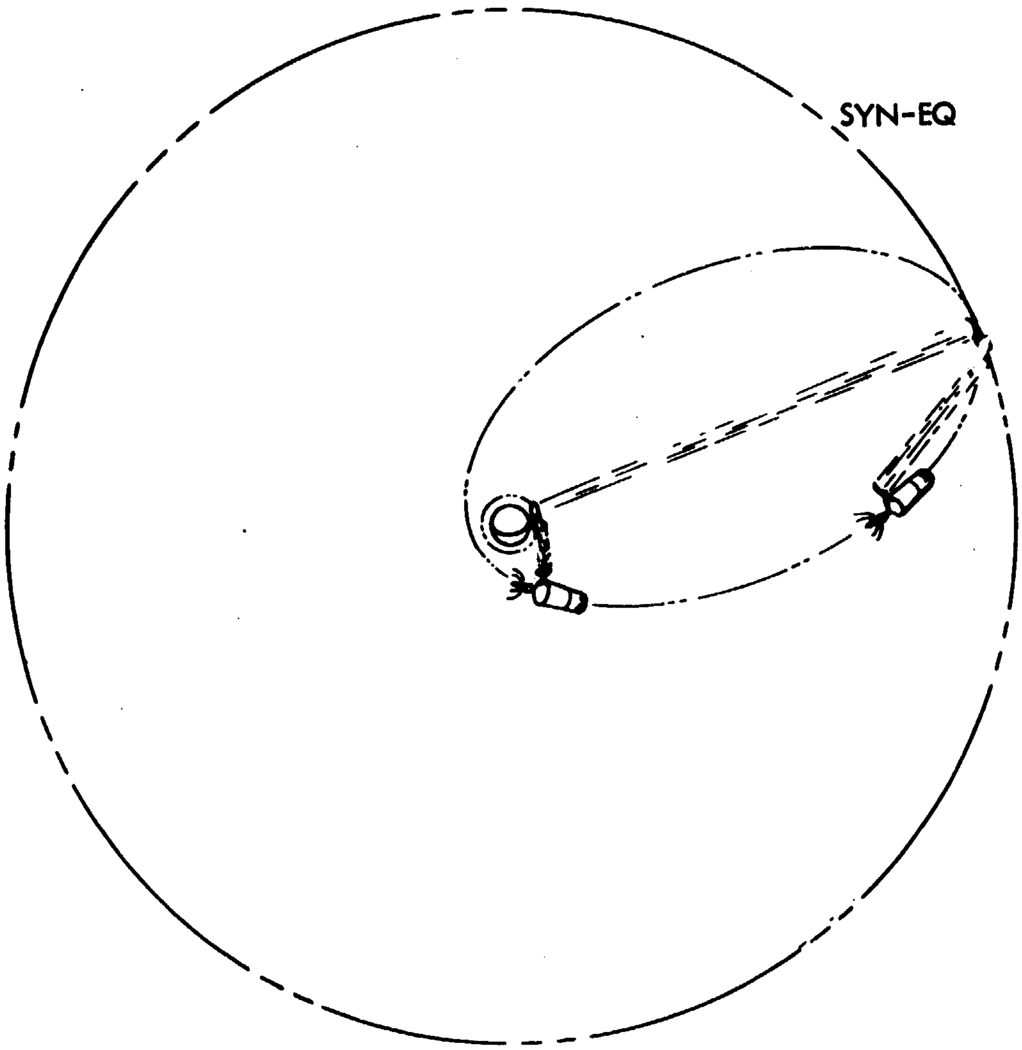


Figure 22. Relay concept

power and independent of the transmitter aperture diameter. Figure 23 shows the normalized power extinction relative to zenith angles for transmitter altitudes of sea level, 6000 ft, and 12,000 ft for three different wavelengths (0.5, 2.7, and 3.8 μm). The computation for the 2.7-μm wavelength assumed that the best four lines of an HF laser were used. Clearly, the higher transmitter altitudes and lower zenith angles are desirable regardless of the wavelength. Absorption by ozone in the visible wavelengths (Figure 24) accounts for the higher extinction at 0.5-μm wavelength shown in Figure 23; however, from a system point of view, the smaller aperture requirements more than overcome the higher losses of extinction.

- Beam Divergence

The dominant factor causing the laser beam to spread in space is jitter, even when the jitter is a small fraction of a microradian. In the atmosphere, however, the beamspread is dominated by atmospheric turbulence and thermal blooming, if present. Thermal blooming is the result of excess heating of the air caused by a high concentration of laser power. This concentrated laser power can be avoided by larger transmitter apertures which spreads the power over larger areas so that the concentration of power is below the thermal blooming threshold. Below the threshold of thermal blooming, atmospheric turbulence is the dominating factor on which larger apertures will have little or no effect. Turbulence is atmospheric dependent as shown in Figure 25 and decreases with altitude of the transmitter above the ground, except there is a spike at the tropopause (in the region of the jet streams, for example). More than 60% of the turbulence occurs in the first 100 m above the ground as shown in Figure 26. This means there is little difference in turbulence whether the laser site is located at sea level or on some mountain top. Figure 27 shows the beam radius, in terms of beamspread angle, plotted against the cosine of the zenith angle (cos Z), and illustrates the point that turbulence does not change significantly with the site altitude. For example, the 20- and 40-m diameter apertures for the 0.5-μm wavelength show the same beam radius for site locations at sea level, 6000 ft, and 12,000 ft. For the 10-m-diameter aperture, the increased radii for the higher power is caused by thermal blooming. In fact, all the curves in Figure 27 that are above the minimum curve in its group are higher because of thermal blooming or turbulence created by atmospheric heating from reaction with the laser beam.

All the above beam radii have assumed a fixed aperture configuration which can be improved upon by the use of adaptive optics with actuators to change the mirror surface figure to compensate for beamspread caused by turbulence. The amount of compensation possible is dependent upon the wavelength and the number of actuators per unit area. Figure 28 shows the beam radius relative to a diffraction-limited beam in relation to the number of corrective actuators for 0.5-, 2.7-, and 3.8-μm wavelengths and 10-, 20-, and 40-m-diameter apertures. As may be noted, more actuators per square meter are needed for the shorter wavelengths. Figures 29, 30, and 31 show the effectiveness of correction for the three wavelengths and three aperture diameters. These data were derived assuming a control system that worked perfectly.

All these data must be considered when selecting a laser transmitter site and when determining deployment parameters for the MERU's.

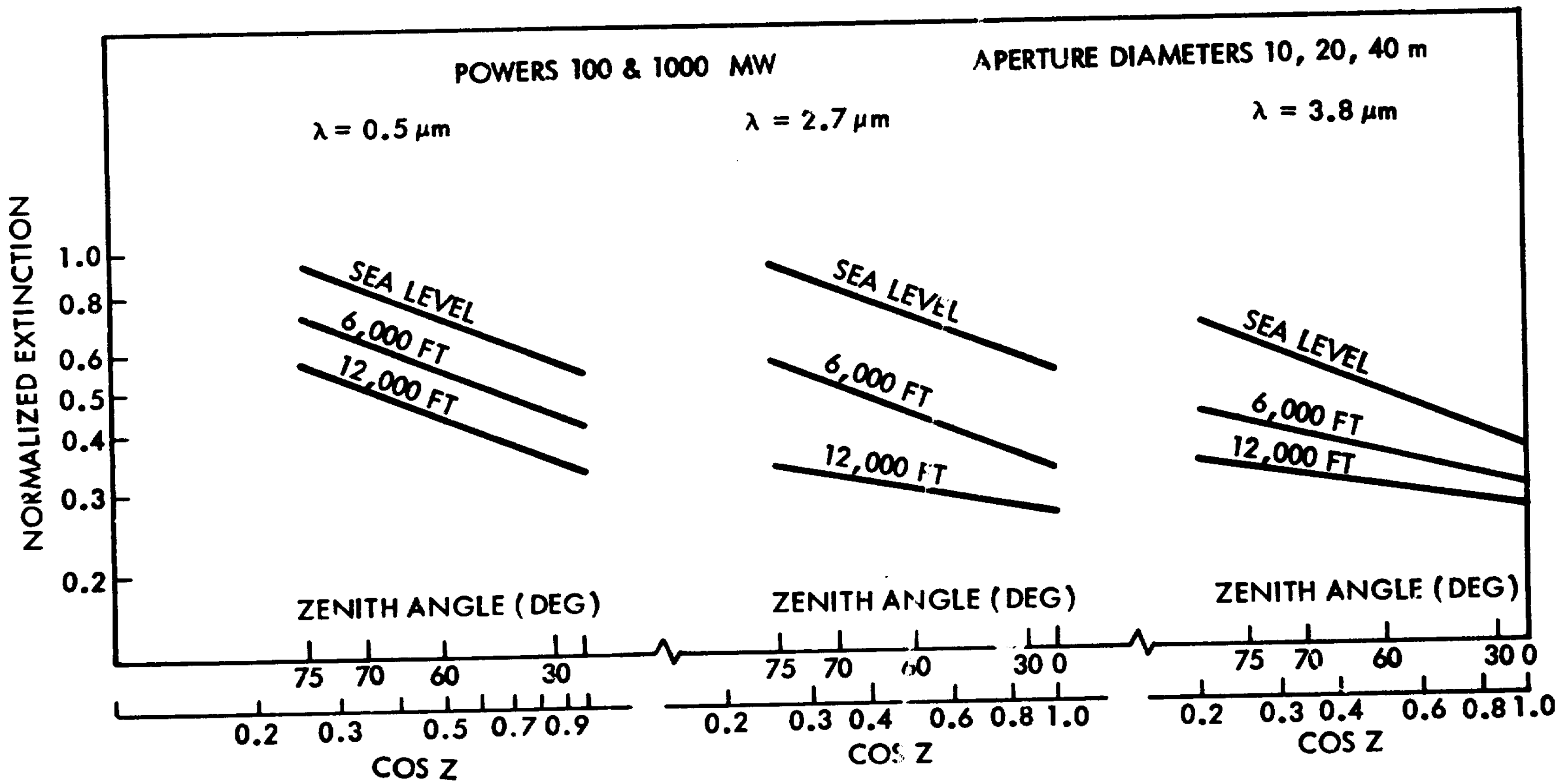


Figure 23. Power extinction versus zenith angle and transmitter altitude

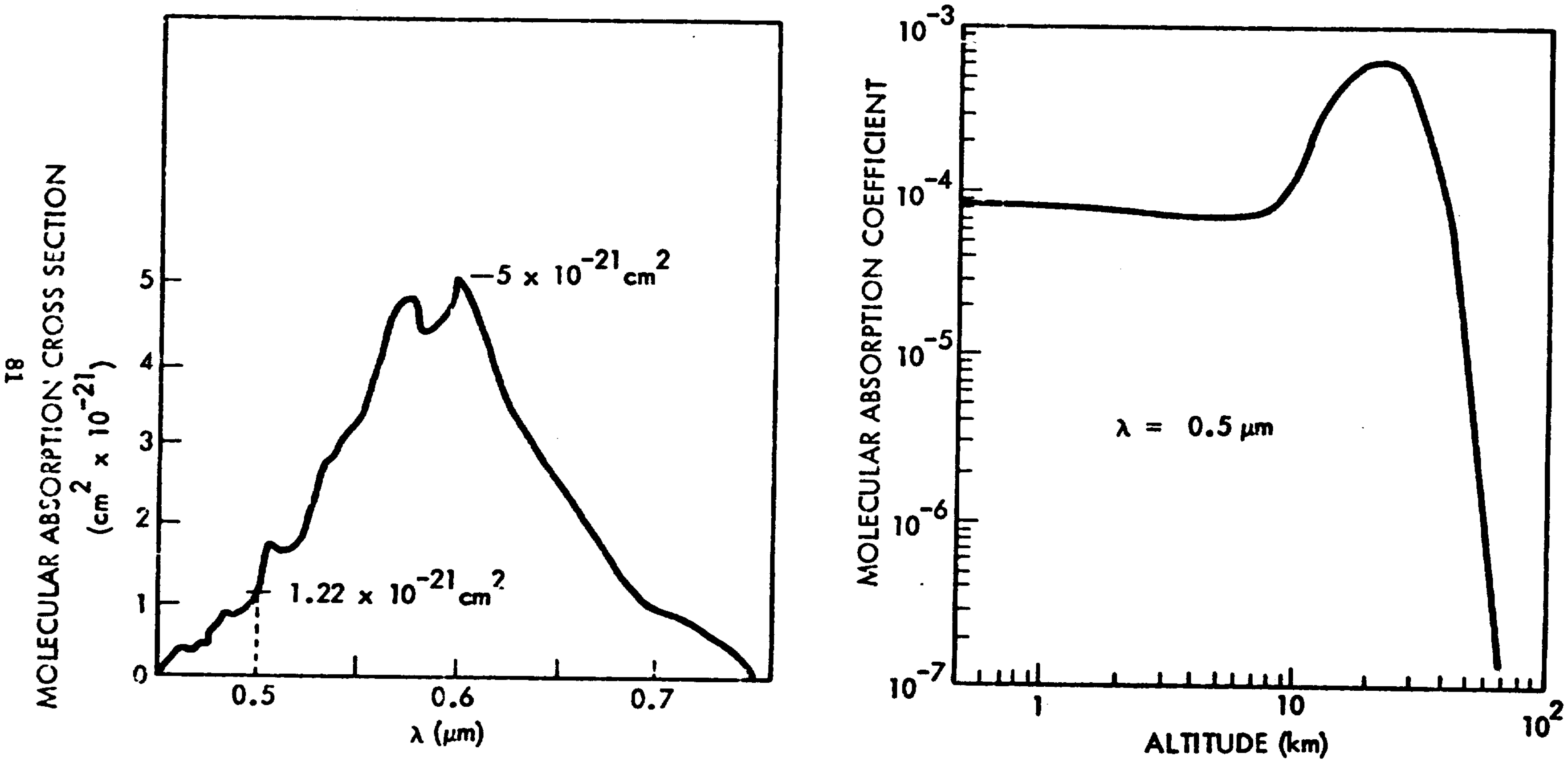


Figure 24. Absorption by ozone in the visible

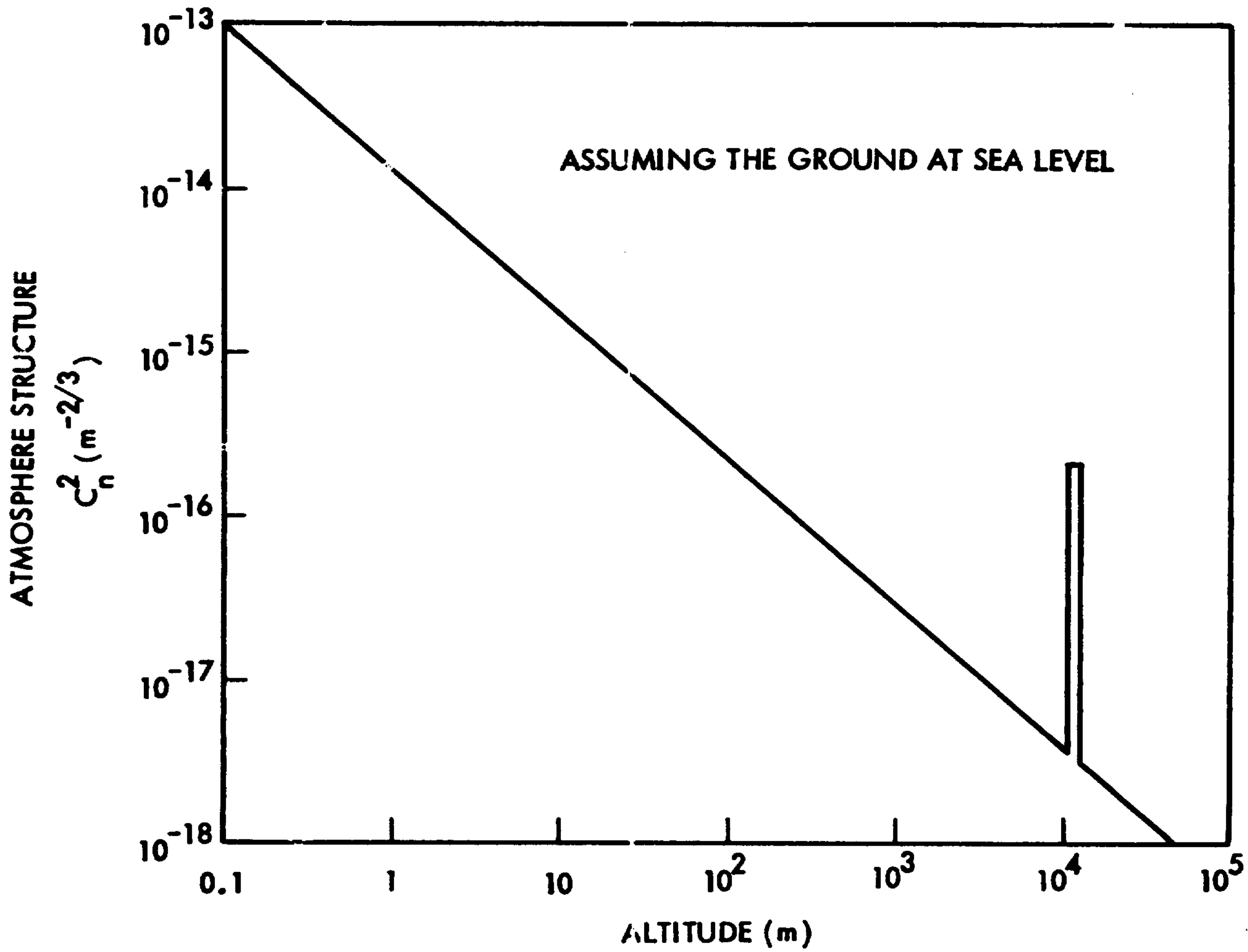


Figure 25. Turbulence altitude dependence

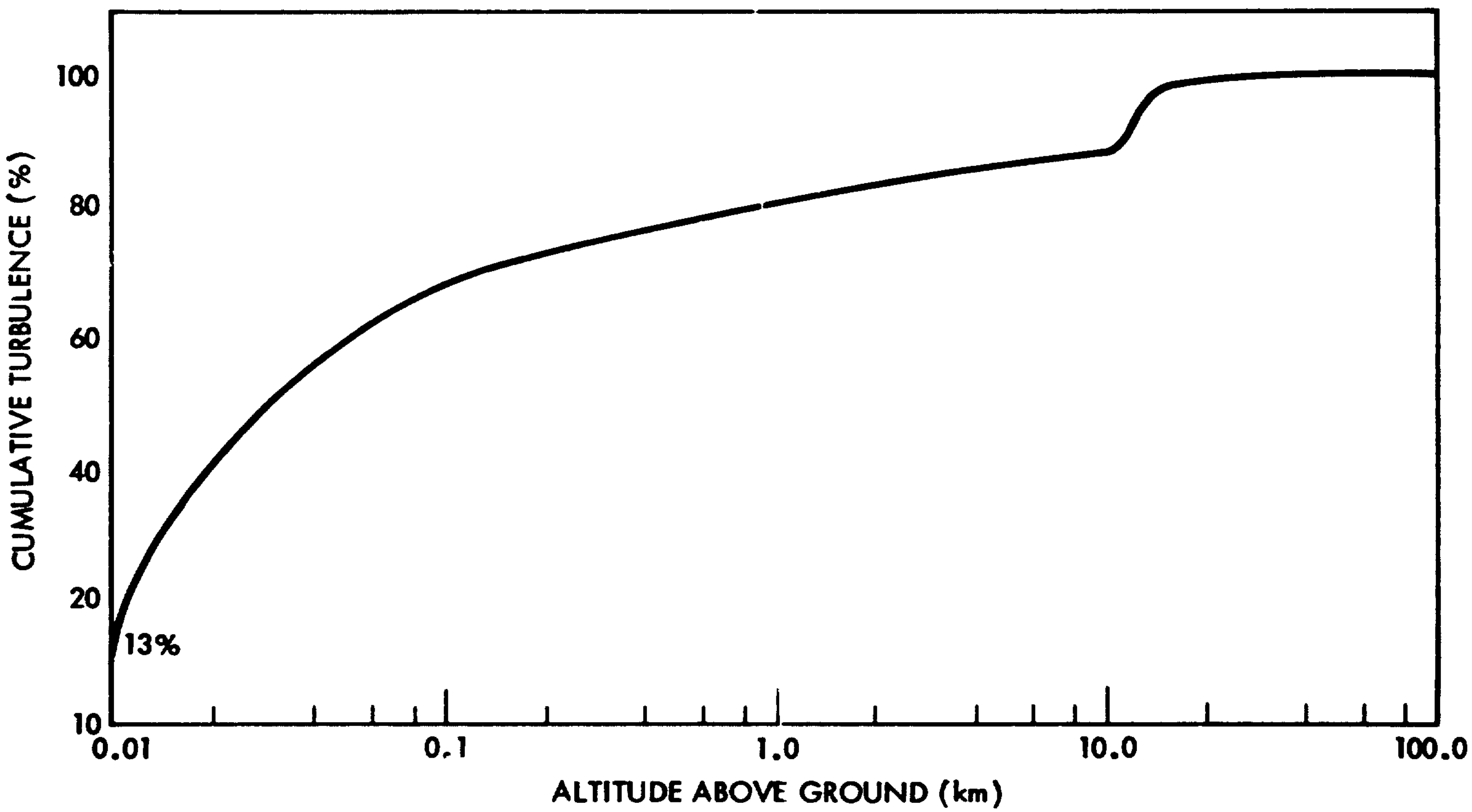


Figure 26. Cumulative turbulence beamspread

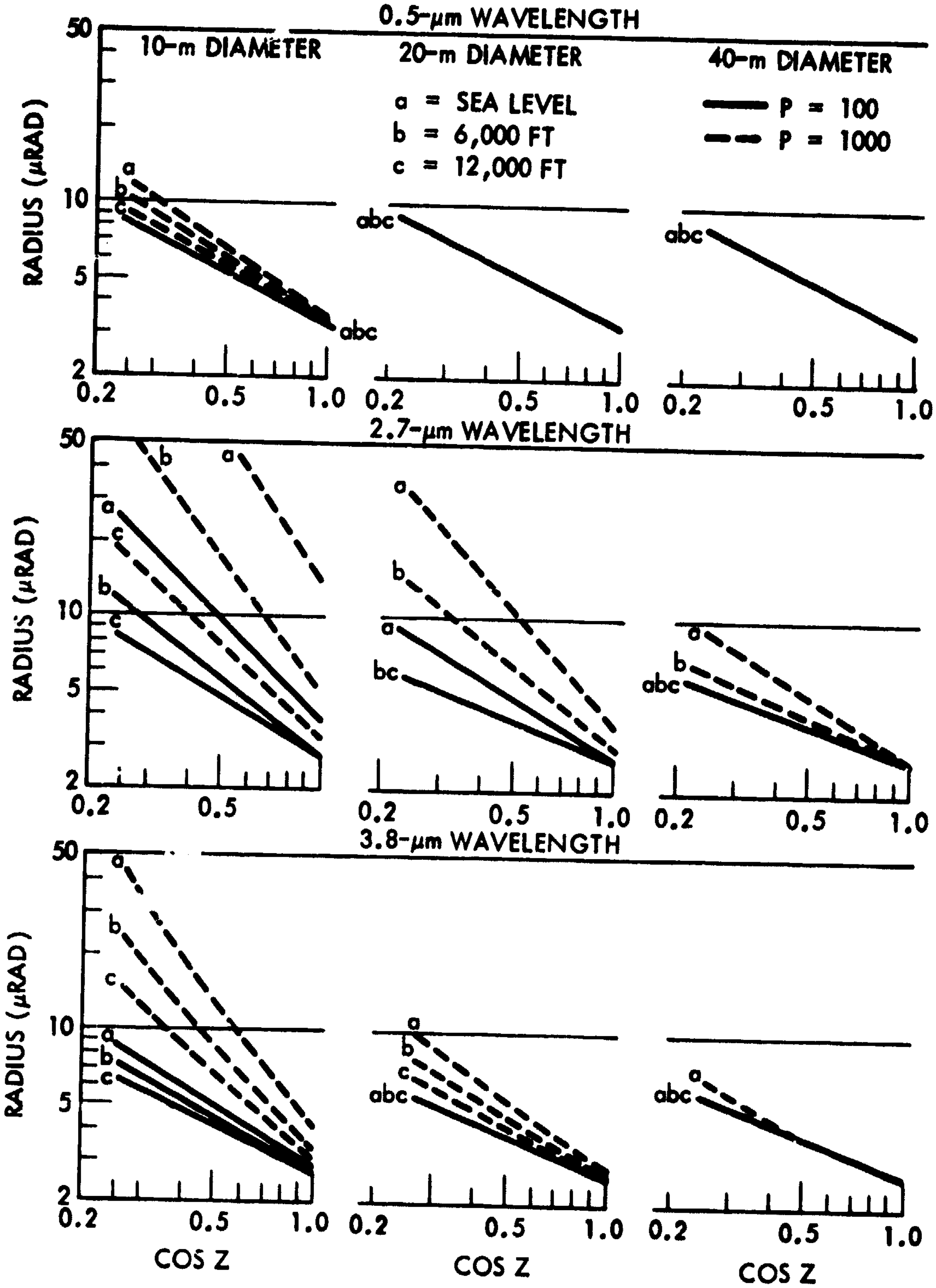


Figure 27. Beam radii

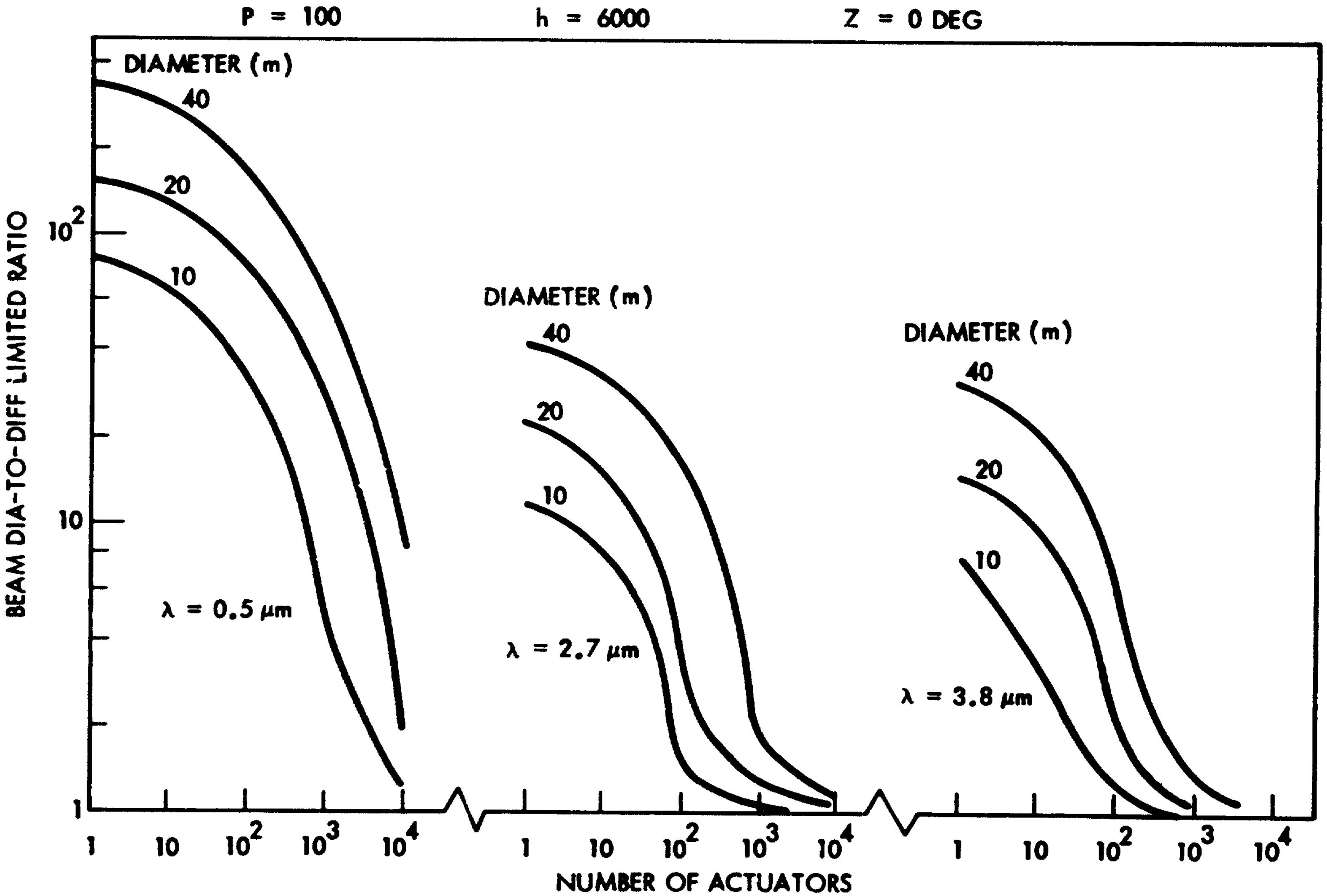


Figure 28. Actuator requirements

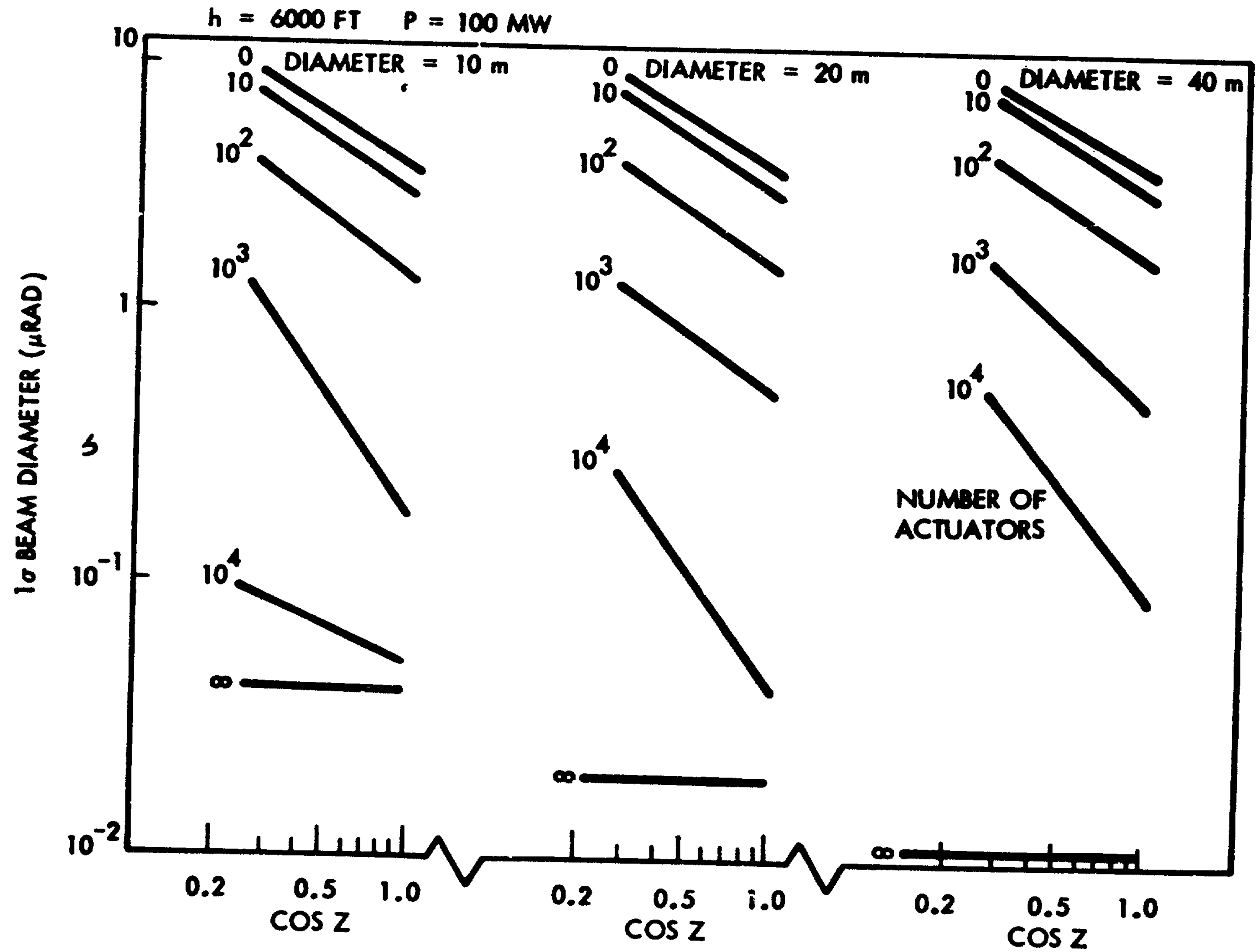


Figure 29. Turbulence correction effectiveness, $\lambda = 0.5\ \mu m$

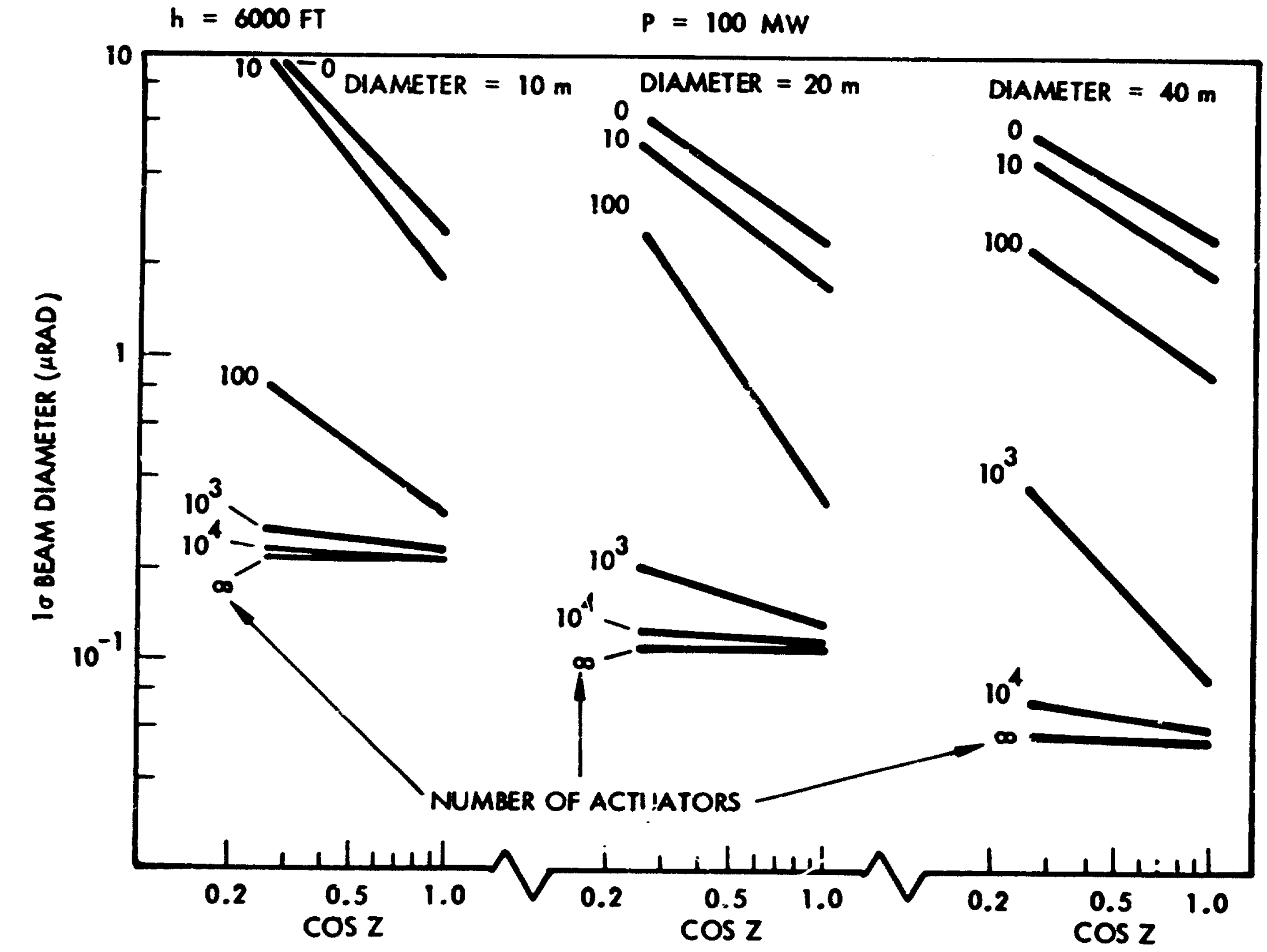


Figure 30. Turbulence correction effectiveness, $\lambda = 2.7\ \mu m$

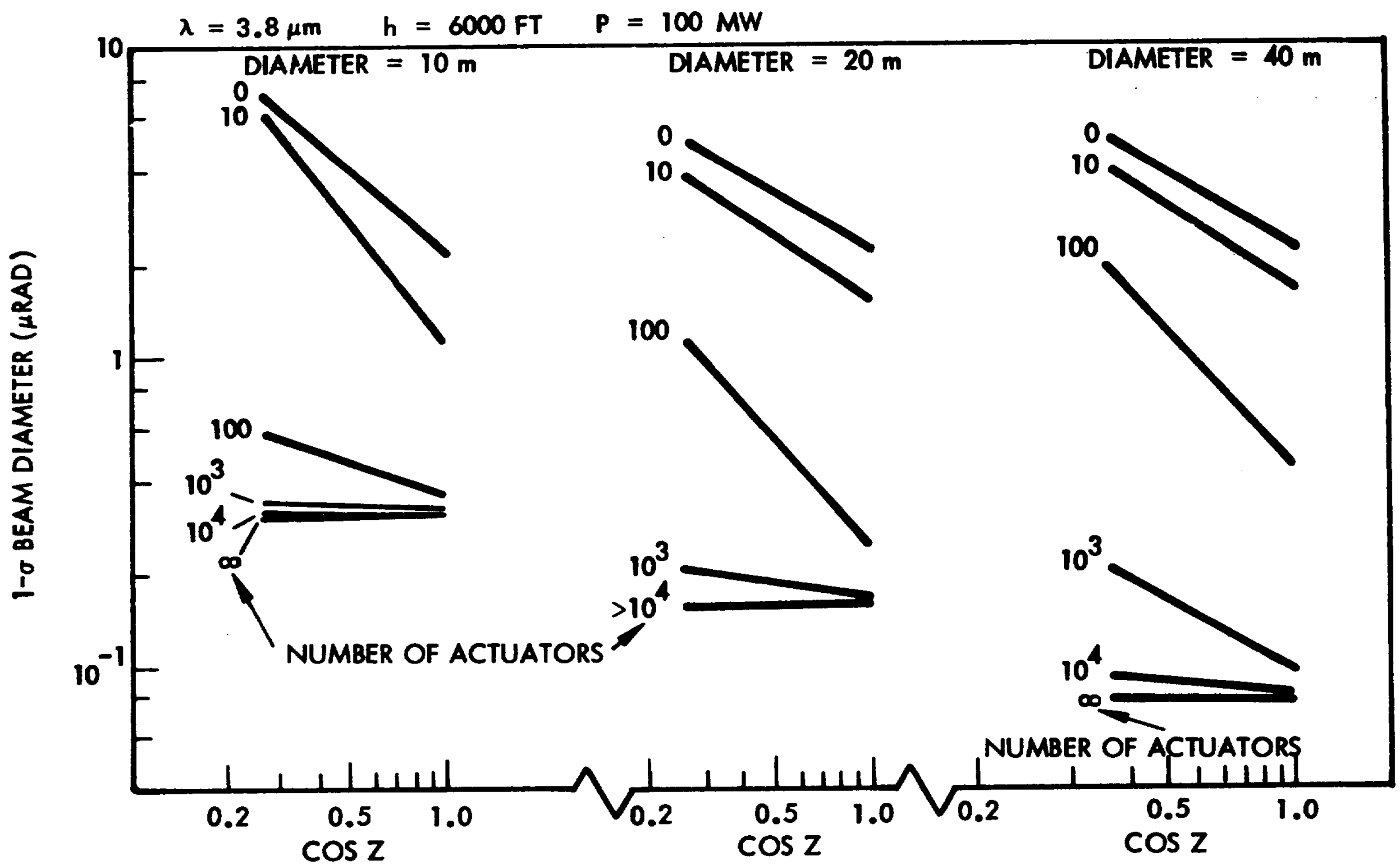


Figure 31. Turbulence correction effectiveness, λ = 3.8 μm

The foregoing has explained that adaptive optics can correct for some aberrations caused by propagation through the atmosphere. Figure 32 illustrates one method of determining what the correction should be. By using a beacon laser with a different wavelength (λ_2) on the relay and knowing the wavefront conditions as it leaves the relay, the beacon beam can be diverted into a wavefront analyzer at the ground transmitter so that the effects of its propagation is known. The primary actuators can then adjust the outgoing beam to compensate for the atmospheric condition. In addition, the relay unit also samples its outgoing beam for wavefront corrections which will also indicate the condition of the HEL wavefront as it is received by the relay. If the top of the atmosphere is considered to be 30 km (16 nmi), then the time from the first atmospheric aberration until the beam is in the analyzer is approximately 100 μs. However most aberrations are within a few hundred meters of the ground laser which is less than 1 μs. The relay unit is in a 4-hr orbit which means it is traveling less than 0.44 mrad/sec (0.025 deg/sec) about the center of a nonrotating earth or in 1 s the relay travels 5,702 m (18,708 ft); therefore, in 100 μs the relay travels 0.57 m (1.87 ft) and the line-of-sight has moved 2.6 mm (0.10 in.) at the top of the atmosphere. If the data processing time is 100 μs, then the LOS has moved an additional 2.6 mm. Actuator response for this application needs to be 1 to 2 kHz and using 1 kHz, the LOS will have moved another 2.6 cm (1.0 in.). The travel time back to the top of the atmosphere adds another 2.6 mm; therefore, the leading edge of the beam is traveling 3.38 cm (1.33 in.) ahead of the original LOS at the top of the atmosphere. This brief analysis was with a nonrotating earth and a rotating earth would be slightly less.

Pointing and tracking between space vehicles is the same as described in the space-based laser rocket analysis, but the GLTU to MERU must consider the refraction that will occur in the atmosphere. Figure 33 shows two sets of curves – the standard index of refraction of air and the refraction differences between wavelengths for various zenith angles. The two wavelengths represent the primary HEL beam and the pointer tracker beacon. The figure illustrates that the distance along the ordinate of the standard index of refraction is the important consideration. That is, if the HEL wavelength is short ($\lambda = 0.5$), then the beacon wavelength also needs to be close to the same wavelength to minimize differences in refraction. As the HEL wavelength increases, a wider choice of beacon wavelengths is permitted because the curve begins to level out above wavelength of 1.06 μm.

3.5.2 Laser Site Selection and MERU Orbit Parameters

Selection of a site for the laser transmitter requires consideration of weather conditions, site altitude, and the interaction with the Medium Orbit Energy Relay Units (MERUs). Figure 34 shows sites with acceptable weather and altitude and need only be considered in relation to the MERU orbital parameters. Orbital parameters can be established so that a MERU is within line-of-sight of any of the locations. For example, Great Falls, Montana, has a relatively high latitude that would require an orbit inclination on the order of 45° to provide a minimum zenith angle approaching zero. Satellites with orbital periods divisible into 24 hr could be spaced so that as one satellite

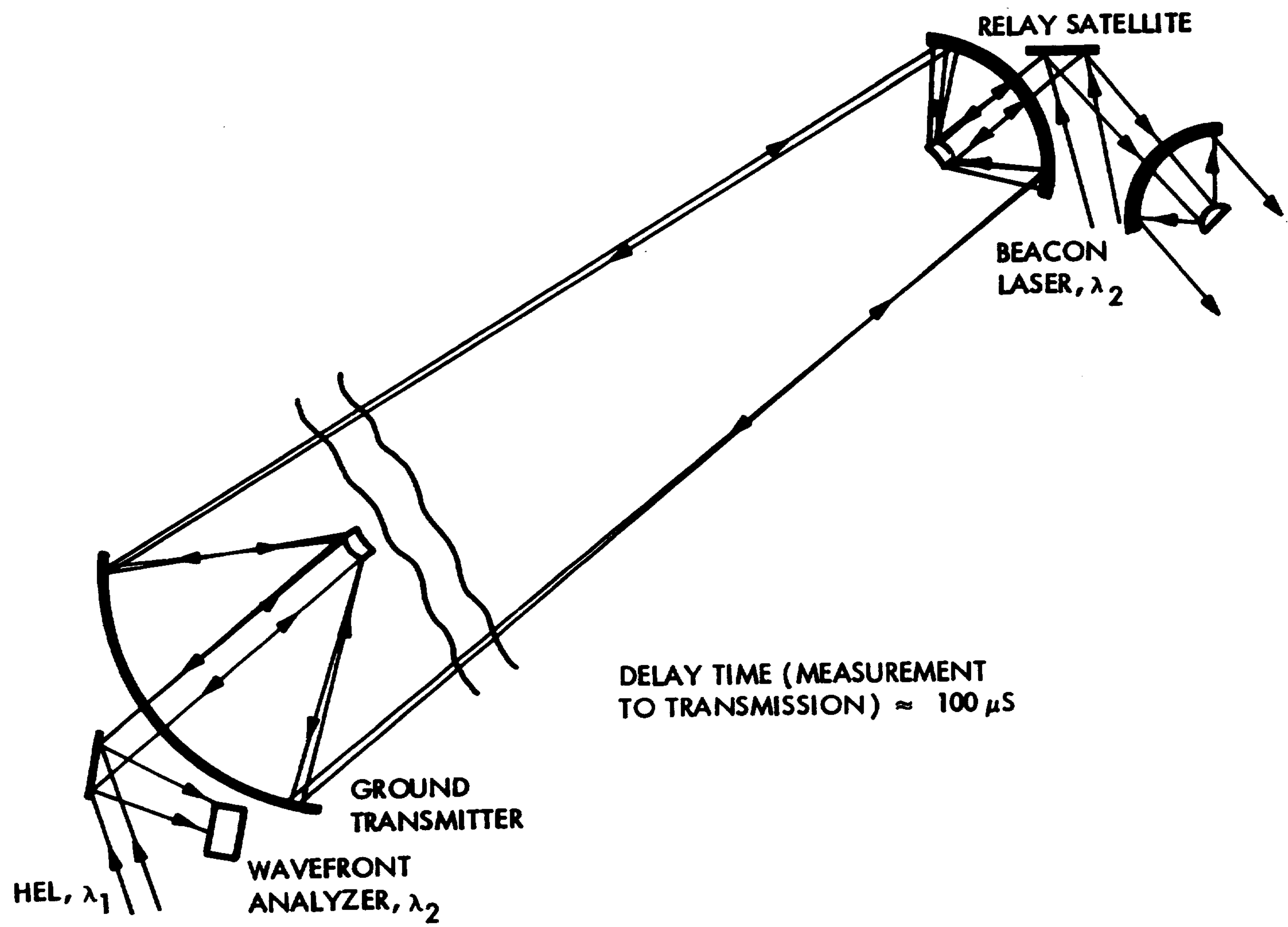


Figure 32. Concept for adaptive optics measurements

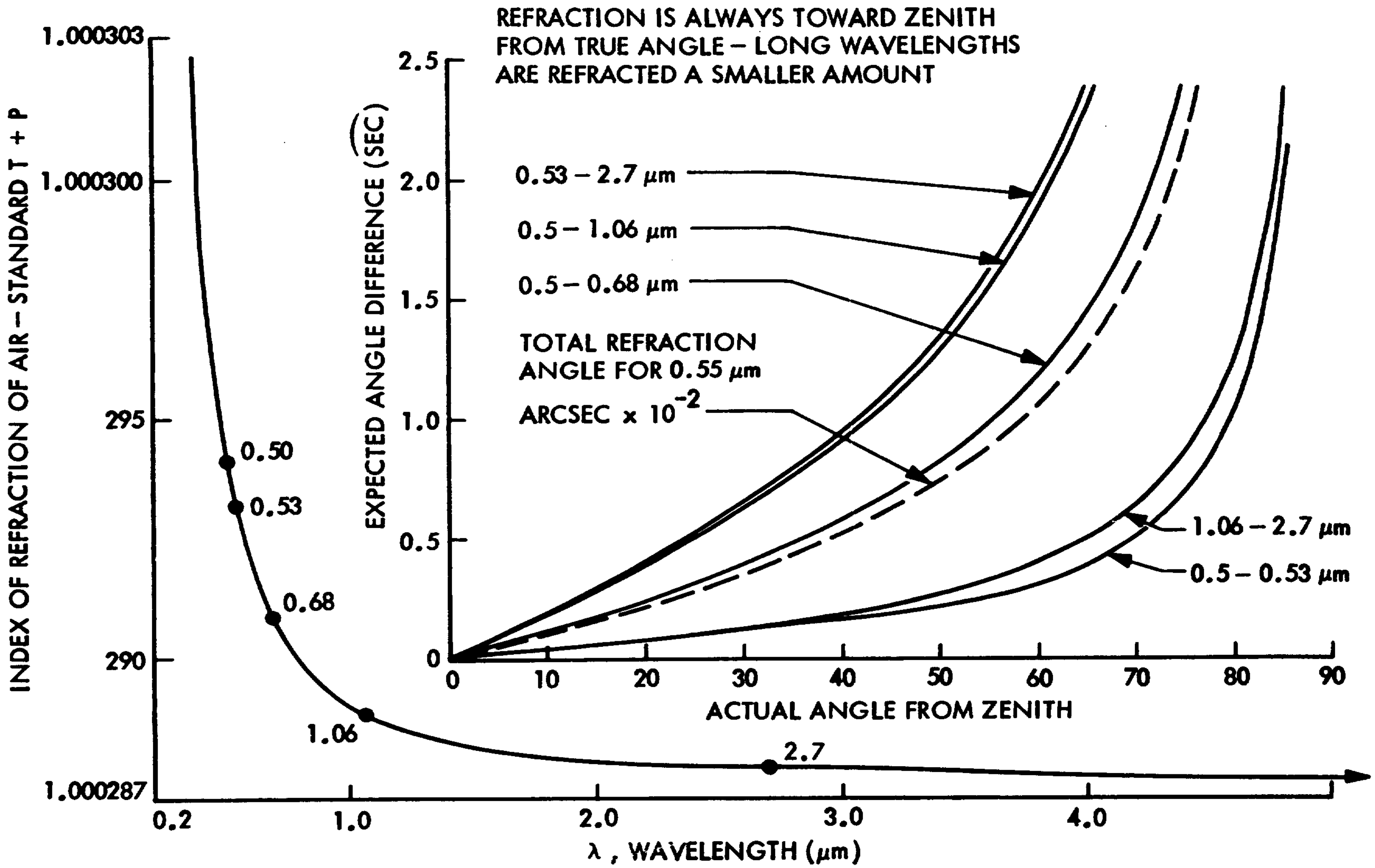


Figure 33. Atmospheric refraction

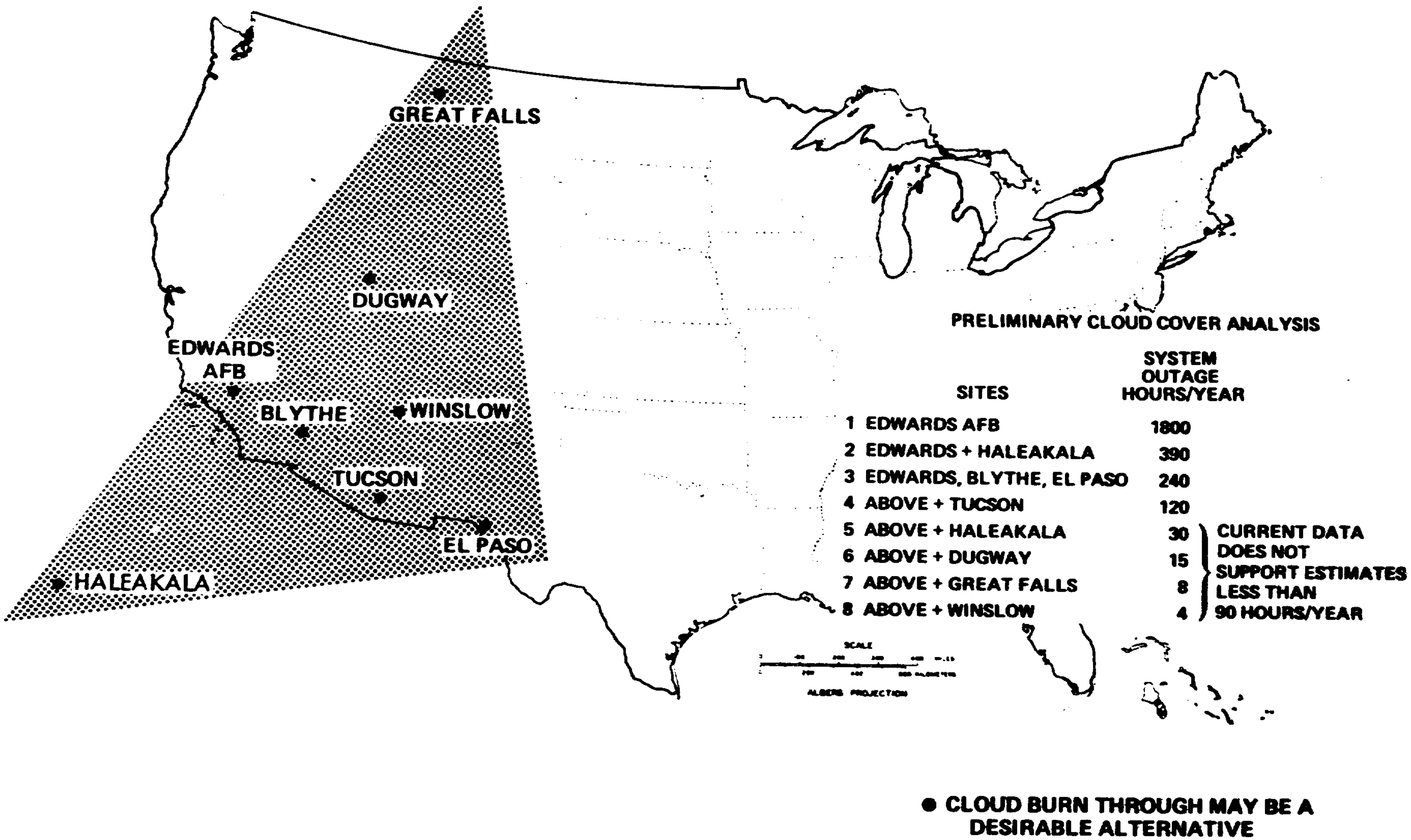


Figure 34. Sites considered

leaves the viewing constraints, another will appear. However, the propulsive impulses of the PUs is desired to be near the equator so that the apogee also remains near the equatorial plane. Off-axis thrusting by the PU can, of course, maintain apogee where it is desired, but the velocity penalty would increase the PU size to accommodate the additional propellant required. On the opposite latitude extreme is Haleakala, Hawaii, which is approximately 20°N latitude, and an equatorial orbit as shown in Figure 35 can be used so that propulsive burns can be made on both sides of the equatorial plane. From the section on atmospheric propagation it may be noted that the amount of energy leaving the atmosphere is a function of the cosine of the zenith angle and decreases rapidly with zenith angles greater than 60°. Figure 36 shows the calculations for determining the maximum and minimum zenith angles and ranges from the ground site to the relay which are:

	Max	Min
Zenith Angle (deg)	61.0	37.4
Range (km)	8,607	7,299

This establishes the ranges that the laser transmitter must transmit energy and the beamspread angle which determines the size of the receiver aperture on the MERU. These parameters permit the GLTU to be usable at any time with six MERUs. This deployment scheme provides more flexibility relative to energy transfer opportunities than the space-based system. The ground system can transmit energy any time the PU is within range of the MERU orbit without respect to the location of the laser, whereas in the space system not only does the PU need to be in range of the transmitter orbit, but the transmitter must also be in that part of its orbit. The space system could also use relays to accomplish a burn any time the PU was in range of the orbit, but it is not required. The short orbit period (~ 1.5 hr) for the space LTU and the fact that it is in the same orbital plane (28.5° inclination) assures that energy transmission opportunities will occur every orbit of the PU.

The 6580-km (3557-nmi) altitude for the MERU give approximately a 4-hr-orbit period. With six MERUs and a nonrotating earth, the MERUs would be in view of the GLTU for 40 min; however, because the earth is rotating in the same direction as the MERUs, the viewing time is nearly 50 min as shown in Figure 37. This figure also shows the zenith angle and range during the view time. These data are required for sizing the GLTU.

3.5.3 Sizing the Ground Laser Transmitting Unit (GLTU)

In the space-based laser rocket system, the laser transmitting aperture was as large as the study bounds would permit to minimize receiving mirror sizes over the long ranges. The GLTU, however, has a maximum range of 8600 km (4648 nmi) to the MERU as shown in Figure 37. Additionally, the atmospheric effect to beamspread does not necessarily lessen with larger apertures. This is dependent upon the number of actuators actively controlling the mirror surface to correct atmospheric-induced

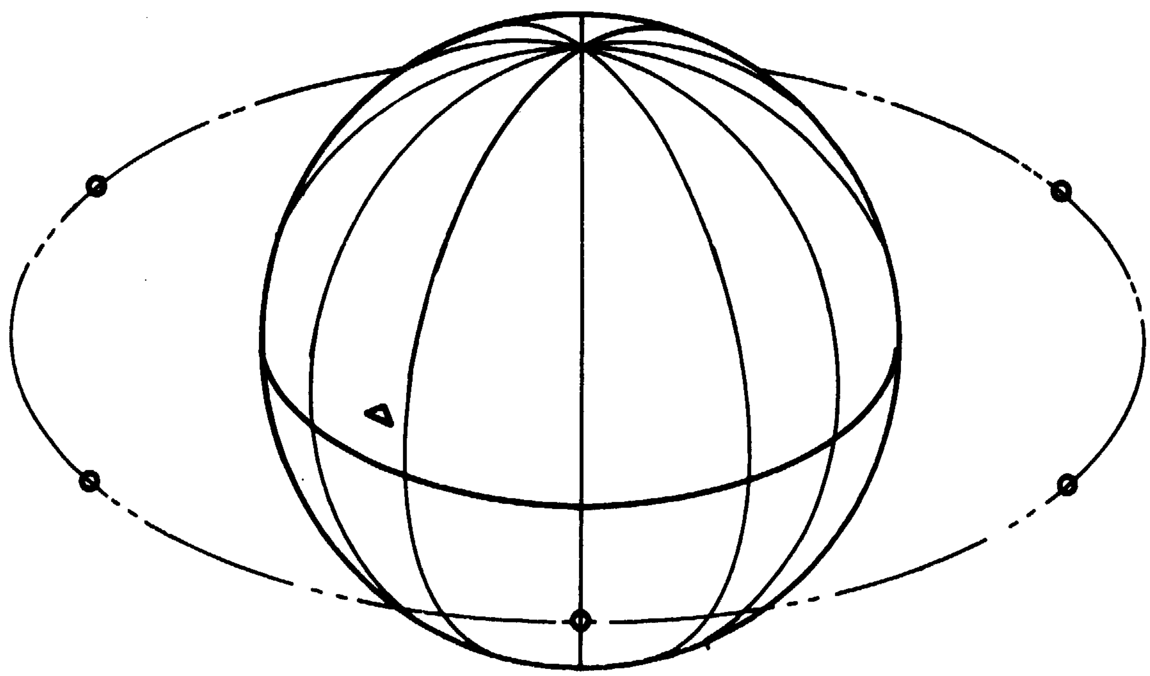
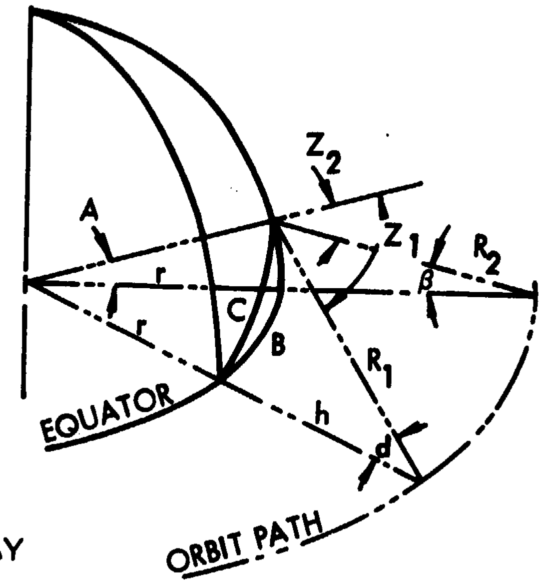


Figure 35. Medium Earth Orbit Relay



A = LATITUDE OF SITE = 20°
r = EARTH RADIUS = 6380 km
h = RELAY ALTITUDE = 6580 km
B = 1/2 ANGLE BETWEEN RELAYS = 30°
C = MAX EARTH ANGLE FROM SITE TO RELAY
R_1 = MAX RANGE
R_2 = MIN RANGE
Z_1 = MAX ZENITH ANGLE
Z_2 = MIN ZENITH ANGLE

$$C = \cos^{-1}(\cos A \cos B) = 35.5313°$$

$$R_1 = \left[r^2 + (r+h)^2 - 2r(r+h)\cos C\right]^{1/2} = 8607.4690 \text{ km}$$

$$d = \sin^{-1}\left(\frac{r}{R_1}\sin C\right) = 25.5156°$$

$$Z_1 = C + d = 61.0469°$$

$$R_2 = \left[r^2 + (r+h)^2 - 2r(r+h)\cos A\right]^{1/2} = 7298.5894 \text{ km}$$

$$\beta = \sin^{-1}\left(\frac{r}{R_2}\sin A\right) = 17.3960°$$

$$Z_2 = A + \beta = 37.3960°$$

RANGE BETWEEN RELAYS = 12,960 km

Figure 36. Maximum and minimum ranges and zenith angles

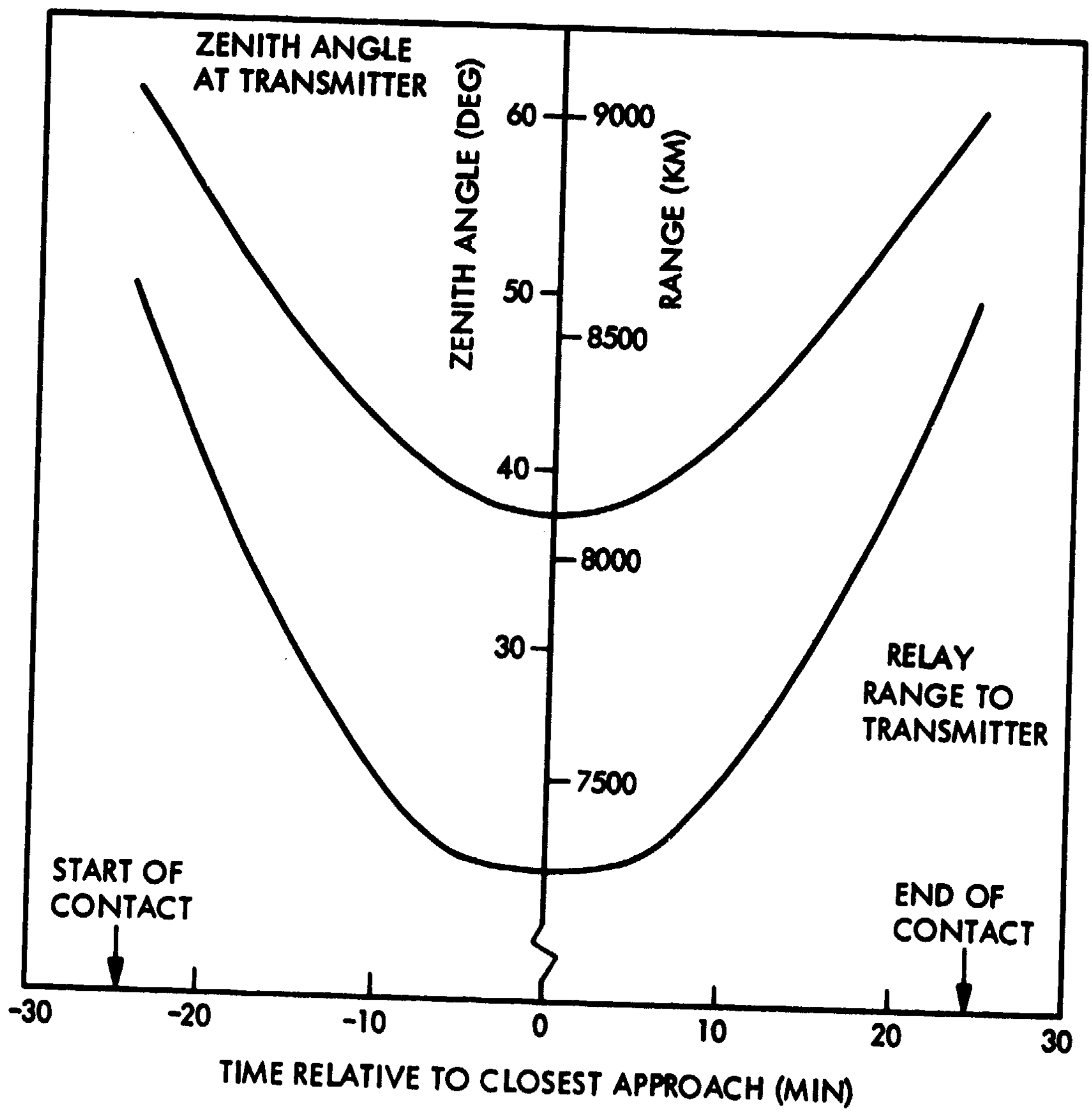


Figure 37. Variation of zenith angle and range during contact

beamspread. Figure 38 illustrates that a 10-m (32.8-ft) diameter mirror with 10,000 actuators for correcting a 0.5-μm wavelength is better than a 20-m (65.6-ft) diameter mirror with the same number of actuators within the zenith angle working range. The percent of power leaving the atmosphere is shown in the right-hand curve and ranges from approximately 52% to 60% within the zenith angle range. As a result a 10-m-diameter transmitting aperture with 10,000 actuators was selected for the ground laser transmitter unit.

The laser power was selected based on 16 MW (P_1) leaving the MERU to provide 13.4 MW at the propulsion unit thruster. The MERU has a total of eight mirrors each of which is 99.7% reflective, therefore power (P_2) into the receiver is:

$$P_2 = P_1/0.997^8 = 16.39 \text{ MW}$$

The MERU receiver is sized to pick up 84% of the power leaving the atmosphere (P_3) therefore:

$$P_3 = P_2/0.84 = 19.51 \text{ MW}$$

The atmospheric absorption and extinction permits 52% of the required power (P_r) to leave the atmosphere at the maximum zenith angle, therefore

$$P_r = P_3/0.52 = 37.5 \text{ MW}$$

Therefore the laser power requirement is established for the 2268-kg payload.

The laser power requirements for the SPS-type payloads is derived in the same way as for the small payloads, except that the receiver must intercept 95% of the energy exiting the atmosphere because of the flux density in the outer diffraction rings. The laser power requirement for the SPS type payloads is 1000 MW.

3.6 TASK 6: CONCEPTUAL DESIGN, GROUND-BASED LASER

As the ground-based laser rocket systems use the same propulsion units as the space-based laser rocket system, the conceptual designs, weight statements, etc., for the propulsion units will not be included in this section. For the propulsion unit data, see section 3.3.

3.6.1 Ground Laser Transmitter Unit (GLTU)

The GLTU is based at approximately 3660-m (12,000-ft) altitude on Mount Haleakala, Hawaii. The laser is assumed to be closed-cycle operating at a wavelength of 0.5 μm. The laser specific power is assumed to be 175 kJ/kg (80 kJ/lbm) similar to the CO_2 EDL, and the electrical/optical efficiency is assumed to be 20%. The cavity flow is

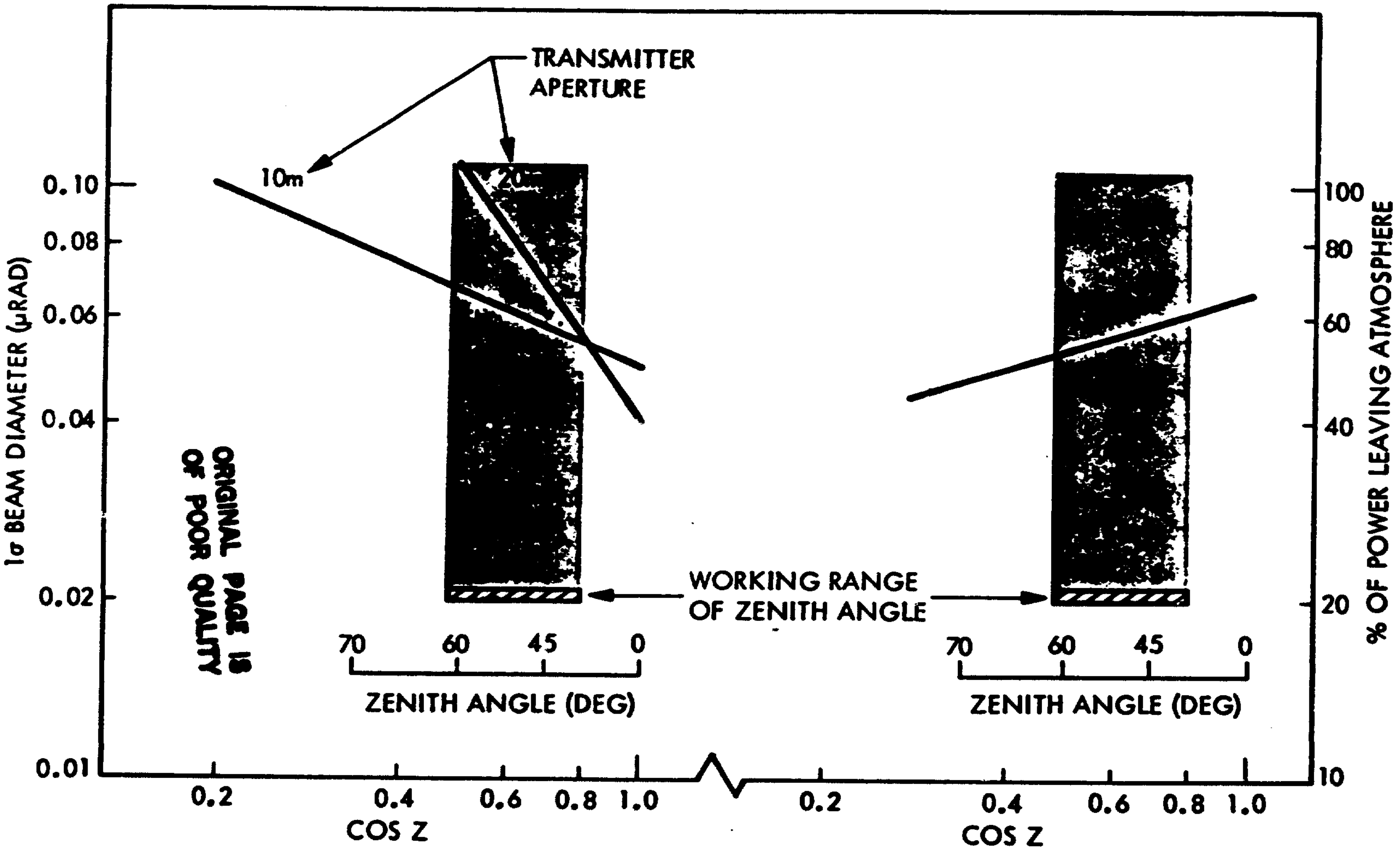


Figure 38. Effect of atmosphere on uplink beam

subsonic and driven by a turbine. Figure 39 shows the basic laser layout including a nonlinear axicon to redistribute and decrease the beam diameter. While this layout is for the 37.5-MW system, the 1000-MW system would be the same; only larger. The heat exchanger uses a closed-cycle water system as shown in the overall site layout Figure 40. The cooling system uses 1200 m^3/min (315,000 gal/min) and requires 18,180 m^2 (196,350 ft^2) of cooling area to dissipate the heat.

Turbo-generators were selected to provide the electrical power because of the intermittent power requirement and the ease of shutting down the generators. Enough fuel is stored on the site for 500-hr operation.

The transmitting aperture is 10-m (32.8-ft) diameter with adaptive segmented, mirror plates. Actuators controlling the mirror figure must have a bandwidth $\geq$ 1 kHz to minimize isoplanatic patch problems. Limited double gimbals provide the necessary pointing angles to track the MERU.

As weight is not a problem on earth, no specific weight statement was prepared.

3.6.2 Medium Orbit Energy Relay Unit (MERU)

The MERUs for both the 37.5- and 1000-MW laser rocket systems operate in a circular, equatorial orbit at 6580-km (3560-nmi) altitude. These relays receive the laser energy from the ground-based laser transmitters, corrects the wavefront errors and refocuses the beam, then relays the energy to another MERU, to the propulsion unit, or to the geosynchronous energy relay unit as appropriate.

Figure 41 shows the inboard profile of the MERU for both the 37.5- and 1000-MW systems, except for the cooling radiators which have been omitted for clarity.

The receiving apertures are segmented, off-axis optical systems to receive an unobscured incoming beam. The aperture is adaptive to maintain figure control which must be near-diffraction-limited performance to avoid inducing additional errors in the wavefront. The receiver reduces the beam diameter and directs it to the secondary which redirects the beam through the optical train. The transmitting apertures for both systems are adaptive, monolithic, Cassegrainian systems. Cooling is required for both systems. The transmitting apertures are double-gimbaled which, when coupled with the receiver, permits receiving and transmitting from and to any direction. Table XXVII shows the weights by subsystem for both the 37.5 and 1000 MW MERUs.

3.6.3 Geosynchronous Energy Relay Units

The Geosynchronous energy relay units (GERUs) for both the 37.5-MW and 1000-MW laser rocket systems operate at geosynchronous equatorial orbit to relay the beamed energy from the MERU at medium earth orbit to the propulsion unit near synchronous altitude. This relieves the propulsion units of the requirement to have large receiving apertures required for the long range. The GERU receives the laser beam, corrects wavefront errors, focuses the beam and directs it to the propulsion units.

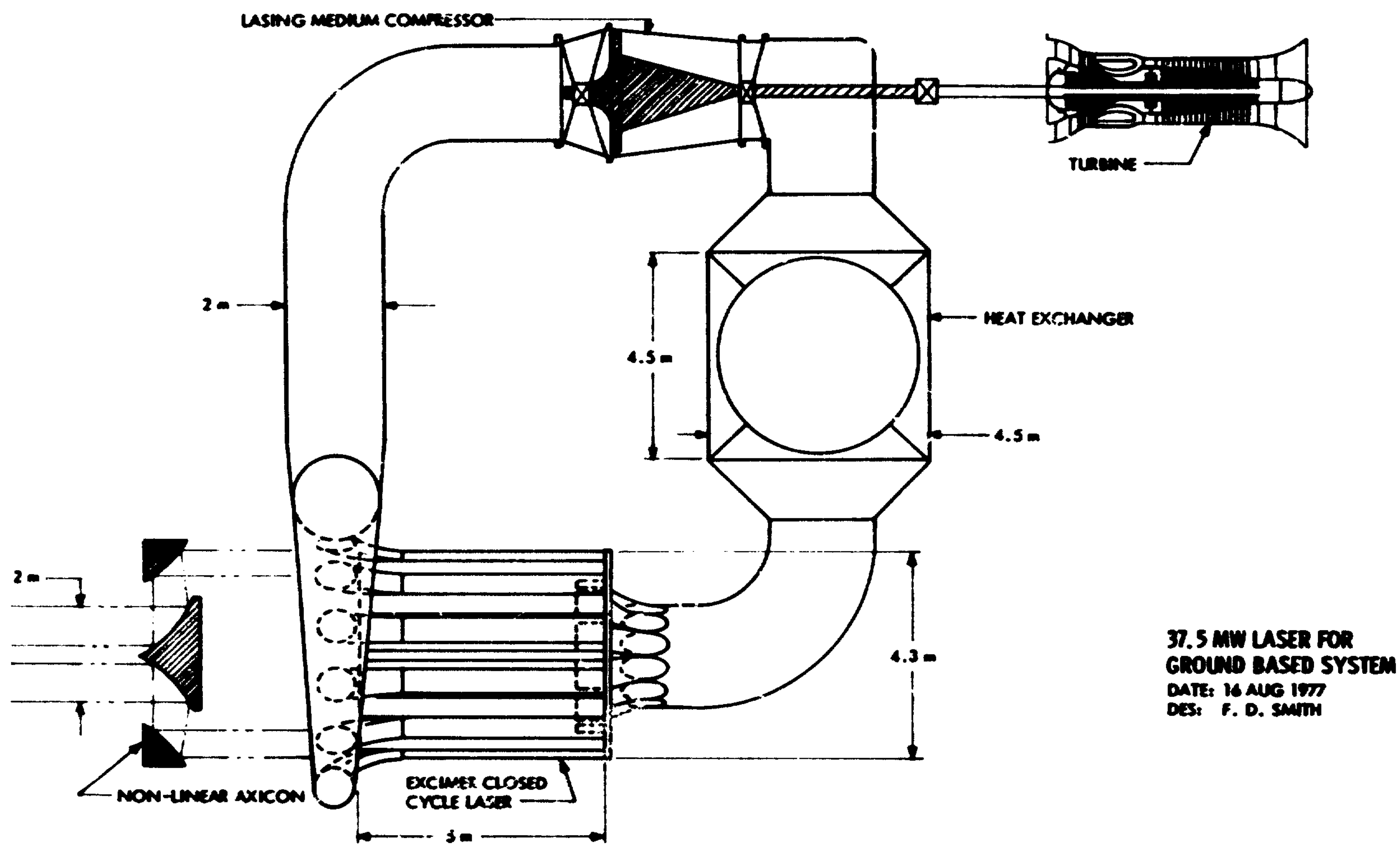
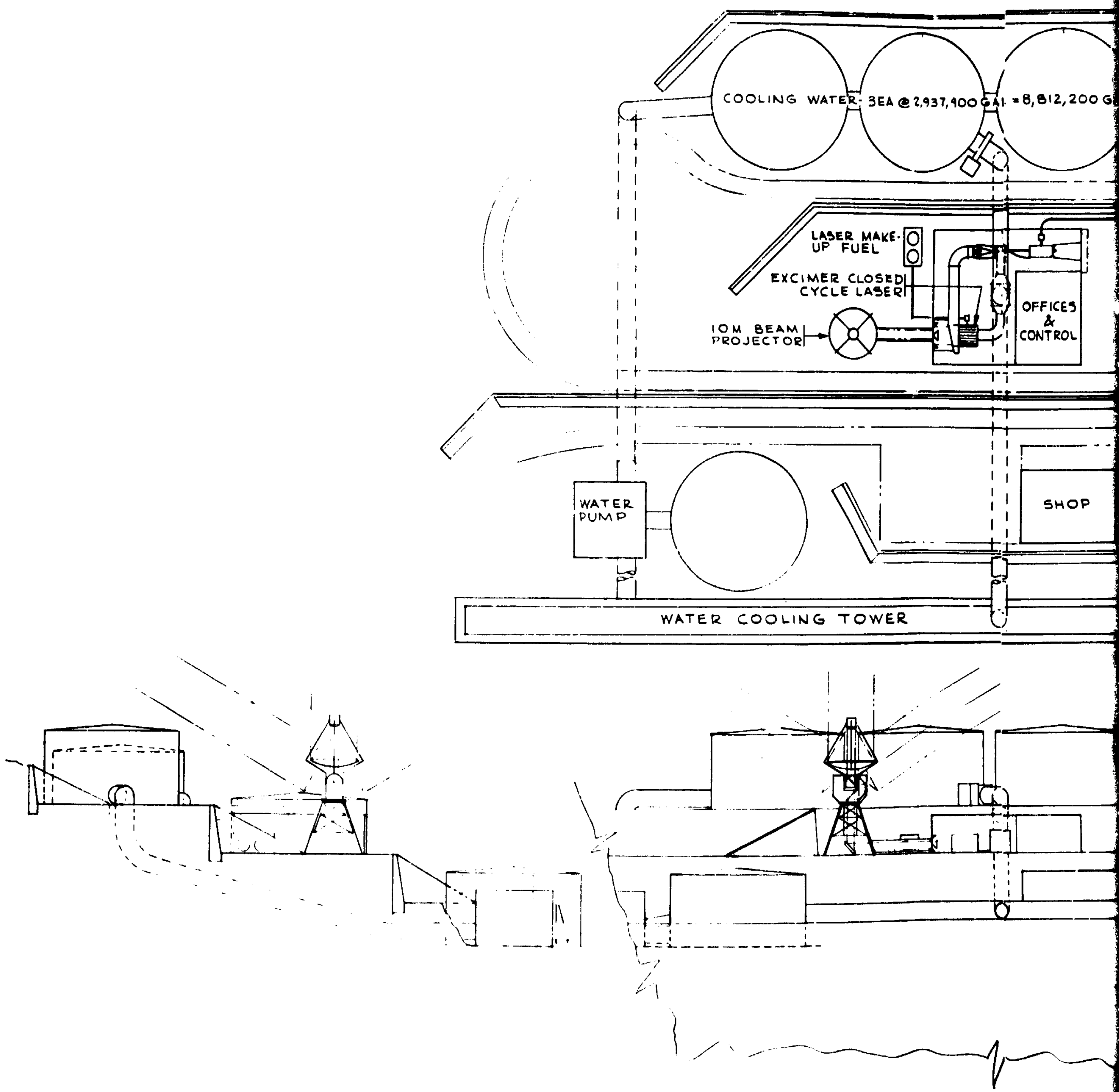


Figure 39. 37.5 MW ground laser transmitter unit



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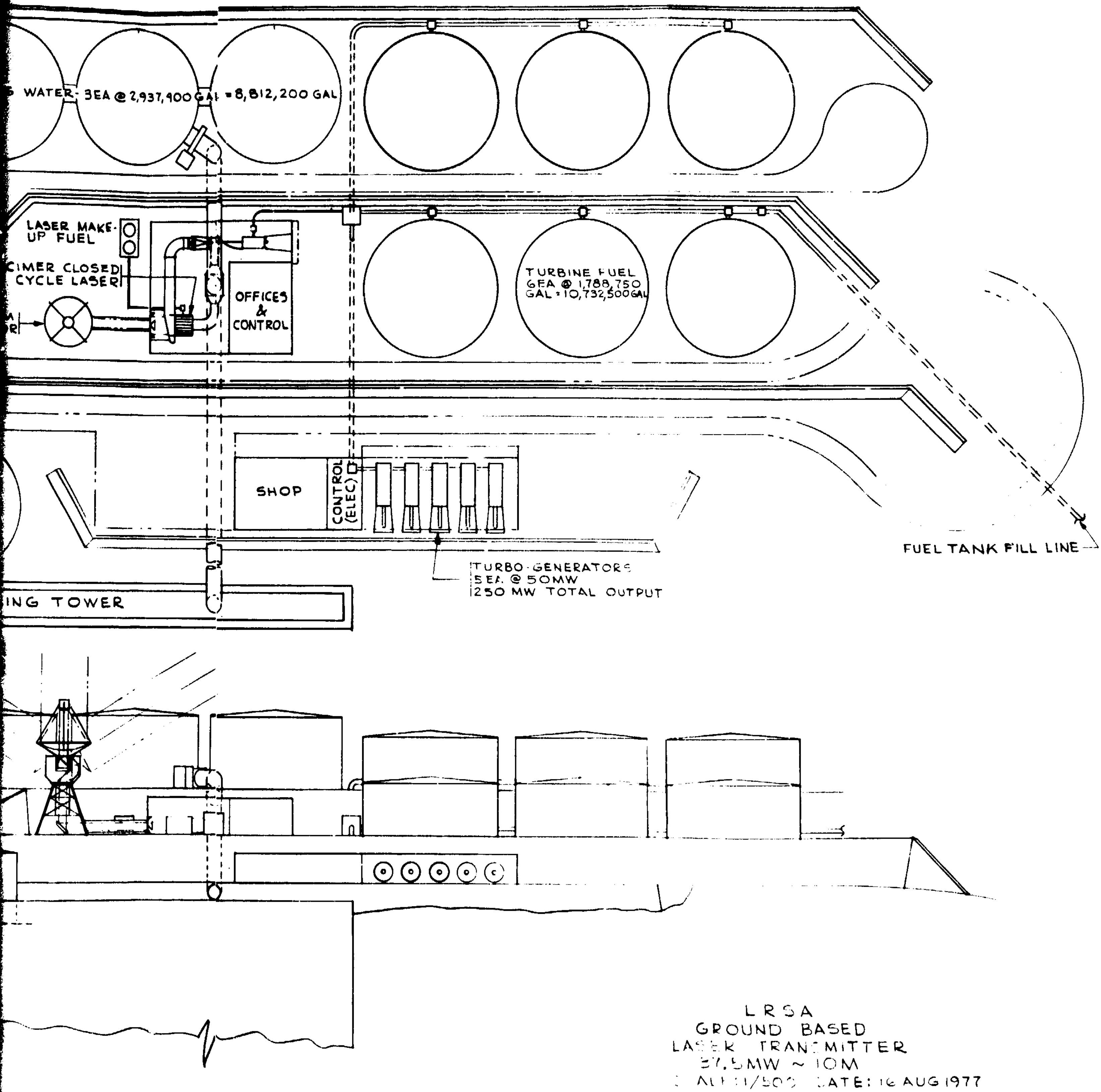

Figure 40. Overall GLTU site layout

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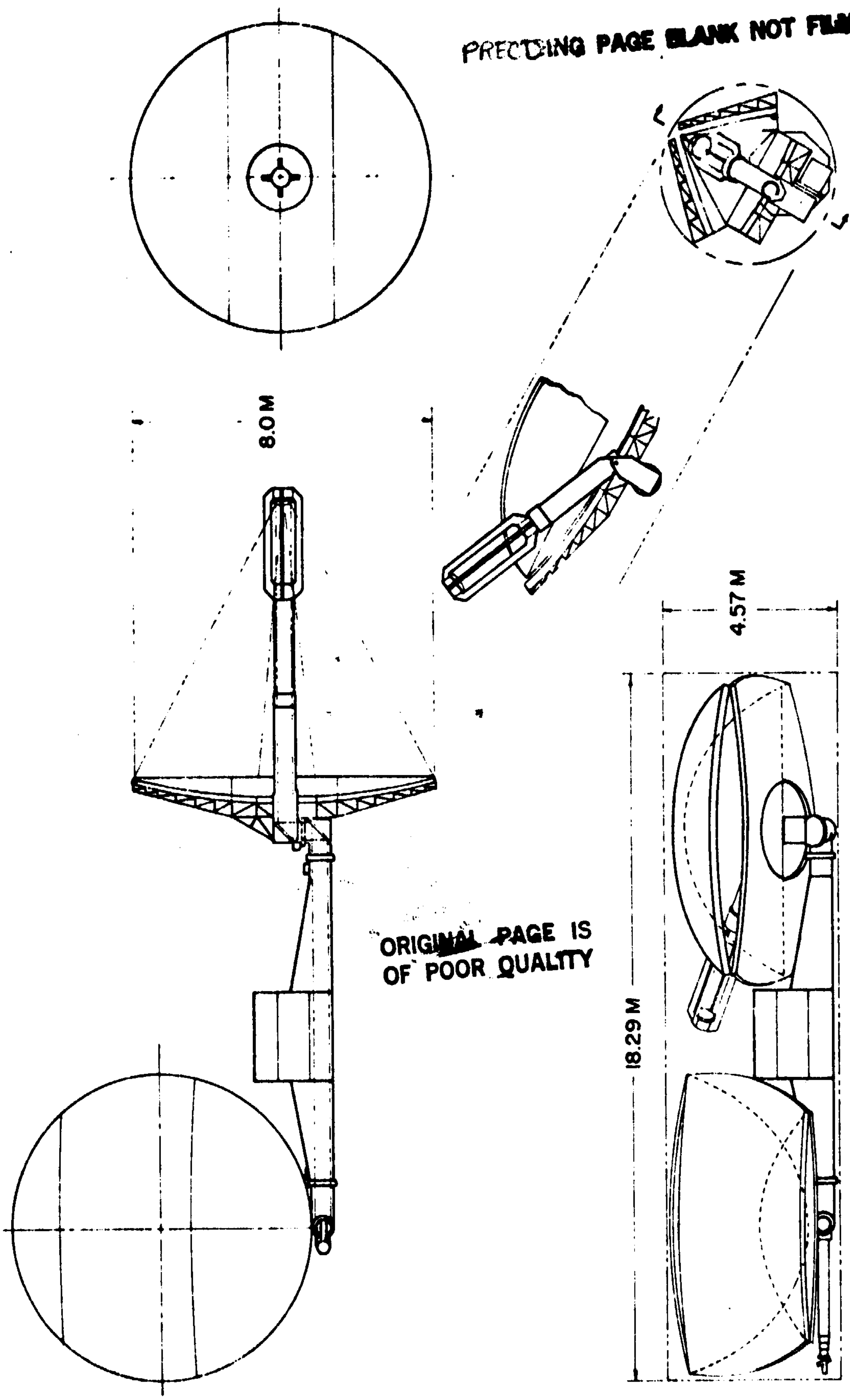

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Figure 41. Medium earth orbit relay unit

TABLE XXVII. WEIGHT STATEMENT FOR MEO RELAY UNIT

(Ground-Based Laser)

(6 Required)

Subsystem	Small Payload MERU		Large Payload MERU	
	kg	lbm	kg	lbm
Acquisition	107	235	107	235
Transmitter Beam Expander	1,896	4,179	2,331	5,140
Transmitter Optical Train	278	612	375	827
Gimbals and CMGs	1,154	2,544	1,338	2,950
Tracker	119	262	119	262
Ranger	29	63	29	63
Receiver Beam Expander	1,884	4,153	2,172	4,750
Receiver Optical Train	276	608	370	816
Gimbals and CMGs	964	2,125	1,158	2,553
Tracker	97	213	97	213
Astrionics	394	868	394	868
Electrical Power	182	401	182	401
Stabilization and Attitude Control	70	154	76	168
Total	7,450	16,417	8,748	19,286

Figure 42 shows the inboard profile of the GERU for the 37.5-MW laser rocket system. The difference between the 37.5-MW system GERU and the 1000-MW system GERU is the propellant tank size and radiators. The predominant radiators are the result of the higher laser power and the amount of energy to be expelled. The larger propellant tank is the result of the extra weight of the mirror cooling system. The space radiator has been omitted in Figure 42 for clarity. For both GERUs, the coolant is carried to LEO in a container and upon deployment is pumped into the cooling system. The integral propulsion system requires deployment at LEO so that the receiving aperture can receive and direct the energy into the thruster for self-propulsion to GEO. The integral propulsion provides a means of transport to GEO at laser propulsion efficiencies with minimum cost when compared to the alternative of a chemical system. The thrusters are the same as used in the respective propulsion units.

The receiving apertures are segmented, off-axis optical systems to receive an unobscured incoming beam. The aperture is adaptive to maintain figure control. This control for figure must be near diffraction-limited performance to avoid inducing additional wavefront errors which the transmitter optics must correct. The receiving aperture reduces the beam diameter and directs it to the secondary which redirects the energy through the optical train. The secondary is sized for 15 kW/cm^2 (96.8 kW/in.^{2}). The third transfer mirror as shown in Figure 42 is reflective on both sides and can direct the energy through the normal train to the transmitting aperture or rotating can direct the beam into the engine during propulsive maneuvers.

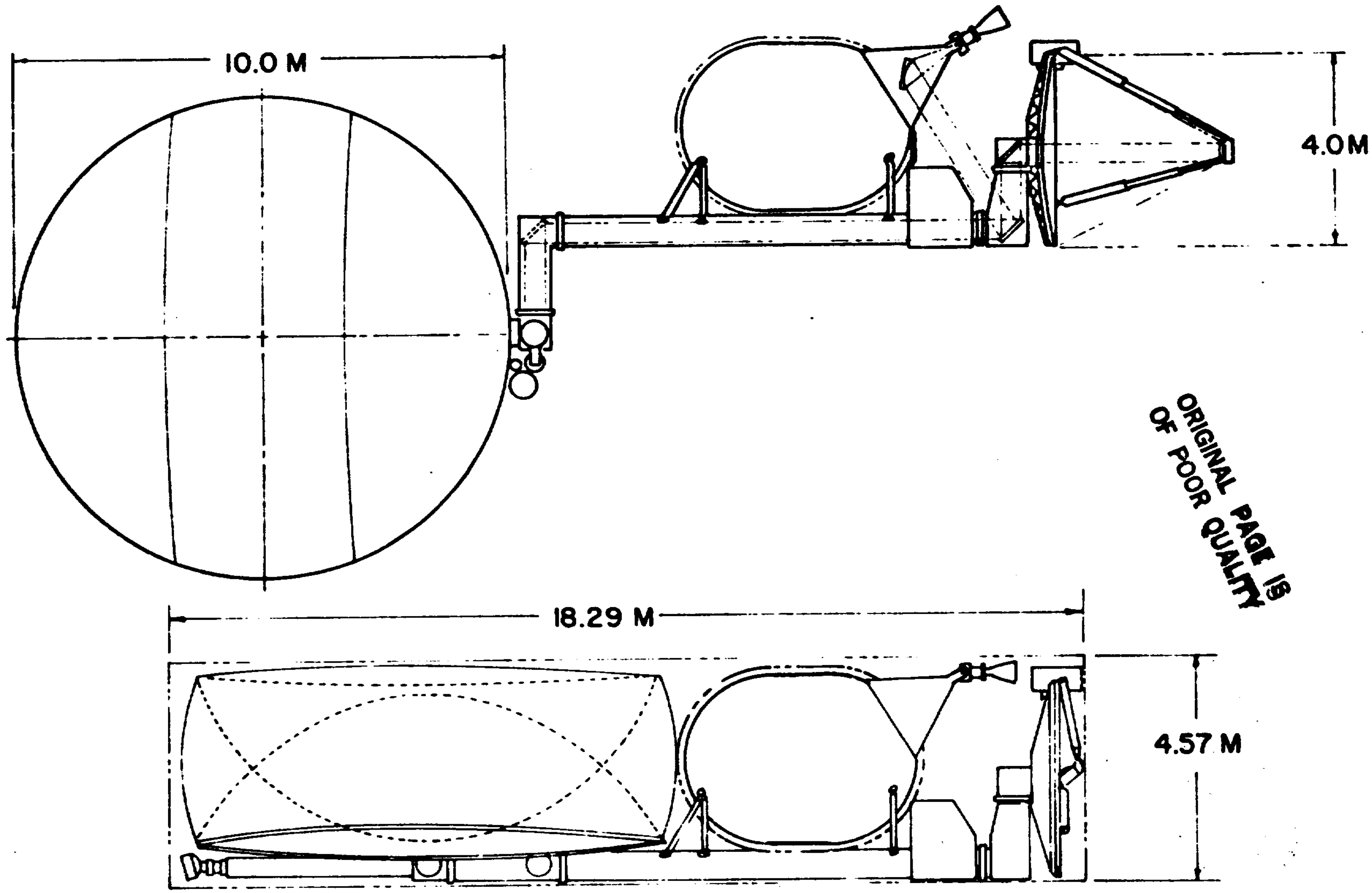


Figure 42. Ground system, synchronous-equatorial orbit relay unit

The transmitter aperture for both the 37.5- and 1000-MW systems is an adaptive, monolithic, Cassegrainian system. Cooling is required for both systems. The transmitter aperture is double gimbaled to provide greater than 2-π-sr pointing capability and, when coupled with the receiver, provides the capability of receiving and pointing from and to any direction.

Table XXVIII shows the weights by subsystem of both the 37.5- and 1000-MW system geosynchronous energy relay units.

TABLE XXVIII. WEIGHT STATEMENT FOR GEO RELAY UNIT

(Ground-Based Laser)

(1 Required)

Subsystem	Small Payload Relay		Large Payload Relay	
	kg	lbm	kg	lbm
Acquisition	107	235	107	235
Transmitter Beam Expander	602	1,327	7,662	16,894
Transmitter Optical Train	244	537	8,492	18,721
Gimbals and CMGs	367	809	4,397	9,694
Tracker	119	262	119	262
Ranger	29	63	29	63
Receiver Beam Expander	2,711	5,976	10,206	22,500
Receiver Optical Train	243	535	8,423	18,560
Gimbals and CMGs	1,388	3,059	3,572	7,875
Tracker	97	213	97	213
Astrionics	372	820	1,115	2,458
Electrical Power	144	317	170	374
Propulsion System, Dry	452	996	2,594	5,719
Propellant	2,428	5,352	17,359	38,270
Stabilization and Attitude Control	188	414	1,114	2,456
Structure	540	1,190	3,696	8,142
Total	10,031	22,105	69,149	152,446

3.7 TASK VII: CONCEPT EVALUATION AND COST, GROUND-BASED LASER

The ground-based laser rocket system concepts were postulated and their life-cycle costs compared to similarly performing chemical OTV systems. The ground-based laser rocket systems were sized to correspond to the space-based basic/small system and the SPS type/large system. (See section 3.4.)

The basic ground-based laser rocket system, capable of round-tripping 2268 kg (5000 lb) to geosynchronous orbit, consists of the following elements:

- Single 37.5-MW ground laser transmitter unit and its power source
- Single energy relay unit in geosynchronous orbit

- Six energy relay units at medium earth orbit altitude
- Multiple propulsion units in earth orbit (fleet size determined by a given mission model)

The basic system is supported by the shuttle transportation to LEO assuming the same load factors as used in the space-based system (90% load factor, 58,500 lb to LEO).

The large SPS-type, ground-based laser rocket system is capable of delivering 148,000 kg (326,000 lb) one way to geosynchronous orbit. This system is comprised of the following elements:

- Single 1.0 GW ground laser transmitter unit and its power source
- Single energy relay unit in geosynchronous orbit
- Six energy relay units at medium earth orbit altitude
- Multiple propulsion units in earth orbit as required by the fleet sizing analysis

This system is assumed to be supported by the independently developed heavy lift launch vehicle (427,500 lb to LEO capability).

Both of these ground-based laser transmitter systems require a ground site location, extensive electrical power generating plants, and powerplant fuel storage facilities. It was assumed that the laser transmitter could be located at an existing ground station (Maui) site in Hawaii and that only specialized facilities would be charged to this program. These facilities consist of: a building to house the laser and a tower to provide the optics pedestal; fuel supply storage farm (tanks sized for 500 hr of lasing); water storage and pumping facilities to provide optics cooling; and an electrical power-generating plant. In case of the basic system, 250-MWe powerplant is required and for the large system a 6.67 GW_e capability must be developed. The 20% laser efficiency plus other electrical requirements results in a 15% efficiency overall. Costs for these facilities were extrapolated from costs of smaller powerplants built by the commercial gas and electric power companies.

These two ground-based laser rocket system concepts require a ground crew to operate and maintain the laser transmitter. The costs of the crew and maintenance of the transmitter account for the majority of the laser operating costs, which also include the JP-4 fuel costs required for the powerplant. The above discussed facilities investment costs and operating costs are based on only one unit having to be built and without space qualification hardware requirements. Other groundrules and assumptions applied to the ground-based laser are the same as used in evaluating the space-based concepts. (See section 3.4.1.)

3.7.1 Laser Rocket System Costs

The ground-laser rocket system costs were generated in a similar and comparable manner as those for the space-based concepts. Table XXIX presents the development

TABLE XXIX. 37.5-MW GROUND SYSTEM DDT&E AND INITIAL DEPLOYMENT COST (FY '77 $M)

	Technology	Ground Laser	Propulsion Unit		1 GEO Relay		6 MEO Relay		Total[a]
			DDT&E	1st Unit	Transmitter	Receiver	Transmitter	Receiver	
Acquisition	$ 1.3	$ 5.904			$ 14.037		$ 5.904		
Tracker		2.485	$ 7.034	$ 1.657	2.485	$ 2.485	2.485	$ 2.485	
Ranger		1.079			4.090		1.079		
Beam Control	2.0	11.902			11.211		17.941		
Gimbals & CMGs		5.658	1.799	0.671	3.174	7.298	6.502	5.810	
Beam Expander	16.9	33.536	22.559	4.584	21.671	60.002	45.766	19.726	
Adaptive Mech.	4.3	8.798	6.344	2.072	5.974	14.865	11.900	4.755	
Optical Train	3.8	6.274	6.016	1.004	8.972	1.589	9.452	1.890	
Subtotal	$28.3	$ 75.636	$ 43.752	$ 9.988	$ 71.614	$86.239	$101.029	$34.666	
Spacecraft			$ 39.962	$13.923	$ 49.519		$ 37.928		
Propulsion			101.269	1.699	4.292				
Laser	8.6	120.294							
Power Supply		133.916							
Subsystems	$36.9	$329.846	$184.983	$25.610	$211.664		$173.623		$ 900.116
Facilities		4.500			22.500				27.000
Additional Units							442.259		442.259
Syst. Eng. & Intg.		49.477	27.747	4.098	31.750		26.043		135.017
System Test		19.791	7.399	1.152	8.467		6.945		42.602
System GSE		6.597	1.675		4.147		3.472		15.891
Launch OPS			4.264		10.030		60.257		74.551
Flight OPS			3.940		11.124		66.869		81.933
C^3 Mods		25.000							25.000
Shuttle Fee[b]			6.750		13.500		98.685		118.935
Data		16.408	9.200	0.926	11.987		13.039		50.634
Prog. Mgm't.		21.331	11.960	1.589	15.583		16.950		65.824
Total	$36.9	$472.950	$257.918	$33.375	$340.752		$908.142		$1,979.762

(a) Total excludes technology and propulsion unit fleet deployment, but includes the deployment of the rest of the system.
(b) In addition to shuttle fee, the costs of placing MEO relays (6) in orbit with the prototype propulsion unit are included.

and initial deployment costs for the 37.5-MW ground system and Table XXX shows comparable cost estimates for the 1.0-GW ground system. Technology development costs shown are not included in the totals.

The propulsion unit fleet investment costs were generated parametrically as a function of the fleet size and those are the same as presented in Figure 17 for both the small and the large space-based system. Also the basic 10-year operation and propulsion unit refurbishment costs are the same for the ground system as shown in Figure 21 for the space system. The fuel resupply costs are also the same, namely $3.537 million per mission for the small system and $4.138 M per mission for the large system.

The relay refurbishment costs are different for the ground-based system, because more relays are involved and the laser transmitter operations and maintenance costs rather than refurbishment costs are incurred. Table XXXI presents the one-time costs of relay refurbishment incurred at the end of the fifth operational year. Table XXXII presents the annual laser transmitter operations and maintenance costs. Not included in the annual laser transmitter operations costs are the mission dependent laser power fuel costs. These were estimated as a function of JP-4 consumption by the turbines and a 6.5¢/lb cost for JP-4. Utilizing an average of 16.7 hr for the round-trip mission to GEO with the small propulsion unit and 12.5 hr one way with the large unit, results in average laser power fuel costs per mission of $13,800 and $267,930, respectively.

3.7.2 Chemical OTV System Costs

The chemical OTV system costs for both the small and the large OTV are the same as presented in Figures 19 and 20 of section 3.4.3.

3.7.3 Sample Case Cost Comparison

A sample case of 1310 missions over a 10-year period was postulated. The mission composition consisted of 10 expendable and 1300 reusable missions. The number of propulsion units, OTVs and refurbishments required was the same as shown in Table XX in section 3.4. The resulting undiscounted LCC costs for this sample case are as shown in Table XXXIII. The cost ratios of chemical LCC/laser rocket system LCC are about 4.7 for both the small and the large systems. These are slightly lower than the ratios shown for the space-based laser rocket system which were slightly over 4.7 as shown in Table XX.

The comparison of the space-based LCC to the ground-based laser rocket system LCC for this sample case shows very little cost difference (1% to 3%) which is well within the cost estimating accuracy of these numbers. Therefore, based on the level of this analysis, no cost advantage can be ascribed to either the space- or the ground-based laser rocket system.

TABLE XXX. 1.0-GW GROUND SYSTEM DDT&E AND INITIAL DEPLOYMENT COST (FY '77 $M)

	Technology	Ground Laser	Propulsion Unit		1 GEO Relay		6 MEO Relay		Total[a]
			DDT&E	1st Unit	Transmitter	Receiver	Transmitter	Receiver	
Acquisition	$ 1.3	$ 5.904			$ 14.037		$ 5.904		
Tracker		2.485	$ 7.034	$ 1.657	2.485	$ 2.485	2.485	$ 2.485	
Ranger		1.079			4.090		1.079		
Beam Control	2.0	11.902			11.211		17.941		
Gimbals & CMGs		29.682	4.197	0.823	13.808	12.405	16.311	12.558	
Beam Expander	16.9	33.536	30.014	5.484	29.985	69.213	57.236	21.403	
Adaptive Mech.	4.3	8.798	6.715	2.206	5.974	14.865	11.900	4.755	
Optical Train	3.8	89.145	56.952	15.268	29.968	6.928	43.732	11.219	
Subtotal	$28.3	$ 182.531	$104.912	$25.438	$111.558	$105.896	$ 156.588	$52.420	
Spacecraft			$ 68.924	$19.280	$ 85.648		$ 83.059		
Propulsion			200.442	6.366	8.539				
Laser	8.6	532.429							
Power Supply		1,235.578							
Subsystems	$36.9	$1,950.538	$374.278	$51.084	$311.641		$ 292.067		$2,928.524
Facilities		57.000			22.500				79.500
Additional Units							607.795		607.795
Syst. Eng. & Intg.		292.581	56.142	7.663	46.746		43.810		439.279
System Test		85.824	14.971	2.043	12.466		11.683		124.944
System GSE		39.011	3.477		6.062		5.841		54.391
Launch OPS			7.624		12.551		79.131		99.306
Flight OPS			7.975		14.603		93.074		115.652
C^3 Mods		25.000							25.000
HLLV Fee[b]			3.724		2.205		31.310		37.239
Data		96.998	18.579	1.824	17.063		19.623		152.263
Prog. Mgm't.		126.098	24.152	3.131	22.182		27.262		199.694
Total	$36.9	$2,673.050	$510.922	$65.745	$468.019		$1,211.596		$4,863.587

(a) Total excludes technology and propulsion unit fleet deployment, but includes the deployment of the rest of the system.
(b) The HLLV fee includes the costs of placing MEO relays (6) in orbit with the propulsion unit.

TABLE XXXI. RELAY REFURBISHMENT COSTS (FY '77 $M)

	37.5 MW System		1.0 GW System	
	GEO Relay	MEO Relay	GEO Relay	MEO Relay
Hardware	$17.203 M	$103.394 M	$22.347 M	$142.095 M
Transportation	5.092	34.722	4.554	26.843
Launch Ops.	2.417	11.238	3.024	14.759
Flight Ops.	1.978	12.771	2.596	17.776
Data and Prog. Mgm't.	0.404	2.209	0.517	2.993
Total	$27.094 M	$164.334 M	$33.038 M	$204.466 M
System Total	$191.428 M		$237.504 M	

TABLE XXXII. LASER TRANSMITTER ANNUAL OPERATIONS AND MAINTENANCE COSTS (FY '77 $M)

	37.5 MW System	1.0 GW System
Crew Costs	$ 0.513 M	$ 0.769 M
Equipment Maintenance	5.770	24.721
Recurring Spares	3.462	14.833
Transportation	0.070	0.297
Facilities Maintenance	0.090	1.140
Prog. Mgm't. and Data	0.807	3.404
Annual Total	$10.712 M	$45.164 M

TABLE XXXIII. 1310 MISSION SAMPLE CASE COST COMPARISON
(FY '77 $M)

	37.5 MW	Chemical Small OTV	1.0 GW	Chemical Large OTV
DDT&E & Initial Deployment	$1,979.8 M	$ 473.0 M	$ 4,863.6 M	$ 992.0 M
Fleet Investment	1,095.8	2,828.2	1,860.2	4,736.3
Subtotal	$3,075.6	$ 3,301.2	$ 6,723.8	$ 5,728.3
Basic 10-yr Ops.	123.6	176.8	239.0	349.6
Relay Refurbishment	191.4		237.5	
Laser 10-yr Ops.	107.1		451.6	
Propulsion/OTV Refurb.	287.1	647.5	521.0	1,102.6
Fuel Resupply	4,456.6	34,544.2	5,213.9	56,855.0
Laser Power Fuel	18.1		350.9	
LCC Total	$8,259.5 M	$38,669.7 M	$13,737.7 M	$64,035.5 M
Cost Ratio: Chemical/Laser	4.68		4.66	
Cost/Mission:	$ 6.30 M	$ 29.52 M	$10.49 M	$ 48.88 M
Cost/Lb:	$1,261	$5,904	$32.17	$149.94
Round-Trip P/L (lb)	5000	5000	N.A.	N.A.
One-Way P/L (lb)	N.A.	N.A.	326,000	326,000

3.7.4 Cost Analysis, Ground-Based System (ECON Inc.)

The same groundrules and assumptions which were utilized in the discussion under the space-based system were employed in the ground-based systems analysis. The cases were compared on the basis of identical mission models and the cryogenic system was utilized as the baseline.

Case 3 (Table XXXIV) represents a mission model composed of all payloads in the 2268-kg or 5000-lb payload range. There is a total of 450 missions where the vehicle can be reused, and 10 outer planet missions which call for an expendable vehicle. The fleet was sized at 22 units for the cryogenic version and 16 for the laser rocket system. For this case, the smaller (37.5 MW) ground-based laser would be utilized. Shuttle costs were used for fuel transportation to low-earth orbit. It was assumed that the IOC date for this system would be 1990 and DDT&E would begin in 1984. All costs were discounted back to 1984. The discounted costs are: 5821.7 million dollars for the cryogenic system and 2590.8 for the laser rocket system.

Case 6 (Table XXXV) is composed of missions with large payloads (148,000 kg or 326,000 lb) sized to represent the solar power satellite segments. This case requires

TABLE XXXIV. LCC COST COMPARISON, CASE 3

Mission Composition:	450	5,000-lb P/Ls
	10	5,000-lb Expendable P/Ls
Number of OTVs =	Laser	Cryo
	16	22

LCC Costs (In Millions of Dollars)

Cryogenic System	Category	Ground-Based Laser Rocket System
473.00	DDT&E	1,249.25
694.43	Investment and Spares	442.99
0.0	Laser System Deployment	730.51
759.16	OTV Deployment and Ops.	295.77
254.21	Refurbs.	327.97
12,611.36	Fuel Resupply	1,627.02
0.0	Laser Ops.	113.47
14,792.16	Total Real Year LCC (FY '77 $M)	4,786.98
5,821.70	LCC Discounted to 1984	2,590.76
	Discounted Cost Ratio	2.25
	Discounted Cost Ratio w/o DDT&E	3.47

TABLE XXXV. LCC COST COMPARISON, CASE 6

Mission Composition:	4,500	326,000 lb P/Ls
	14	326,000 lb Expendable P/Ls
Number of OTVs =	Laser	Cryo
	87	160

LCC Costs (In Millions of Dollars)

Cryogenic System	Category	Ground-Based Laser Rocket System
992.0	DDT&E	3,992.20
7,787.86	Investment and Spares	3,736.53
0.0	Laser System Deployment	871.41
6,544.79	OTV Deployment and Ops.	1,304.22
3,491.46	Refurbs.	1,693.98
203,685.22	Fuel Resupply	18,678.93
0.0	Laser Ops.	1,902.98
222,501.33	Total Real Year LCC (FY '77 $M)	32,180.25
70,568.306	LCC Discounted to 1987	12,224.91
	Discounted Cost Ratio	5.77
	Discounted Cost Ratio w/o DDT&E	7.64

the performance of 4,500 reusable missions and 14 expendable missions. The fleet size for this case is 160 vehicles for the cryogenic system and 87 for the laser system. This mission model would require the 1.0 GW ground-based laser with an IOC date of 1995. A heavy-lift launch vehicle is assumed for fuel transportation to low-earth orbit. The discounted costs are: 70568.3 million dollars for the cryogenic case and 12224.9 million dollars for the laser system discounted back to 1987.

Cases 8 (Table XXXVI) and 11 (Table XXXVII) are mixed cases encompassing both large and small payloads. The IOC date is taken to be 1995 and the DDT&E costs are 10% larger than for strictly large-payload DDT&E to cover the additional costs of developing the two vehicle sizes. The mixed case also assumes that the fuel transportation to low-earth orbit will be performed by the heavy-lift launch vehicle.

Case 8 is a mission model calling for 4000 large payloads and 400 small payloads during the 10 years of operations in addition to 85 expendable missions. The costs discounted to 1987 are: $64599.4 M for the cryogenic system and $12551.0 M for the laser rocket system.

Case 11 is a mission model calling for 8000 large payloads and 439 small ones including 14 small expendable payloads. The discounted costs are: $124523.5 M for the cryogenic system and $18625.7 M for the laser rocket system.

3.8 TASK VIII: PARAMETRIC ANALYSIS, AIRBORNE LASERS

During the analyses for space and ground-based laser rocket systems, it became obvious that airborne laser rocket systems would not be competitive, if they could perform the mission model at all. The weight and volume constraints of aircraft precluded the use of laser powers and burn times that could provide a thrust-to-weight ratio which would result in acceptable round-trip times. The largest powered laser device capable of being installed in an L-1011 or DC-10 type aircraft is an open-cycle, HF chemical laser with about 15-MW power and 200-s burn time. In addition, the lower thrust-to-weight would require multiple lasers and additional propulsion units. The aircraft and laser fuel costs for a 10-year life cycle would be prophibitive.

Also, as may be noted in the ground system analysis, a 10-m-diameter aperture for a 0.5-μm wavelength was required. The 2.7-μm wavelength of the HF chemical device would not reduce the size requirements even though the aircraft would be flying above 6100 m (20,000-ft) altitude. With a 10-m aperture, there is no reasonable means of locating it inside the aircraft out of the air stream. In the air stream, the possibility of maintaining a near-diffraction-limited performance is very remote.

As a result of these preliminary findings, the airborne system analysis was not carried further.

TABLE XXXVI. LCC COST COMPARISON, CASE 8

Mission Composition:	4000		326,000 lb P/Ls	
	400		5,000 lb P/Ls	
	85		5,000 lb Expendable P/Ls	
Number of OTVs =	Laser		Cryo	
	67	Large	133	
	85	Small	85	

LCC Costs (In Millions of Dollars)

Cryogenic System	Category	Ground-Based Laser Rocket System
1,091.2	DDT&E	4,391.42
8,835.1	Investment and Spares	4,855.28
0.0	Laser System Deployment	871.41
7,970.96	OTV Deployment and Ops.	2,076.91
3,317.82	Refurbs.	1,945.09
182,207.45	Fuel Resupply	16,696.53
0.0	Laser Ops.	1,744.41
203,422.53	Total Real Year LCC (FY '77 $M)	32,581.05
64,599.43	LCC Discounted to 1987	12,551.00
	Discounted Cost Ratio	5.15
	Discounted Cost Ratio w/o DDT&E	6.98

TABLE XXXVII. LCC COST COMPARISON, CASE 11

Mission Composition:	8000		326,000 lb P/Ls	
	425		5,000 lb P/Ls	
	14		5,000 lb Expendable P/Ls	
Number of OTVs =	Laser		Cryo	
	19	Large	24	
	135	Small	270	

Cryogenic System	Category	Ground-Based Laser Rocket System
1,091.2	DDT&E	4,391.42
12,758.74	Investment and Spares	5,510.44
0.0	Laser System Deployment	871.41
12,075.31	OTV Deployment and Ops.	2,098.51
5,330.43	Refurbs.	2,691.69
362,059.55	Fuel Resupply	33,294.82
0.0	Laser Ops.	3,029.85
393,315.23	Total Real Year LCC (FY '77 $M)	51,828.14
124,523.49	LCC Discounted to 1987	18,625.74
	Discounted Cost Ratio	6.69
	Discounted Cost Ratio w/o DDT&E	8.12

3.9 TASK IX: CONCEPTUAL DESIGN, AIRBORNE LASER

As the parametric analysis was not completed, see section 3.8, no conceptual design for the Airborne Laser Rocket Systems was prepared.

3.10 TASK X: CONCEPT EVALUATION AND COSTS, AIRBORNE LASERS

The parametric analysis and the conceptual design for Airborne Laser Rocket Systems were not completed. Therefore, concept comparison and cost analysis was not completed.

3.11 TASK XI: REPORTS

This task covered the monthly reports and briefings for the Laser Rocket Systems Analysis Study including the rough and final report drafts.

3.12 TASK XII: ADVANCED SOLAR ARRAY

This task was added to the original contract for the purpose of evaluating the systems effect of using gallium-arsenide (GaAs)-type solar arrays at higher sun concentrations and operating at higher temperatures. The approach to accomplishing this task was to synthesize electrical power systems equivalent to the electrical power systems used in the Space Based Laser Systems for both the small and large payloads, to estimate the costs including the effects of the life cycle operation, and to establish new DDT&E and first-unit cost for the system performing the same mission models as previously described. Fleet sizing, not having been affected by the new power supply subsystem, would remain the same as previously described. New life cycle costs were estimated and compared to the previous estimates established for the silicon-type solar arrays. The results showed about a 5% decrease in the DDT&E and first-unit cost relative to the laser transmitter however, the primary savings resulted in the significantly reduced weights. The weights for both the small payload system and the large payload system were reduced about 50%, thereby, reducing the transportation cost to low earth orbit. The deployment costs were also reduced by about 50% which represented about $225 M savings for the small payload system and $375 M savings for the large payload system.

These savings, while significant, do not greatly affect the overall results when included with the 10-year life cycle costs. For example, the Case 3 Cost Comparison of 460 small payloads as shown in Table XXIII shows a discounted cost ratio of 2.37. The new system with a GaAs electrical power subsystem shows a discounted cost ratio of 2.53.

3.12.1 Conceptual Design of a New Electrical Power Subsystem

A literature search and analysis was conducted to determine the optimum concentration for use with the GaAs cells as well as the operating temperature. As shown in Figure 43, efficiencies of GaAs cells peak at about 500-suns concentration. At this

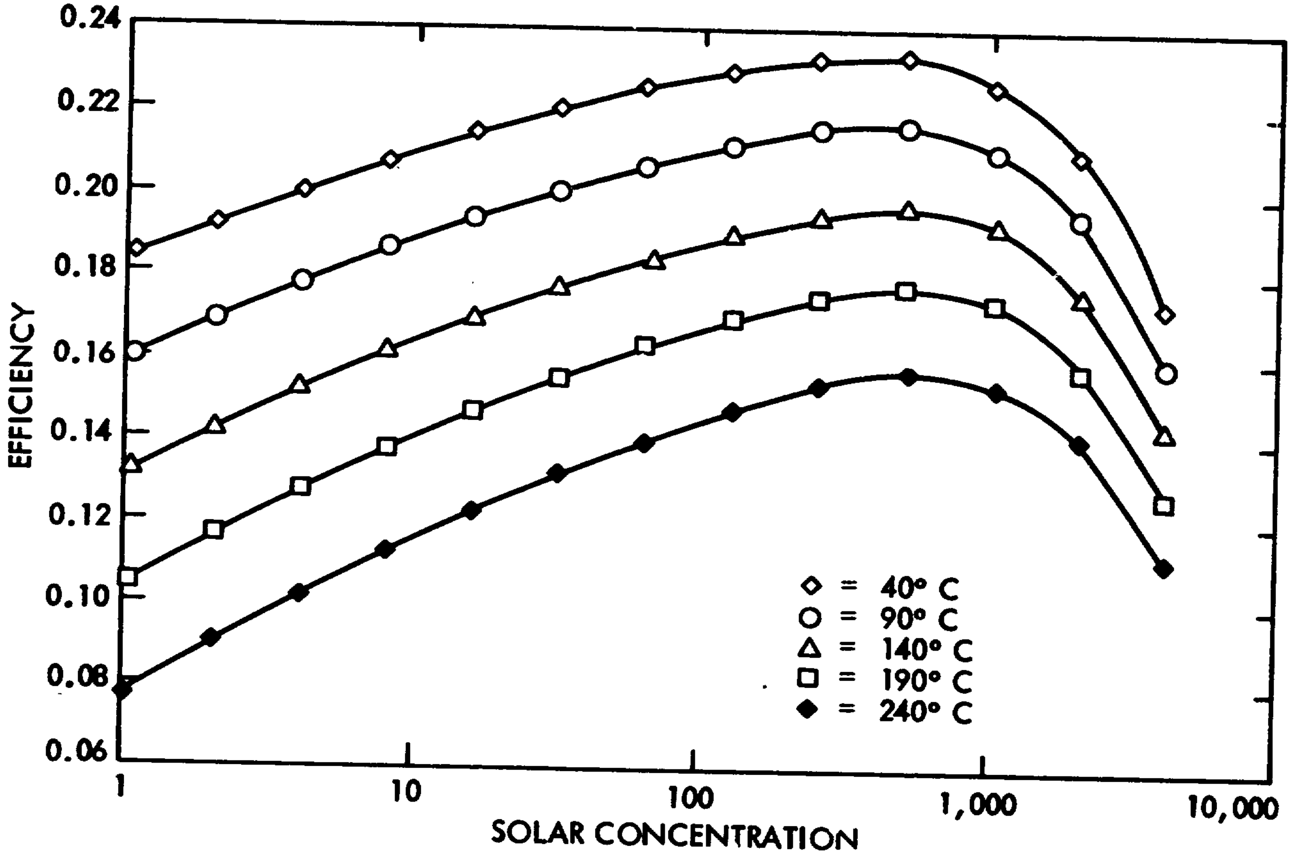


Figure 43. Efficiency versus concentration and temperature

concentration, better than 23% efficiency can be attained while operating at 40° C but would cause undue penalties because of the radiators required to operate at such a low temperature. The beginning-of-life efficiency of the GaAs cells operating at 190° C is approximately 18%, with an end-of-life efficiency just under 13%. Five hundred-suns concentration and a 190°C operating temperature was selected for the conceptual design of the new electrical power system. As shown on Figure 44, a 44-m-diameter solar array would be required to provide the 131 MW of electrical power for the small payload system. The collector size to provide the 500-sun concentration is 1,067-m diameter. The radiator area required to maintain the 190°C reflector is an aluminized Mylar or Kapton membrane constructed in conical shapes simulating a parabolic reflector concentrating the energy on the 44-m-diameter solar array. The radiator is used as a part of the structure to support the entire transmitter vehicle. The solar cells are mounted on a heat exchanger to carry the heat to the radiators. Table XXXVIII shows the GaAs power supply specifications for both the small and large payload systems. Table XXXIX shows the weight breakdown for the small and large payload systems, and it may be noted that even though the electrical power supply subsystem was reduced by approximately 50%, it still dominates the system weight.

3.12.2 Concept Evaluation and Cost, GaAs Solar Array

Costs were estimated for substituting the GaAs solar array on the laser transmitter unit; then the space-based rocket system economics were reevaluated. The costing assumptions and ground rules used were the same as stated in section 3.4.1.

The GaAs solar cell power supply costs were based on a Jet Propulsion Laboratory Report which projects $500 per kilowatt in 1986 and $250 per kilowatt in 1990 for the multicolor array. To keep the evaluation in FY 77 dollars, $470 per kilowatt was used for the larger 490-MW laser systems. The $250 per kilowatt was arrived at by assuming the same relative cost impact due to demand on the GaAs cells as was done on the silicon cells. (See section 3.4.2.)

Tables XL and XLI show the reestimated costs for the GaAs 16-MW and the 490-MW laser systems, respectively. Both these tables portray the DDT&E and initial deployment cost for the Space Based Laser Rocket System, excluding the propulsion unit deployment which is mission-dependent. As shown in these tables, the cost impact on the power supply is rather minor, about 5% decrease as compared to silicon cells. However, the major impact is due to the savings in deployment. GaAs power supply subsystems are considerably lighter which reduces the initial deployment costs either via the shuttle or the Heavy Lift Launch Vehicle (HLLV) and results in $225 and $375 million dollar savings to the 16-MW and 490-MW laser rocket systems, respectively. (See Tables XVIII and XIX for comparison.) The refurbishment costs of the laser transmitter unit were also adjusted in proportional fashion. All other laser system costs are the same as shown in section 3.4.2.

The same ground rules and assumptions specified in section 3.4 were employed in the GaAs array option cost analysis. The cases were compared on the basis of the same

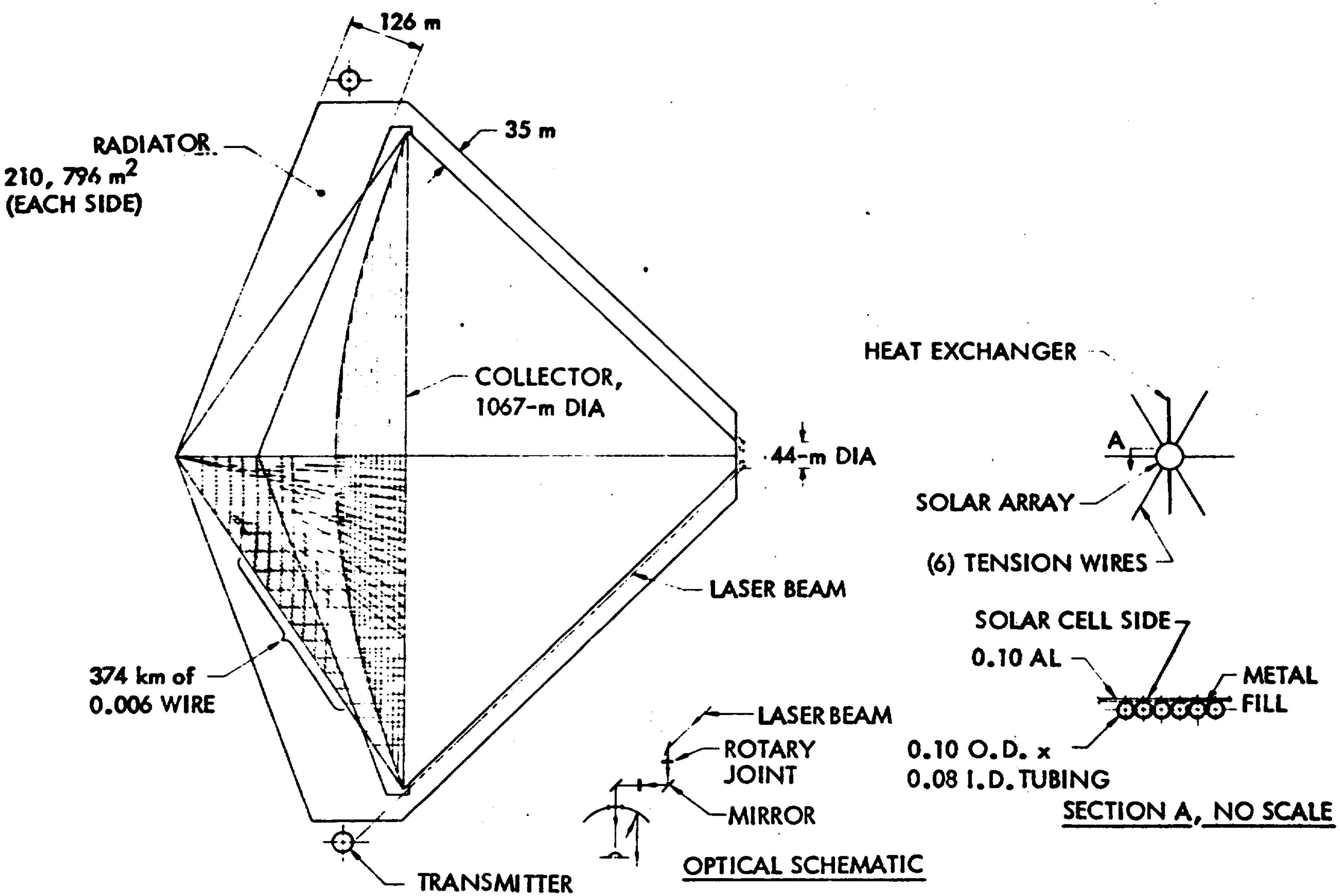


Figure 44. GaAs 131-MW electrical power supply

TABLE XXXVIII. GaAs POWER SUPPLY SPECIFICATIONS

	SMALL PAYLOAD	LARGE PAYLOAD
POWER REQUIRED (MW)	131	4,000
COLLECTOR EFFICIENCY (%)	85	85
SOLAR CELL OPERATING TEMP (°C)	190	190
SOLAR CELL EFFICIENCY (% END)	12.8	12.8
COLLECTOR DIAMETER (m)	1,067	5,870
SOLAR CONCENTRATION (SUNS)	500	500
SOLAR ARRAY DIAMETER (m)	44	239
HEAT EXCHANGER AREA (m^2)	210,769	6,026,800

TABLE XXXIX. WEIGHT STATEMENT FOR GaAs-POWERED LASER TRANSMITTER UNITS

	16-MW UNIT		490-MW UNIT	
	kg	LBm	kg	LBm
ACQUISITION (2)	58	128	58	128
TRACKER (2)	298	657	298	657
RANGER (2)	70	154	70	154
BEAM EXPANDER (2)	39,274	86,583	62,034	136,760
OPTICAL TRAIN (2)	1,376	3,034	27,040	59,612
GIMBALS AND CMGs	15,594	34,379	24,630	54,299
ASTRIONICS	272	600	1,059	2,319
FIRE CONTROL COMPUTER	194	428	194	428
SPACECRAFT ELECTRICAL POWER	2,124	4,683	6,866	15,127
LASER DEVICE	3,204	7,064	59,183	130,425
COMPRESSOR AND MOTOR	2,238	4,934	6,291	13,869
REFRIGERATION	9,238	20,366	177,650	391,647
LASER ELECTRICAL POWER SUPPLY	204,966	451,869	6,142,855	13,542,538
POWER CONDITIONING	6,668	14,700	188,643	415,882
STABILIZATION AND CONTROL	1,241	2,735	19,287	42,520
TOTALS	286,815	632,314	6,716,158	14,806,365

TABLE XL. 16-MW/GaAs ARRAY SPACE SYSTEM – DDT&E AND INITIAL DEPLOYMENT COST (FY '77 $M)

	Technology	Space Laser Transmitter	Propulsion Unit DDT&E	Propulsion Unit 1st Unit	Relay Transmitter	Relay Receiver	Total(a)
Acquisition	$ 1.3	$ 14.037	$ –	$	$ 5.904	$ –	
Tracker		19.933	7.034	1.657	5.873	2.485	
Ranger		4.090	–	–	1.079	–	
Beam Control	2.0	26.672	–	–	10.284	–	
Gimbals and CMGs		8.425	1.799	0.671	2.733	2.852	
Fire Control		14.856	–	–	–	–	
Beam Expander	26.3	165.629	22.559	4.584	15.588	52.424	
Adaptive Mechanisms	4.3	44.643	6.344	2.072	4.491	11.900	
Optical Train	3.8	12.353	6.016	1.004	8.485	3.448	
Subtotal	$37.7	$ 308.638	$ 43.752	$ 9.998	$54.437	$73.109	
Spacecraft		68.139	39.962	13.923	$ 49.263		
Propulsion		–	101.269	1.699	3.747		
Laser	8.6	146.296	–	–	–		
Power Supply		104.461	–	–	–		
Subsystems	$46.3	$ 627.534	$184.983	$25.610	$180.556		$ 993.073
Facilities		22.500	–	–	–		22.500
System Engr. and Intg.		94.130	27.747	4.098	27.083		148.960
Systems Test		25.101	7.399	1.152	7.222		39.722
Systems GSE		12.551	1.675	–	3.536		17.762
Launch Ops.		29.000	4.264	–	9.874		43.238
Flight Ops.		40.349	3.940	–	11.050		55.339
C^3 Mods		25.000	–	–	–		25.000
Shuttle Fee**		148.500	6.750	–	13.500		168.750
Data		34.047	9.200	0.926	9.577		52.824
Program Management		44.261	11.960	1.589	12.450		68.671
Total	$46.3	$1102.973	$257.918	$33.375	$274.948		$1635.839

(a) Total excludes technology and propulsion unit fleet deployment, but includes first prototype on-orbit deployment.

(b) Based on shuttle net payload to LEO of 58,500 lb (65,000 × 0.9 load factor) and $13.5 M/flight fee in 1976 dollars (65,000 × $208/lb).

TABLE XLI. 490-MW/GaAs ARRAY SPACE SYSTEM – DDT&E AND INITIAL DEPLOYMENT (FY '77 $M)

	Technology	Space Laser Transmitter	Propulsion Unit DDT&E	Propulsion Unit 1st Unit	Relay Transmitter	Relay Receiver	Total[(a)]
Acquisition	$ 1.3	$ 14.037	$	$	$ 5.904	$	$
Tracker		19.933	7.034	1.657	5.873	2.485	
Ranger		4.090	–	–	1.079	–	
Beam Control	2.C	26.672	–	–	10.284	–	
Gimbals and CMGs		12.915	4.197	0.823	7.416	5.595	
Fire Control		14.856	–	–	–	–	
Beam Expander	26.3	171.105	30.014	5.484	23.029	59.878	
Adaptive Mech.	4.3	44.643	6.715	2.206	4.491	11.900	
Optical Train	3.3	76.621	56.952	15.268	42.755	30.078	
Subtotal	$37.7	$ 384.872	$104.912	$25.438	$210.767		
Spacecraft		81.378	68.924	19.280	60.452		
Propulsion		–	200.442	6.366	10.264		
Laser	8.6	714.927	–	–	–		
Power Supply		1179.264	–	–	–		
Subsystems	$46.3	$2370.441	$374.278	$51.084	$281.483		$3026.202
Facilities		22.500	–	–	–		22.500
System Engr. and Int'g.		355.566	56.142	7.663	42.222		453.930
System Test		94.818	14.971	2.043	11.259		121.084
System GSE		47.409	3.477	–	5.424		56.310
Launch Ops.		184.580	7.624	–	15.647		207.851
Flight Ops.		261.493	7.975	–	19.082		288.555
C^3 Mods		25.000	–	–	–		25.000
HLLV Fee**		218.200	3.724	–	2.232		224.156
Data		133.472	18.579	1.824	15.005		167.056
Program Management		173.514	24.152	3.131	19.506		217.172
Total	$46.3	$3886.998	$510.922	$65.745	$411.860		$4809.780

(a) Total excludes technology and propulsion unit fleet deployment, but includes first prototype on-orbit deployment.

(b) Based on use of existing HLLV with net payload to LEO of 427,500 lb (450,000 × 0.95 load factor) and $6.3 M/flight fee in 1976 dollars (450,000 × $14/lb).

mission model and the cryogenic system was utilized for the baseline costs. The number of reuses and refurbishments remains unchanged from the silicon cell cases.

All earlier comments apply to the GaAs option as well as the following additions. The comments discuss the contrast between the GaAs option and the silicon cell method.

In terms of cost, the impact from the switch to the GaAs option is reflected by the decreased costs related to the laser transmitter. Specificially, these items are the DDT&E costs related to the laser development and procurement, the deployment costs, and the refurbishment costs. The amount of these reductions can be determined in Tables XLII through XLV which provide the cost breakdowns for the cryogenic system and the GaAs option for Cases 3, 6, 8, and 11.

Because 90% of the cost reductions occur before initial operating capability (IOC), the savings increase as a percentage of the cryogenic costs when the life-cycle costs are discounted, as shown in Table XLVI. While examining Table XLVI, it can be noted that Case 3 shows a higher savings than the other cases; the explanation is that Case 3 is not demanding (i.e., there are not as many flights per laser transmitter) as in other cases examined; thus, the laser transmitter represents a relatively high percentage of total life-cycle costs. Comparatively, Case 11 represents a highly demanding situation and the relative savings are less because the laser transmitter represents a smaller portion of total life-cycle costs.

From an economic viewpoint, the GaAs Option appears favorable if there are no overriding environmental or technology-risk questions; then the choice for gallium arsenide solar cells would be indicated. The GaAs Option would not provide any significant change to a choice between a Laser Rocket Propulsion System and a Cryogenic OTV.

3.13 TASK XIII: ELECTRIC PROPULSION COMPARISON

This task was added to the original contract with the purpose of synthesizing solar electric propulsion system concepts to perform the mission model developed in Task I for comparison to laser propulsion and conventional chemical propulsion systems. The study considered deployment and operational implications of the use of solar electric propulsion and was structured so the Solar Electric Transfer Vehicle (SETV) would not be unduly penalized (i.e., state-of-the-art projected to the late 1980's would be considered). Based on the work done in previous tasks, this task is limited to conceptual definition of two SETVs. A small SETV (SSETV) was conceived to transport a 2268-kg (5000-lbm) payload round-trip from low-earth orbit (LEO) to geosynchronous equatorial orbit (GEO). A large SETV (LSETV) was conceived to transport a 148,000-kg (326,300-lbm) payload from LEO to GEO and return empty. The two SETVs required electrical input power to the thrusters of 406 and 9430 kW_e, respectively.

The two conceptual designs are presented in Figures 45 and 46. Both concepts feature concentrating solar collectors, absorbers/receivers which include thermal storage material for full-orbit operation, Brayton rotating units for conversion of thermal energy to electrical energy, 100-cm thrusters, and cryogenically stored argon for the

TABLE XLII. LCC COST COMPARISON 3 (GALLIUM ARSENIDE OPTION)

MISSION COMPOSITION: 450 5,000 LB P/L's
10 5,000 LB EXPENDABLE P/L's

NUMBER OF OTV'S =

LASER		CRYO
16	SMALL	22

LCC COSTS (IN MILLIONS OF DOLLARS)

CRYOGENIC SYSTEM	CATEGORY	GALLIUM-ARSENIDE SPACE-BASED LASER ROCKET SYSTEM
473.00	DDT&E	1368.49
694.43	INVESTMENT & SPARES	442.99
0.0	LASER SYSTEM DEPLOYMENT	267.35
759.16	OTV DEPLOYMENT & OPS.	295.77
254.21	REFURBS	269.81
12611.36	FUEL RESUPPLY	1627.02
14792.16	TOTAL REAL YEAR LCC (CONSTANT YEAR)	4271.43
5821.70	TOTAL PRESENT VALUE COST (1984) (DISCOUNTED)	2297.93
	DISCOUNTED COST RATIO	2.53
	DISCOUNTED COST RATIO (CRYO/LASER)	4.93

TABLE XLIII. LCC COST COMPARISON 6 (GALLIUM ARSENIDE OPTION)

MISSION COMPOSITION: 4,500 450,000 LB P/L's
14 450,000 LB EXPENDABLE P/L's

NUMBER OF OTV'S =

LASER		CRYO
87	LARGE	160

LCC COSTS (IN MILLIONS OF DOLLARS)

CRYOGENIC SYSTEM	CATEGORY	GALLIUM ARSENIDE SPACE-BASED LASER ROCKET SYSTEM
992.0	DDT&E	4089.22
7787.86	INVESTMENT & SPARES	3736.53
0.0	LASER SYSTEM DEPLOYMENT	720.56
6544.79	OTV DEPLOYMENT & OPS	1304.22
3491.46	REFURBS	1974.22
203685.22	FUEL RESUPPLY	18678.93
222501.33	TOTAL REAL YEAR LCC (CONSTANT YEAR)	30503.68
70568.306	TOTAL PRESENT VALUE COST (1987) (DISCOUNTED)	11710.37
	DISCOUNTED COST RATIO (CRYO/LASER)	6.03
	DISCOUNTED COST RATIO W/O DDT&E	8.39

TABLE XLIV. LCC COST COMPARISON 8 (GALLIUM ARSENIDE OPTION)

MISSION COMPOSITION:	4000	450,000 LB P/L's
	400	5,000 LB P/L's
	85	5,000 LB P/L's

NUMBER OF OTV'S =

LASER		CRYO
67	LARGE	133
85	SMALL	85

LCC COSTS (IN MILLIONS OF DOLLARS)

CRYOGENIC SYSTEM	CATEGORY	GALLIUM ARSENIDE SPACE-BASED LASER ROCKET SYSTEM
1091.2	DDT&E	4498.14
8835.1	INVESTMENT & SPARES	4855.28
-0-	LASER SYSTEM DEPLOYMENT	720.56
7970.96	OTV DEPLOYMENT & OPS.	2076.91
3317.82	REFURBS	2225.33
182207.45	FUEL RESUPPLY	16696.53
203422.53	TOTAL REAL YEAR LCC (CONSTANT YEAR)	31072.75
64599.43	TOTAL PRESENT VALUE COST (1987 $) (DISCOUNTED)	12096.02
	DISCOUNTED COST RATIO (CRYO/LASER)	5.34
	DISCOUNTED COST RATIO W/O DDT&E	7.61

TABLE XLV. LCC COST COMPARISON 11 (GALLIUM ARSENIDE OPTION)

MISSION COMPOSITION:	8000	450,000 LB P/L's
	450	5,000 LB P/L's
	14	5,000 LB EXPENDABLE P/L's

NUMBER OF OTV'S =

LASER		CRYO
135	LARGE	270
19	SMALL	24

LCC COSTS (IN MILLIONS OF DOLLARS)

CRYOGENIC SYSTEM	CATEGORY	GALLIUM ARSENIDE SPACE-BASED LASER ROCKET SYSTEM
1041.20	DDT&E	4498.14
12758.74	INVESTMENT & SPARES	5510.44
0.0	LASER SYSTEM DEPLOYMENT	720.56
12075.31	OTV DEPLOYMENT & OPS	2098.50
5330.43	REFURBS	2971.93
362059.55	FUEL RESUPPLY	33234.82
393315.23	TOTAL REAL YEAR LCC (CONSTANT YEAR)	49034.39
124523.49	TOTAL PRESENT VALUE COST (1987) (DISCOUNTED)	17765.35
	DISCOUNTED COST RATIO (CRYO/LASER)	7.01
	DISCOUNTED COST RATIO W/O DDT&E	8.81

TABLE XLVI. LIFE-CYCLE COST COMPARISON BETWEEN GALLIUM ARSENIDE AND SILICON CELL LASER POWER SUPPLIES

CASE	CHANGE IN CONSTANT YEAR DOLLARS	CHANGE IN PRESENT VALUE DOLLARS
CASE 3	5.7 % LESS	6.4 % LESS
CASE 6	1.4 % LESS	2.0 % LESS
CASE 8	1.4 % LESS	2.0 % LESS
CASE 11	0.8 % LESS	1.4 % LESS

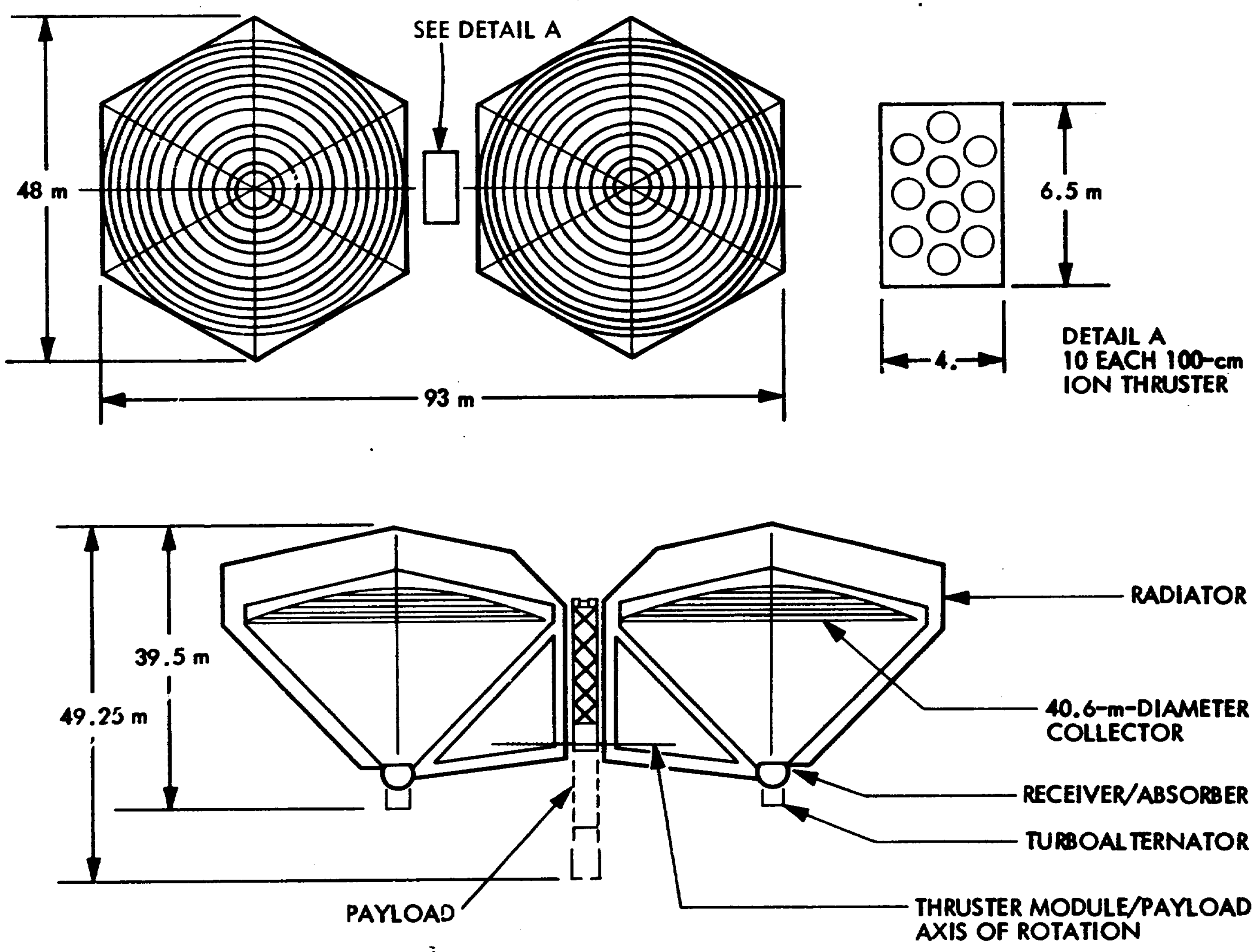


Figure 45. Small SETV conceptual configuration

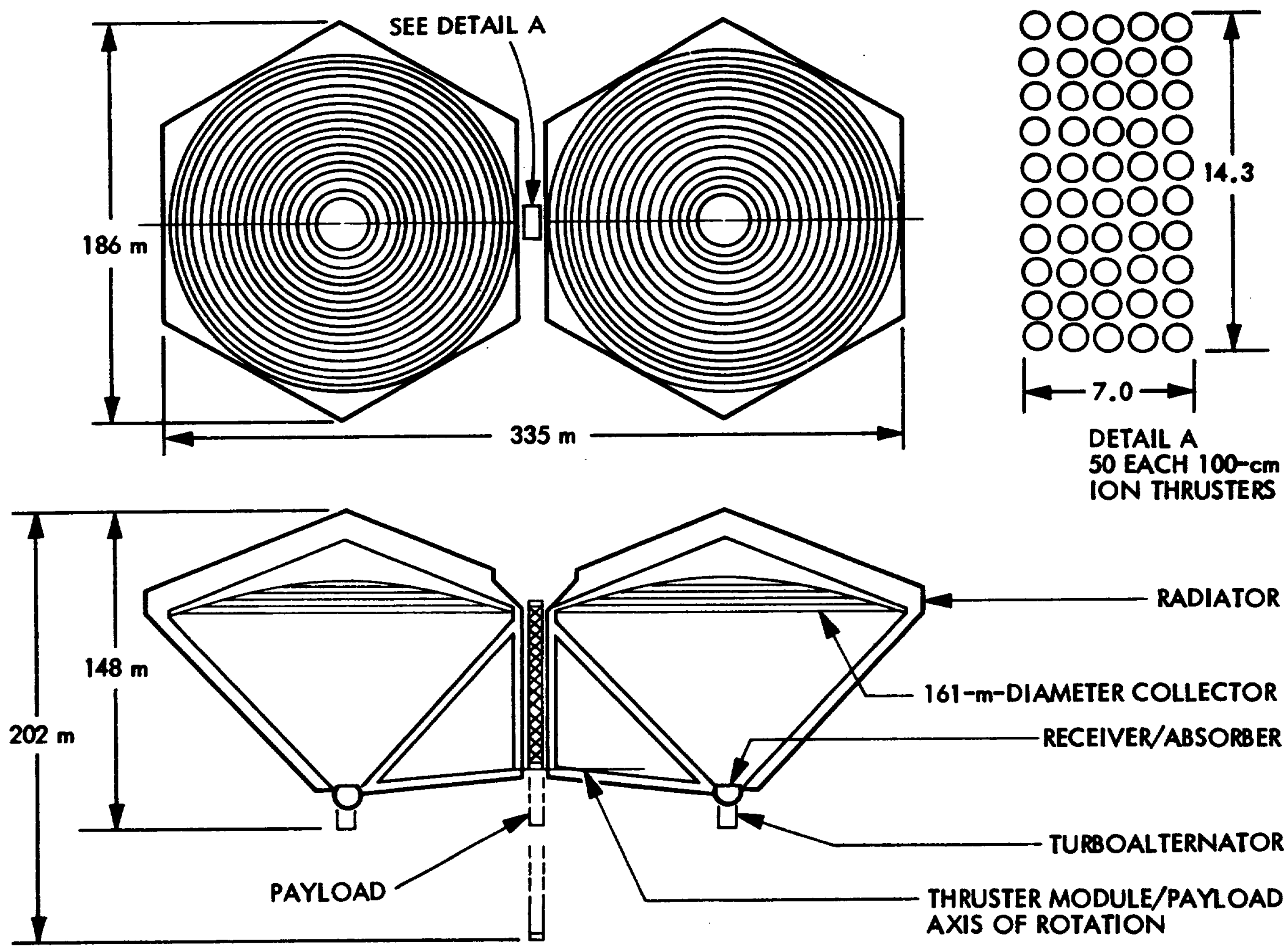


Figure 46. Large SETV conceptual configuration

propellant. Both designs feature a single rotation of the solar collectors for sun tracking. The axis is located at the predicted vehicle center of mass (CM) so thrusting is always through the CM. A truss structure is employed on both designs so ion thruster exhaust impingement on the concentrating collectors does not occur when thrusting parallel or near-parallel to the collectors. The SSETV and LSETV have predicted round-trip times of 220 and 161 days, respectively, for the baseline missions. These times include rendezvous and docking with payloads, servicing upon return to low-earth orbit, and periodic maintenance.

3.13.1 SETV Conceptual Design

The general approach to the task was to initially review available reports, papers, etc., to obtain reference data and establish a base for the solar electric propulsion vehicle design. Since the acronym SEPS has been identified with the Solar Electric Propulsion Stage using a 25-kW_e solar array, it was decided that the concepts defined for this task would be identified as Solar Electric Transfer Vehicles (SETVs). However, the SEPS was used as the base for the studies reported herein, and the SEPS bus, less solar array and electric propulsion subsystems, was used as defined in Reference 14 for support subsystems such as guidance, command and control, communication, data handling, and the hydrazine reaction control subsystems (RCS). The RCS was varied from the SEPS to accommodate increased hydrazine propellant associated with a heavier vehicle and rendezvous and docking associated with retrieval of payloads.

The second phase of the task evaluated the effect of the Van Allen belt degradation on solar cells, since the literature search has shown significant degradation effects. Because of this degradation, generation of electric power by dynamic power conversion, such as Rankine or Brayton cycle rotating machining was evaluated and compared to solar arrays.

The SSETV and LSETV conceptual designs were definitized by conducting trade studies in the areas of "sun-only" versus continuous operation of the ion thrusters, with limited efforts of using the power available to either increase thrust level (decreased trip time) or increase specific impulse (decrease propellant). Supporting analyses included computer simulations for thrust-to-weight ratios of 1×10^{-3} to 1×10^{-5} for LEO to GEO trips, Brayton and Rankine power cycle definition, propellant selection, and effect of sequential thrusting versus continuous optimal thrusting. Only ion thrusters were considered for this study. While magneto-plasma-dynamic (MPD) thrusters have some highly desirable attributes (e.g., low cost and simple system design such as a single power supply), due to a lack of a technology base, the MPD thruster was eliminated from consideration.

The definitized SSETV and LSETV concepts were then subjected to cost studies and predicted total life cycle costs were compared to those of the laser rocket propulsion system concept.

The assumptions used in this task are listed in Table XLVII.

TABLE XLVII. SOLAR ELECTRIC TRANSFER VEHICLE STUDY ASSUMPTIONS

(a) Two Basic Solar Electric Transfer Vehicles (SETVs)

(1) An SSETV to transfer a 2268-kg payload from LEO (300 km) with an inclination of 28.5° to GEO and return to LEO with the same payload mass, i.e., 2268 kg

(2) An LSETV to transfer a 148,000-kg payload from LEO to GEO and return the LSETV to LEO without a payload

(b) Both SETVs shall be self-contained space vehicles since they are multiple-use and must return to LEO (i.e., shall have complete electrical power and all subsystems required to rendezvous, dock, and transfer a "dumb" payload).

(c) Each SETV does the entire transfer mission without any other stages and only requires refueling and refurbishment as required before going on the next mission

(d) All subsystem equipments shall be 1990 state-of-the-art

(e) The electric propulsion subsystem shall be ion propulsion

(f) The SETVs are reusable for a 10-year orbital life

Solar Array Environmental Degradation

The initial evaluation of the SEPS (Ref. 14) and other documents resulted in a prediction that a SEPS or other type of solar electric space vehicle operating in earth orbit, using solar rrays for electric power generation, would suffer severe degradation due to radiation damage when traversing the Van Allen Belts.

Solar array (solar cell) power degradation as a function of transfer vehicle trips from LEO to GEO was reported on extensively in Reference 15 and is plotted in Figure 47. These data were confirmed by data presented in Reference 15 and by in-house LMSC evaluations. The results of the latter two efforts are shown in Figure 48. All results verify that extensive degradation will occur with conventional silicon solar cells. It can be seen that for a reusable electric propulsion transfer vehicle operating from LEO (300 km) that solar cell degradation will approach 70%. Figure 47 also illustrates that even if the SETV were operated from a minimum altitude of 13,000 km that solar cell degradation would approach 40%. LMSC in-house studies show (see Table SLVIII) that an advanced silicon solar cell design can reduce radiation damage from the Van Allen Belts slightly but it is still significant, i.e., approximately 41% for one pass through the belts. The table also shows that the use of a GaAs solar cell would not be advisable for a reusable solar electric propulsion transfer vehicle. GaAs cell efficiency is higher than that of a silicon solar cell and as noted in Table XLVIII, the cell

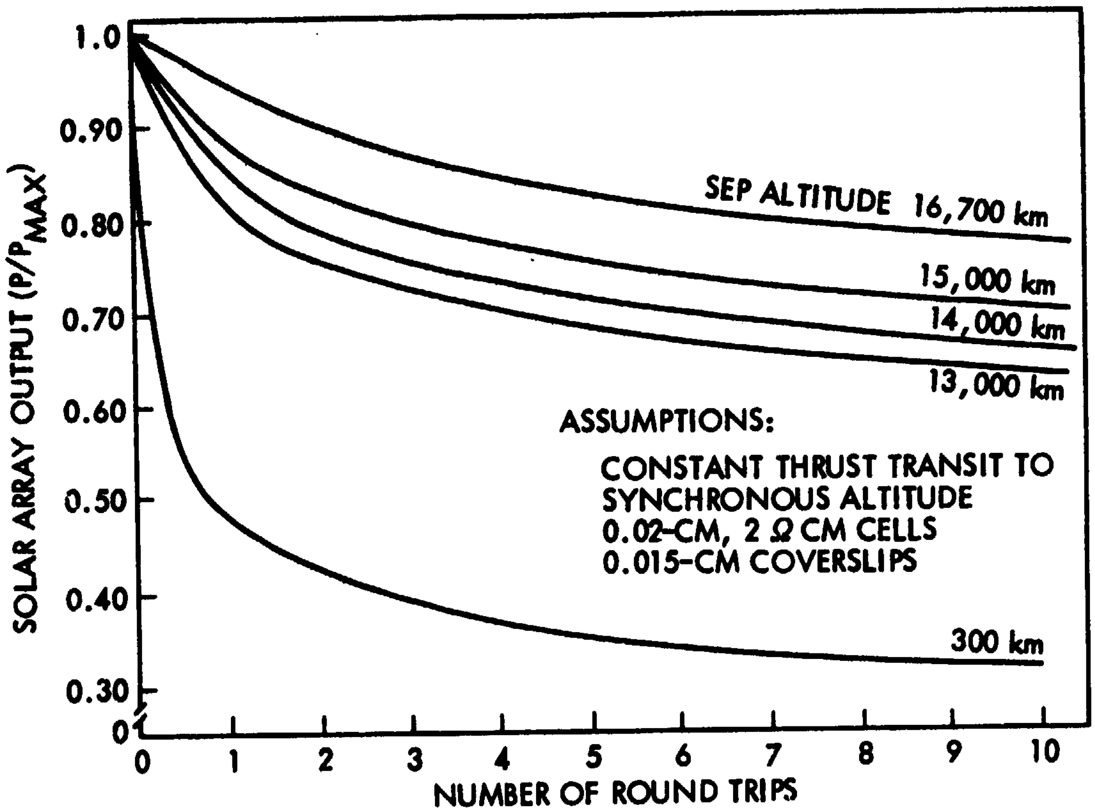


Figure 47. S/A degradation with conventional cells

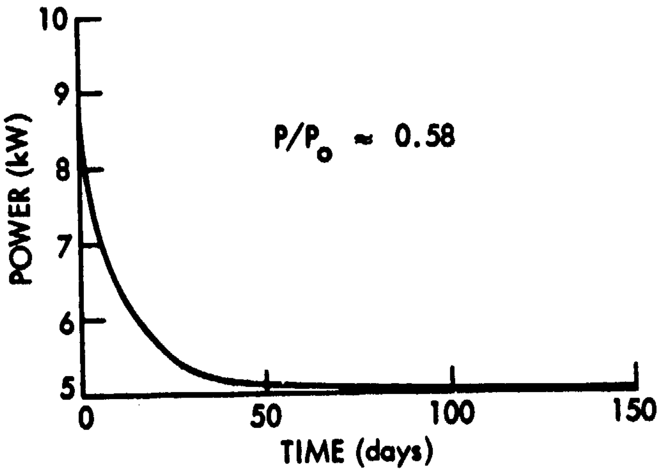


Figure 48. Typical power degradations for spiral ascent

TABLE XLVIII. SOLAR CELL COMPARISON

		Conventional Silicon No Concentration	Advanced Silicon No Concentration	GaAs 450 Suns Concentration
Operating Temperature (°C)		55	45	200
GEO Operation				
LEO to GEO (80 Days)	$\frac{P_{EOL}}{P_{BOL}}$	0.524	0.590	0.339
7 Years at GEO	$\frac{P_{EOL}}{P_{BOL}}$	0.653	0.723	0.448
EOL at GEO	$\frac{P_{EOL}}{P_{BOL}}$	0.500	0.574	0.324
EOL Specific Power (W/cm^2)		0.0078	0.0089	3.160

can be operated at higher temperatures with high solar concentrations which increase power density significantly. However, significantly greater radiation damage occurs in traversing the Van Allen Belts.

The possibility of annealing the solar array cells after each trip was considered and rejected based on the following:

(a) Single-pass degradation is severe and in the order of 45%, thus to maintain power and minimize trip times, the solar array would have to be approximately twice as large as that required for a given power output at the beginning of life.

(b) Solar array extension in orbit would be required as degradation occurred in order to match power output to power required and minimize heat buildup due to excess power.

(c) While annealing will supposedly result in recovery of approximately 50% of the degradation, multiple annealing would be required for a reusable vehicle and solar cell "experts" are not sure this would be effective for multiple restoration of the degradation loss. In fact, the "experts" are not in agreement that annealing is practical and will restore the loss.

Electrical Power Subsystem (EPS) Selection

Based on the negative factors associated with solar cell power generation, dynamic power conversion systems were evaluated for the SETV. Both closed-loop Brayton and Rankine cycles were considered. The Brayton cycle was selected to be studied in this task based on higher potential efficiency and comparable weights. Using the Brayton cycle data available at LMSC from recent laser power conversion studies and the data from Reference 16 on other components of dynamic electric power conversion systems, parametric electric power system (EPS) mass as a function of power level output were calculated and are presented in Figure 49. Also plotted therein is EPS mass for solar arrays considering no degradation and degradation as predicted in Reference 14 for 10 LEO/GEO round trips. (See Figure 47.) However, the curves are not directly comparable as the Brayton cycle data are for continuous power output even though the space vehicle is periodically in the Earth's shadow. That is, the Brayton EPS includes a thermal storage phase-change material and collectors are sized for the extra energy collection during sunlight periods. If the solar array EPS has battery storage added for full-orbit operation, then the EPS mass will increase by 3,800 to 10,000 kg (8,377 to 22,046 lb) for power levels of interest (i.e., $\sim$400 kW_e). Additionally, after the above parametric data were generated, a lightweight concentrating collector design concept generated by LMSC was substituted for the aluminum collector used in Reference 16 and a new EPS mass was calculated for a Brayton cycle electric generation system of 200 kW_e output. This reduced the EPS mass for a continuous operating Brayton cycle machine to less than a solar array system with no radiation degradtion or battery storage. The solar array and Brayton cycle power generation concepts were traded off and the results are presented in Table XLIX. The Brayton cycle concept has many advantages over the solar array approach. Additionally, a mass comparison at 406-kW_e output (SSETV selected output) was made and the results are presented in Table L.

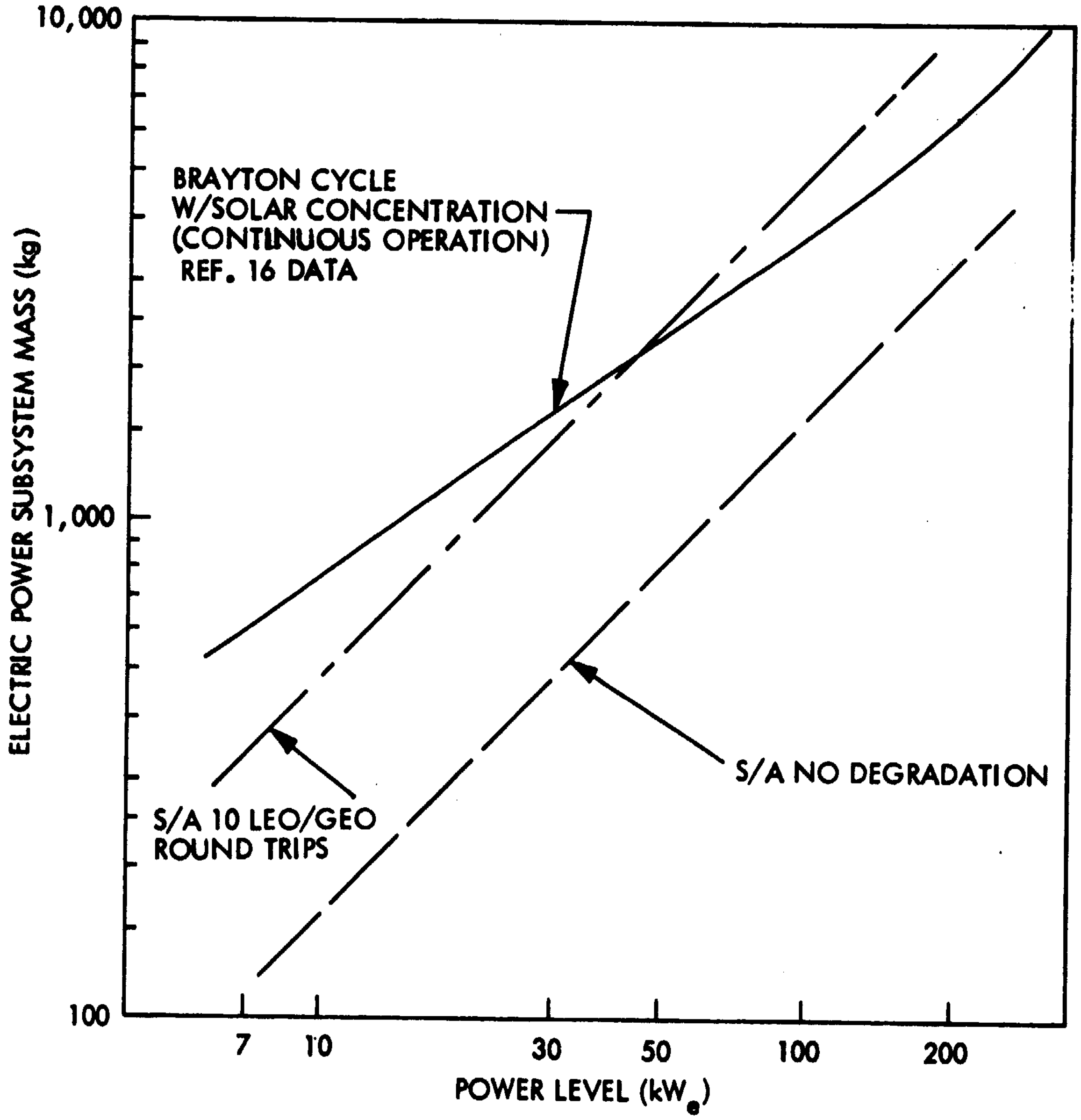


Figure 49. Electric power subsystem mass characteristics

TABLE XLIX. EPS CONCEPT COMPARISON (200 kW_e)

	Solar Array	Dynamic
Van Allen Belt Degradation	~ 70% for 10 round trips	None
Subsystem Mass	Less – If no degradation and sun only operation	Less – Above 40 kW_e for continuous operations
Voltage Level	28 V	1000/2000 V
On-Orbit Deployment	Simple foldout	Requires on-orbit assembly due to radiator size
Efficiency	~ 11%	23% or greater
Life	Seven to ten years demonstrated	Approximately 4 years demonstrated for space application (10.5 kW_e)

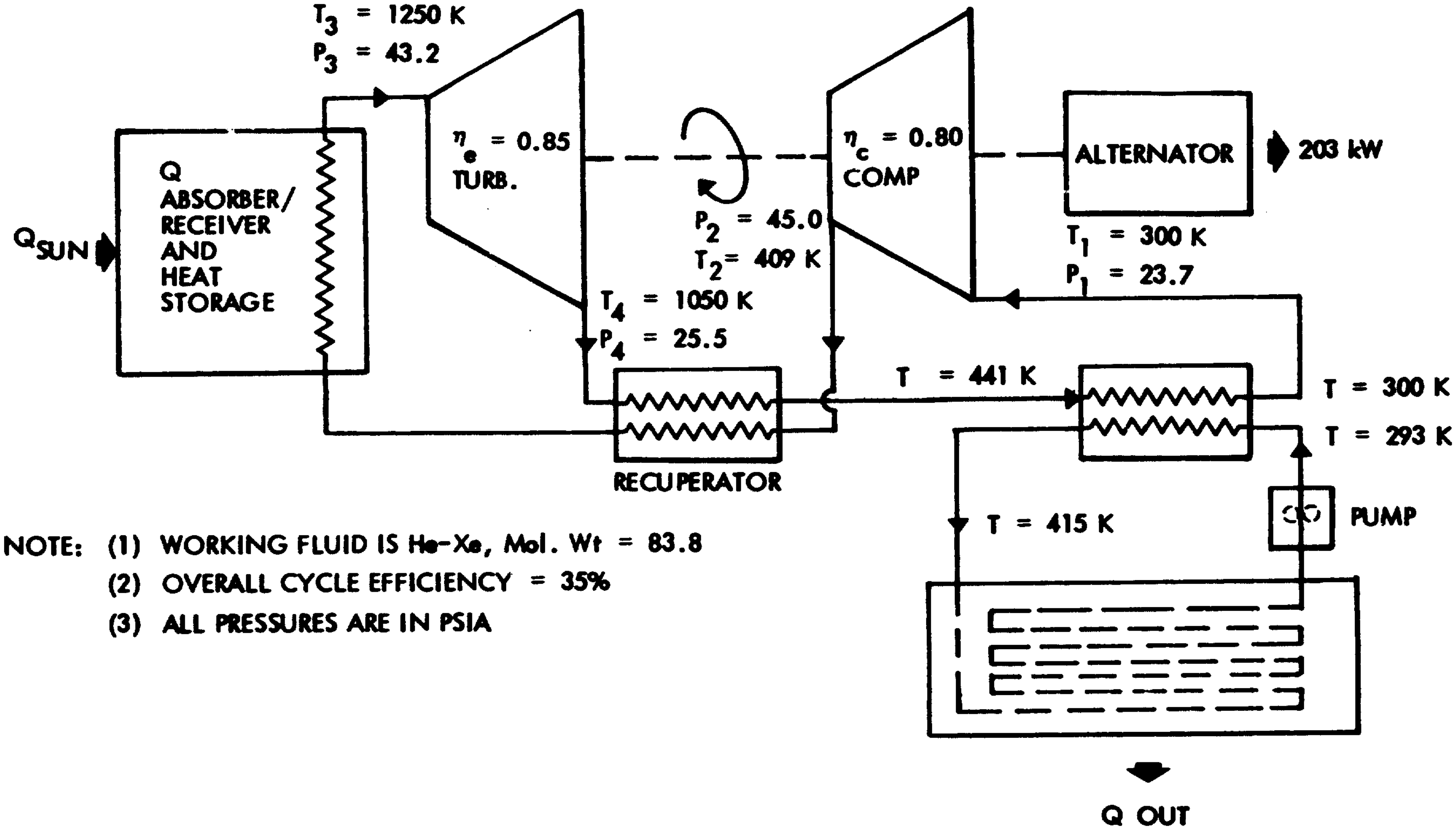
TABLE L. SSETV ELECTRIC POWER SYSTEM MASS COMPARISON (406 kW_e)

	Full Sun Only Operation (kg)	Continuous Operation (kg)
S/A – No Degradation	6500	12,750
S/A – 70% Degradation	9600	13,410
Brayton – Al Collectors	8790	11,520
Brayton – Mylar Collectors	2240	5,090

Note: The solar array concepts assume use of metal sulfide batteries at a storage capacity of 120 W-hr/kg. If $Ni-H_2$ batteries are used, the masses shown would increase by a factor of greater than 2.

Base on the above comparison, the Brayton cycle concept was selected for the SETVs. The significant decrease in mass of the Brayton cycle meant shorter trip times, less propellant, less burn time on the thrusters, and resulted in overall lower costs.

The Brayton cycle selected for the SSETV is shown schematically in Figure 50. Detailed studies of the cycle were not conducted and cycle efficiency, specific mass, and typical pressures and temperatures were determined from data generated in the Laser Power Conversion Analysis Study conducted by LMSC for NASA LeRC. As can



NOTE: (1) WORKING FLUID IS He-Xe, Mol. Wt = 83.8
(2) OVERALL CYCLE EFFICIENCY = 35%
(3) ALL PRESSURES ARE IN PSIA

Figure 50. SSETV Brayton cycle electrical power system

be noted on Figure 50, the power output is 203 kW_e for each Brayton EPS. A vehicle concept of two completely independent Brayton EPS (406-kW_e total power) was selected to increase mission reliability and to provide power generation redundancy. The selection of the power level for both the SSETV and the LSRTV will be discussed subsequently.

The higher power level required for the LSETV (~ 5 MW_e) resulted in evaluation of both Brayton and Rankine cycle electric power conversion systems. Cycles which could increase overall cycle efficiency to 50% or greater were studied in order to reduce overall EPS mass. Using the results LMSC had generated on the Laser Power Conversion Analysis Study, an advanced power system was considered. However, after preliminary evaluations were conducted, the Brayton cycle was selected and is shown schematically in Figure 51. The cycle features the energy exchanger and three-stage compression with intercooling.

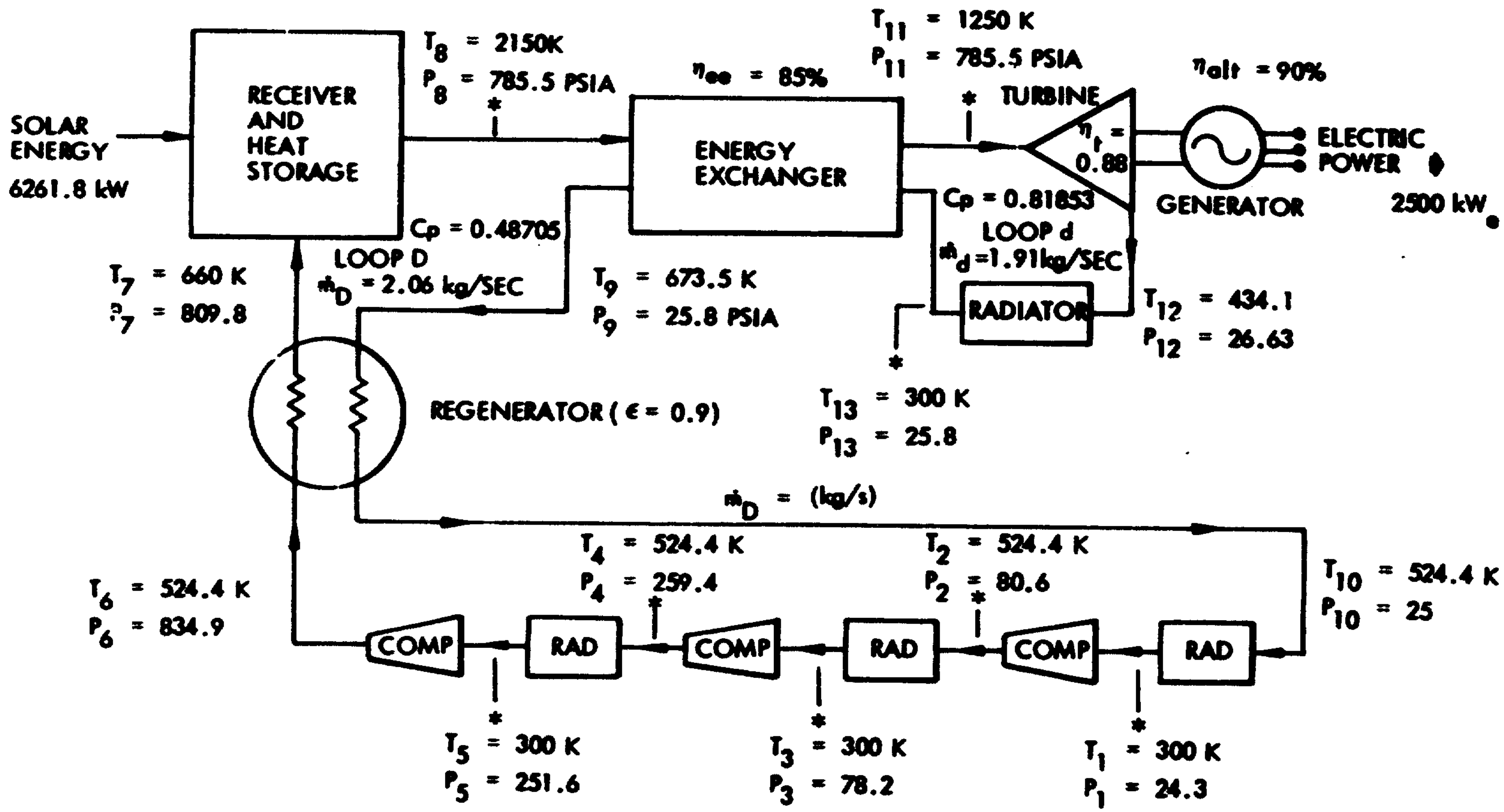
The resultant overall cycle efficiency was calculated to be 50.4%. The Brayton cycle is once again state-of-the-art and features no undue risk technologies. The energy exchanger concept would require technology development before it can be committed to use at the power level shown. The EPS for the SSETV and LSETV feature phase change material in the absorber/receiver so the EPS can be operated continuously even though the SETVs pass in and out of the earth's shadow at the beginning of the mission. Both the SSETV and LSETV concentrating collectors are sized to not only supply heat energy directly to the Brayton working fluid, but are oversized so the phase change materials can be melted during the sun-bathing period. Based on worst-case studies, the collectors were sized for a 62.5% sun to a 37.5% shade orbit as it was assumed that the initial parking orbit would be at 300 km. The solar insolation assumed was 1.353 kW/m^2 and for the power output levels of interest, i.e., 200 to 2500 kW_e, 85% efficiency each for the collectors and for the absorber/receiver was assumed.

This resulted in a power/efficiency budget for the EPS for the SSETV and the LSETV as shown in Table LI.

TABLE LI. SETV POWER/EFFICIENCY BUDGET

	Efficiency (%)	Power Per Module (kW) SSETV	Power Per Module (kW) LSETV
In-Orbit Energy to	85	1752	27,500
Energy Receiver/Absorber to	85	1490	23,375
Heat Energy to	37.5	1265	19,890
Energy Store to	62.5	433	7,460
Working Fluid to	35/50.4*	721*	11,300*
Shaft Power to	0.90/0.92**	152	5,690
Alternate Output		203*	4,715*

*Includes thermal or electrical losses as appropriate.
**Denotes increased efficiency for the LSETV in these two categories.



NOTE: (1) ALL PRESSURES ARE IN PSIA
(2) LOOP D FLUID IS He-Xe, Mol. Wt = 83.3
(3) LOOP d FLUID IS He-Xe, Mol. Wt = 43.72

$$\text{OVERALL } \eta = \frac{2500 \text{ kW}_e}{6261.8} = 0.399$$

Figure 51. LSETV Brayton electrical power subsystem

The phase change material for the EPS for the SSETV is silicon and for the LSETV it is 3BeO-2 MgO. These materials result in melting-point temperatures which are compatible with that required for the respective thermal input for the Brayton cycle of each vehicle.

The selected phase change materials also have a relatively high equivalent thermal storage capability (Figure 52).

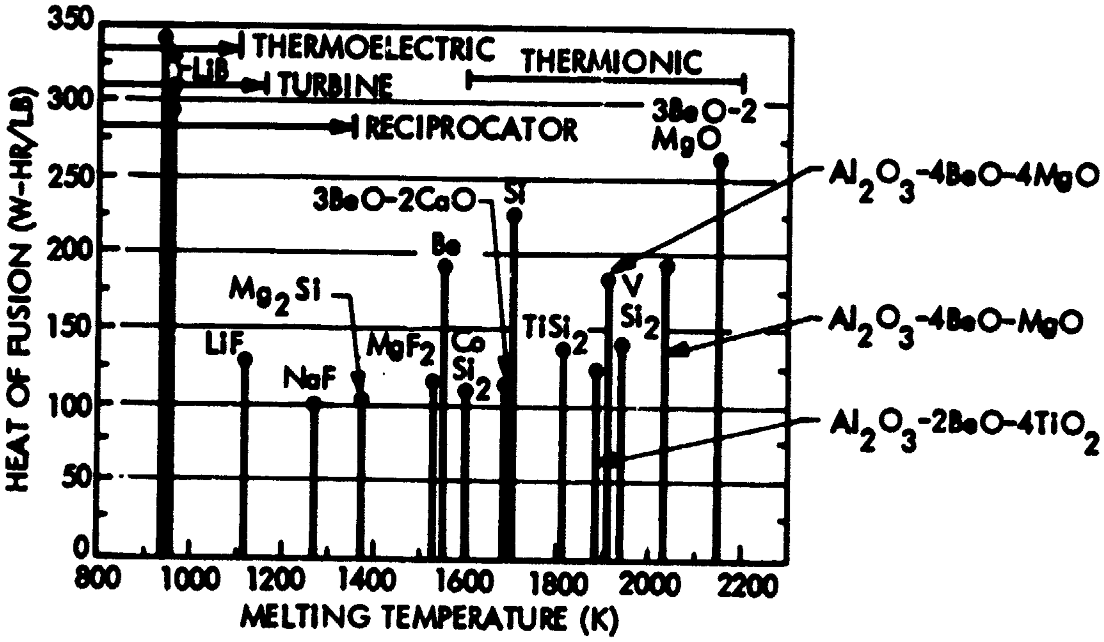


Figure 52. Candidate thermal storage materials (Ref. 4)

Each absorber/receiver for the SSETV has a total mass of 1470 kg of which 772 kg is silicon. The proposed concept is shown in Figure 53.

Each absorber/receiver for the LSETV has a total mass of 27,000 kg of which 13,000 kg is 3BeO-2 MgO. Because of the high operating temperature, the mass estimate assumed use of tungsten-rhenium for the gas/heat storage media tubes with thermal insulation between the shell wall and the tubes. The gas/heat storage media tubes in absorber/receiver of the SSETV assumed use of columbium as the material.

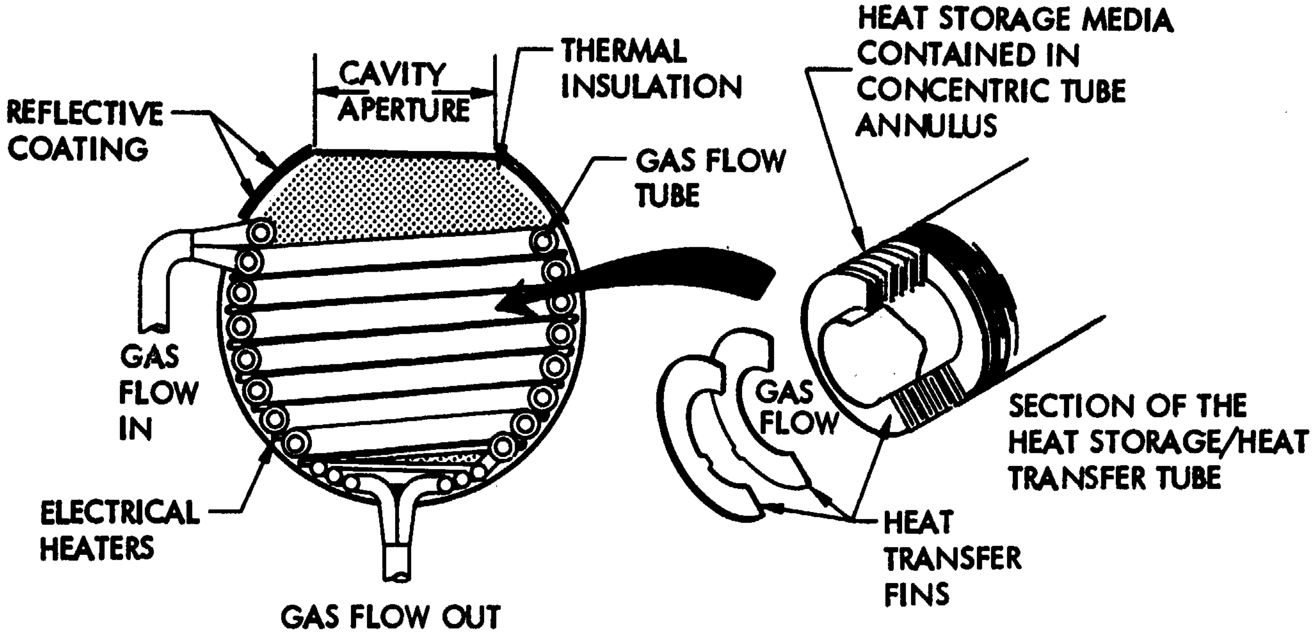
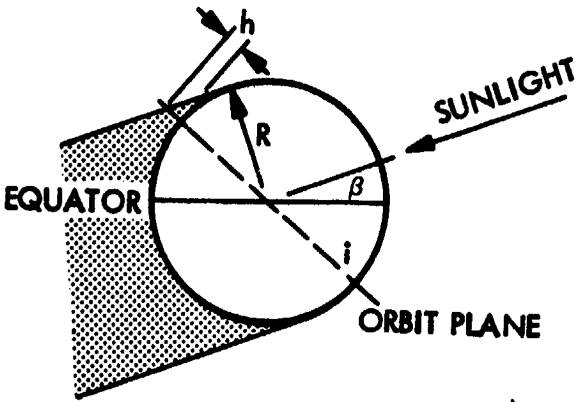


Figure 53. Typical absorber/receiver design concept

Continuous Operation Versus Sun Only Operation

A study was conducted to select the method of operation of the ion thrusters, i.e., should they be operated continuously or should they be operated only in sunlight, thus minimizing the EPS size and deleting any requirement for energy storage for the electric propulsion subsystem. To operate the SETVs only in the sunlight, the thrusters and EPS have to be either shut down or else initial operating altitude must be increased so no earth-shadowing occurs. Using the relationships shown in Figure 54, and the equation thereon, sunlit orbital altitudes were calcuated for various inclinations, as follows:

Inclination (deg)	Sunlit Altitude (km) Solstice	Equinox
28.5	1716	6989
40	749	1344
55	131	1408



$$h = R\left(\frac{1}{\sin(i + \beta)} - 1\right)$$

Figure 54. Sun/shade/Earth model (Ref. 18)

It can be seen that only at the higher inclinations are reasonable altitudes achieved. In fact if "anytime" departure is desired, a chemical transfer stage would be required to place the SETV at the required altitude. It was therefore concluded that the SETVs would be deployed at 300 km and the effect of occultations would be assumed.

Assuming constant thrusting, the time to climb from 150 to 35,000 km was calculated for a range of thrust-to-weight (T/W) ratios from 1×10^{-3} to 1×10^{-5} and ranges of interest are shown in Figure 55. Using these data and assuming 60% of the orbital time is available for thrusting, it was estimated that approximately 310 occulations would occur at an initial T/W of 1×10^{-4}. Subsequently, a more sophisticated analysis was obtained from Reference 19. Calculations from these data, (Figure 56), show that for the probable trip times (~ 110 days each way for the SSETV) that approximately 500 occultations would occur for an anytime departure capability. In fact, the Reference 19 analysis assumed constant thrusting by use of chemical propulsion during the shade periods; therefore, the actual occultations would be greater than 500.

Since the SETVs are to be reusable vehicles and must return to LEO, it can be seen that a single trip could result in 600 to 1000 cycles on the electric propulsion subsystem, whereas if the EPS were designed to supply continuous power, only 2 cycles per trip would be required. Also, an EPS designed to provide continuous power is sized for excess capability during sun-bathing periods as energy must be supplied to storage.

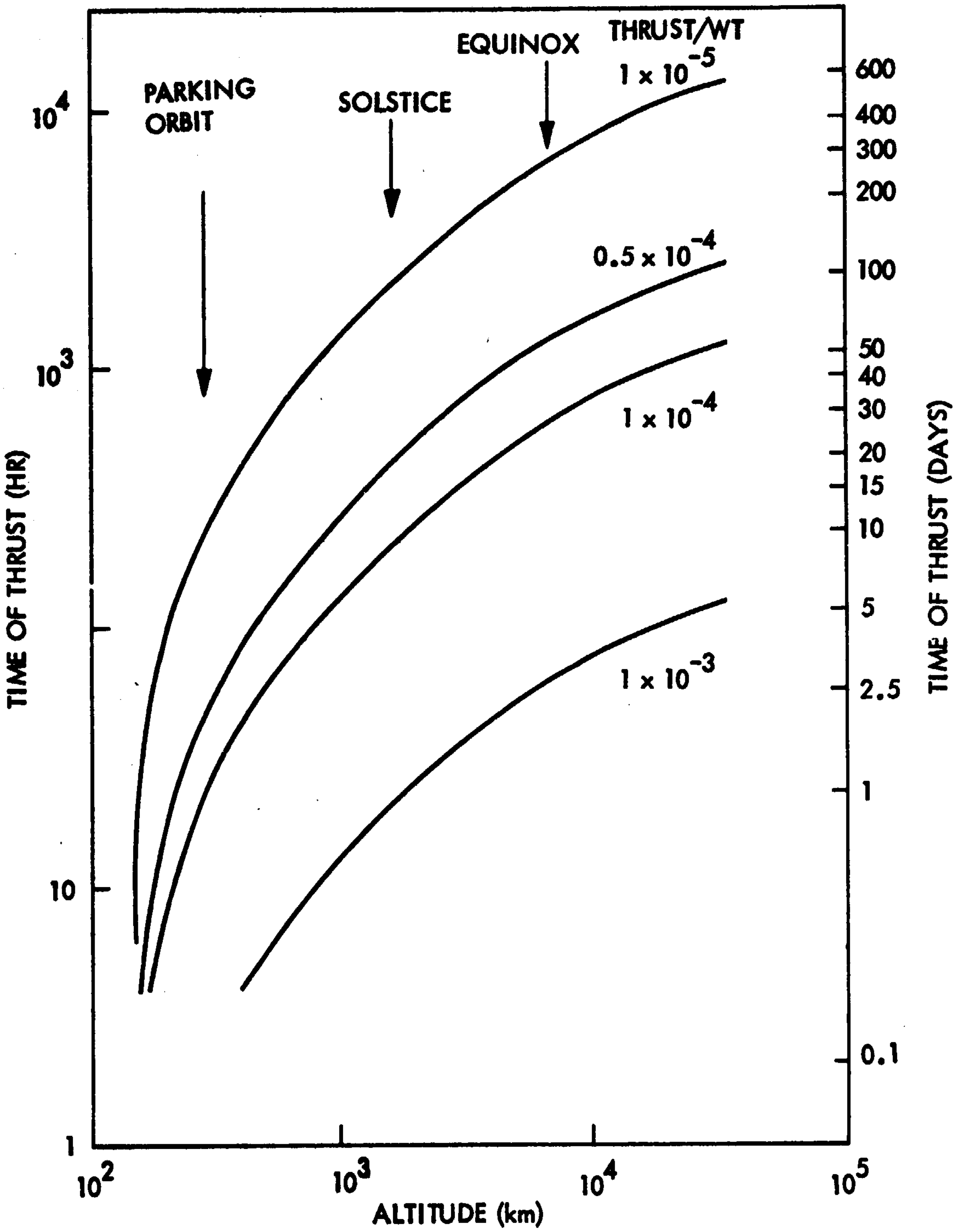
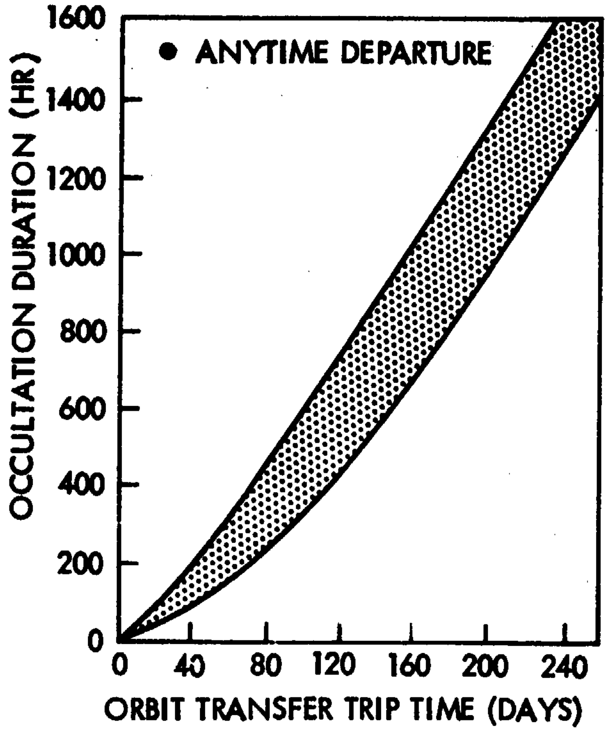
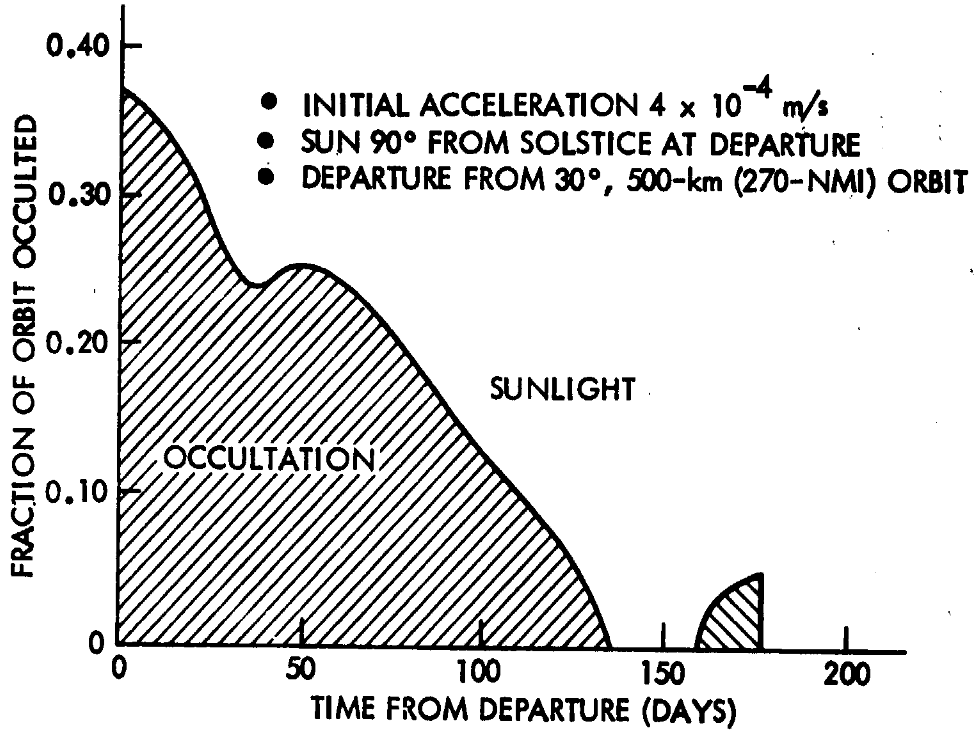


Figure 55. T/W effort on thrusting time



a. Occultation versus SETV Transfer Time



b. Occultation Duration

Figure 56. Occultations for anytime departure

As the SETV climbs out from LEO and the percentage of the orbit in the shade decreases, this excess power design will require less precise sun-pointing. Based on the above, the higher reliability associated with minimal cyclic life, and approximately equal trip times, the continuously operating EPS was selected for both SETVs.

Thrusters and Propellant Selection

Prior to selecting a thruster size, an evaluation was made of what propellant should be used for the SETV. While most of the current development work is being done on ion bombardment thruster using mercury as the propellant, recent studies have shown that use of the rare gasses, specifically argon, zenon, and neon may have many advantages when vehicle integration and contamination, ecology, and cost are considered. As reported in Table LII, argon is estimated to be considerably more abundant in the earth's atmosphere than the other two rare gasses.

TABLE LII. ATMOSPHERIC CONSTITUENCY – SEA LEVEL

Argon	4.84×10^{16} kg
Neon	9.36×10^{13} kg
Zenon	4.16×10^{11} kg

Argon also has another advantage in that the normalized cost per liquified mass is less than the other rare gasses, e.g., if argon is 1 Krypton is 488 times greater, and xenon is 1100 times greater. Based on the above, argon was selected as the propellant for both SETVs.

Thruster Size Selection

The initial evaluation of thruster size began with consideration of the 30-cm ion thruster under development for the SEPS. Based on the probable mass of the SETV and the results of trip time studies which showed T/W ratios should be in the 1×10^{-4} to 1×10^{-5} range, it became apparent that the 30-cm thruster under development had an unacceptably low thrust level. For instance, the studies showed that for the SSETV that total thrust should be 7 N or greater for reasonable trip times, i.e., less than 300 days. At the 0.127-N thrust level of the 30-cm thruster, 55 thrusters would be required. When the LSETV is considered, approximately 800 would be required.

Using the data of Reference 20, presented in Figure 57, for a range of specific impulses from 3000 to 7500 s and a propellant utilization of 90%, calculations were made for a 30-cm-thruster diameter and a 100-cm-diameter thruster.

For the SSETV, as a function of specific impulse and thruster diameter, the following minimum number of thrusters would be required:

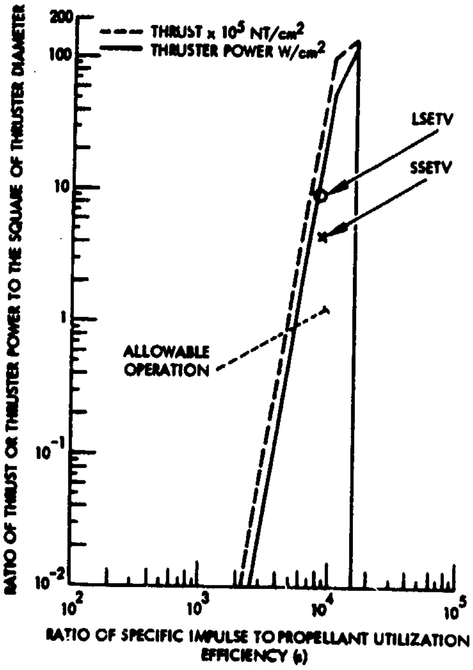


Figure 57. Allowable thruster operating regime

Thruster Diameter (cm)		Specific Impulse (s) 3000	5000	7500
30	No. of Thrusters	7750	320	72
100		700	29	7

These thruster quantities are based on the maximum allowable power per thruster from Figure 57.

Preliminary discussions with the costing analysts indicated that higher specific impulse with the resultant lower propellant requirements should reduce overall mission model costs. Based on the physical aspects stated above and the probable reduction in costs, the 100-cm-diameter thruster at a specific impulse of 7500 s was selected for both the SSETV and LSETV. To operate within the allowable limits shown in Figure 57, a ten-thruster arrangement was selected for the SSETV and 50 thrusters for the LSETV. The number of thrusters and the selected specific impulse are near-optimum; however, a full optimization analysis was not conducted.

Design Concept Selection

As shown in Figures 45 and 46, the SSETV and the LSETV each consist of two independent EPSs with a combination support subsystem/electric propulsion subsystem bus between the EPSs. Since each SETV has the function of a self-contained transfer vehicle, all necessary support subsystems are incorporated in the overall mass estimates. Data for these subsystems were drawn heavily from Reference 15 supplemented with analysis during the study. The mass estimates for the SSETV support bus are as follows:

SSETV Support Bus	Weight (kg)
Structure/Mechanisms/Cabling	184
Communications	30
Command, Computer, Data Handling	10
Guidance, Navigation and Control	80
RCS Components	47
Power Control and Distribution	54
Thermal Control	41
Adapter	55
	501
Weight Growth	50
SSETV Support Bus Total	551

Assuming the same subsystems in the LSETV and accounting for size increase in such items as RCS tanks and thrusters, adapters, etc., a mass of 1377 kg was estimated for the LSETV support bus.

The SETV support bus mass estimates include the equipments required for a redenzvous and docking subsystem, i.e., TV cameras and laser radar.

The control subsystem includes the sensors and reaction control propulsion subsystem that stabilizes the vehicle when the electric propulsion subsystem is not operating. The RCS thrusters are arranged for 3-axis control, maneuvering, and translation.

Thruster/Argon Tankage Characteristics

The thruster mass estimate followed the approach cited in Reference 20 wherein new packaging techniques were assumed which permitted reduction in the masses of structural components. Application for the use of argon as the propellant led to the computation of argon ion thruster mass as a function of diameter and these data are presented in Figure 58. The selection of the 100-cm-diameter thruster resulted in an estimated mass of 20 kg. This mass was used for both the SSETV and LSETV thrusters.

Argon thruster efficiency as a function of specific impulse used in all calculations is shown in Figure 59. A propellant utilization (efficency (η_u) of 0.9 was assumed. While these assumptions are higher than those achieved from present test projections, they are considered to be achievable values based on projections of state-of-the-art in the late 1980's.

The ion thrusters would be gimbaled in two axes and the selected arrangement would result in pitch, roll, and yaw torque capability during electric propulsion operation.

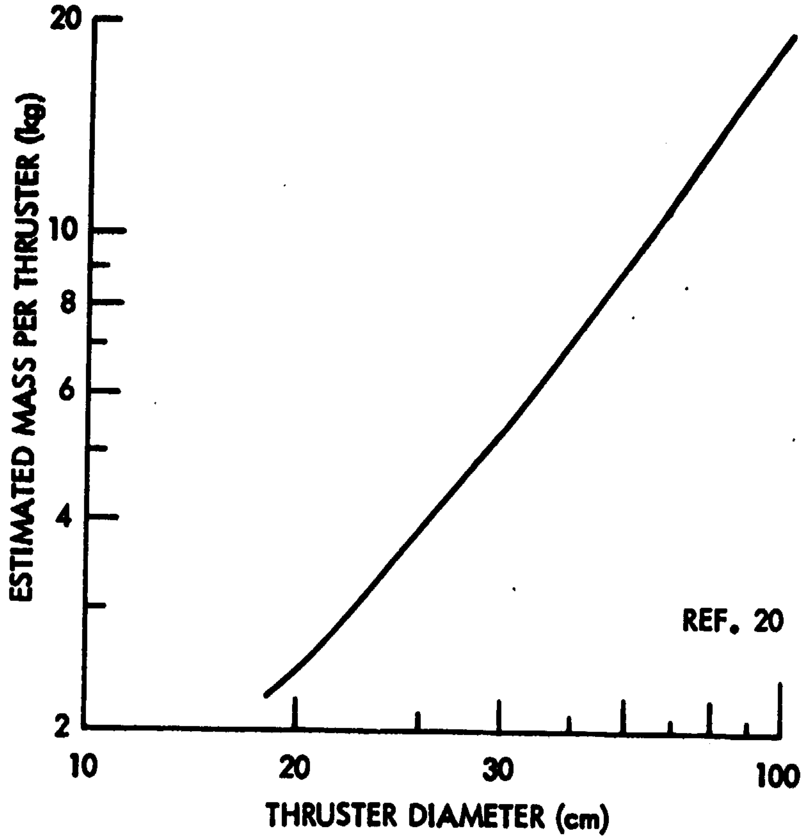


Figure 58. Estimated argon thruster mass

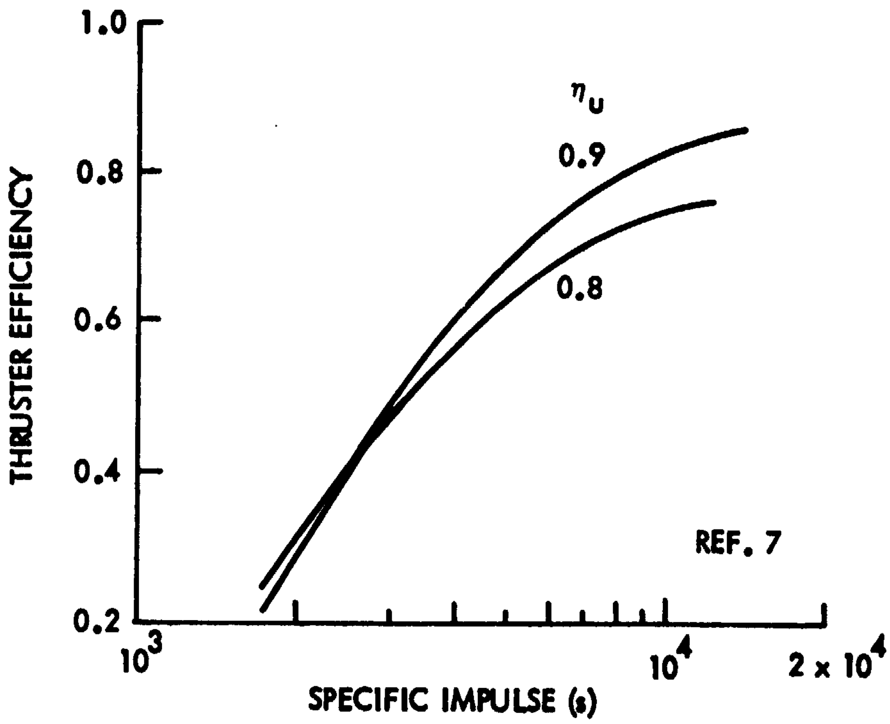


Figure 59. Argon thruster efficiency as a function of specific impulse

If the generated torques are not sufficient to control or maneuver the vehicle, the RCS thrusters can supply additional or primary torques. All 10 thrusters are gimbaled on the SSETV. For the LSETV, only 16 thrusters (all outer rows) were assumed to be gimbaled. The gimbal assembly mass was estimated to be 29.5 kg each. This estimate was made by taking the estimated mass from Reference 14 for the ion thruster gimbal assembly, gimbal actuators, yoke latches, and thruster support structure (11.8 kg) and multiplying by the ratio of masses of the 100- to the 30-cm thruster. A two-tankage arrangement was selected for both the SSETV and LSETV. The argon is assumed to be stored cryogenically.

Tankage mass as a function of propellant mass was estimated by using the same approach as that used in Reference 20 and is shown in Figure 60. The argon propellant tanks are assumed to be cooled by venting off of argon. Using the Reference 20 approach, an argon boil-off rate per square meter of tank surface area of 0.164×10^{-5} kg/s was calculated

Based on the calculated usable propellant required for the baseline SSETV mission, and the boil-off for a round-trip mission, the two tanks in the SSETV are 1.16 m in diameter. This results in a boil-off of 360 kg of argon. Using the same approach for the LSETV, the two propellant tanks were calculated to be 2.83 m in diameter. Based on a 161-day estimate round-trip, total argon boil-off will be 1150 kg.

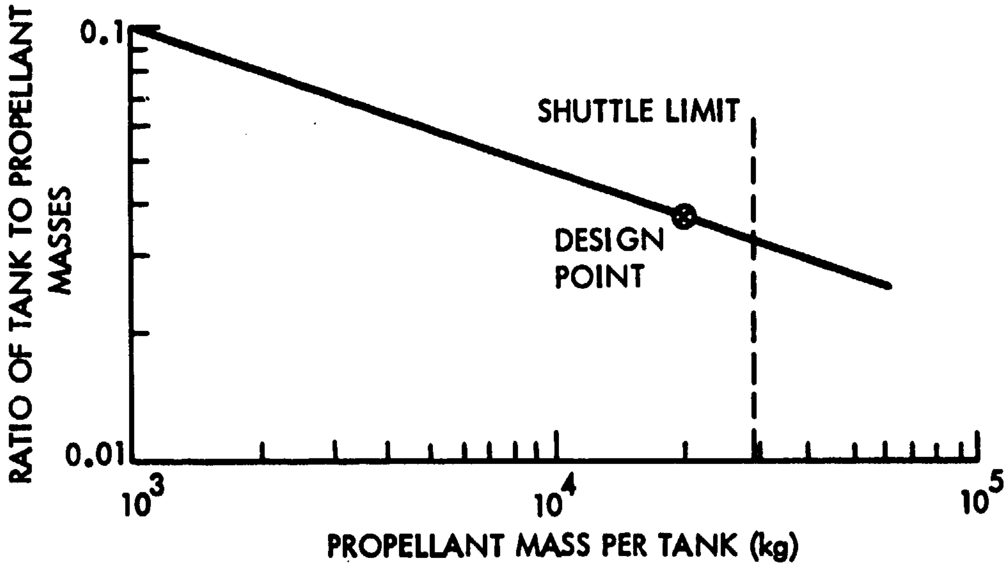


Figure 60. Ratio of tank to propellant mass

SSETV Bus

The SSETV Bus is composed of the support bus, the thruster module with gimbal assemblies, the thruster power conditioning equipments, the argon tanks, and the structure interconnecting the support bus to the thruster section. The equipments did not include power processor units for the ion thrusters as the use of the Brayton cycle turboalternator EPS resulted in voltage compatible with that required by the ion thrusters, i.e., 1000 to 2000 v. The mass of the SSETV bus is as follows

- The total mass of the SSETV is as follows:

Support Bus	551	kg
100-cm Thrusters (50)	200	
Gimbal Assemblies (16) including Thruster Mount Structure	295	
Power Conditioning and Switching Matrix Equipment	88	
Argon Tankage (2)	203	
SSETV Bus	1337	kg
Solar Collectors (2)	120	
Absorber/Receivers (2)	2940	
Brayton Rotating Units (2)	1564	(includes radiation and regeneration)
Structure	463	
SSETV Dry Mass	6424	kg
Payload	2268	
Total Dry Mass	8692	kg
Usable Argon	1674	
N_2H_4	200	
Vented Argon	360	
Deployed Mass	10926	kg

- The total mass of the LSETV is as follows:

Support Bus	1337	kg
100-cm Thrusters (50)	1000	
Gimbal Assemblies (16) including Thruster Mount Structure	472	
Power Conditioning and Switching Matrix Equipment	1243	
Argon Tankage (2)	1248	

LSETV Bus

- The same approach was used for the LSETV bus, and the mass was as follows:

LSETV Bus	5340	kg
Solar Collectors (2)	1625	
Absorber/Receivers (2)	54000	
Brayton Turbogenerators (2)	14145	(includes regenerator)
Energy Exchangers (2)	600	
Radiators	4330	
Structure	1180	
LSETV Dry Mass	81220	kg
Payload	148,000	
Total Dry Mass	229,220	kg
Usable Argon	28,720	
Vented Argon	1,150	
N_2H_4	3,600	
Total Deployed Mass	262,690	kg

These subsystem mass estimates were calculated by using LMSC computer programs for the collectors and the LSETV radiator. For the SSETV, the Brayton Rotating Unit Mass was estimated from LMSC-generated data for the Laser Power Conversion Analysis study and includes the mass of the heat rejection radiator and the cycle regenerator. The LSETV mass was estimated by using the Reference 18 value of 1.5 kg/kW_e for the Brayton turbogenerator.

The RCS hydrazine values are estimates based on part LMSC experience in studies for such vehicles as the space shuttle, the reusable Agena, the space tug and other vehicles which require 3-axis orbital control, rendezvous, and docking.

Operational Considerations

The two primary factors that have a major affect on operations of the SETVs is trip time and propellant required, both of which affect costs. The trip time affects the number of vehicles required to transfer payloads in the mission model as a function of time. The propellant used on each trip affects the cost by requiring numerous earth to LEO operations to resupply the SETVs.

Trip Time

Trip time is primarily affected by the vehicle thrust-to-weight ratio as shown in Figure 55. These trip times are for transfer from the parking orbit to synchronous orbit altitude, i.e., 35,000 km. A 300-km parking orbit was assumed and resulted in a velocity increment of 4757 m/s for the altitude change. Figures 61 and 62 show that if the inclination change is made at low altitude, velocity losses will be significantly greater than if done at geosynchronous altitude. Thus for trip time calculation, a sequential maneuver approach was taken; that is, increase altitude to 35,000 km, then change inclination from 28.5° to 0°. This approach was designated sequential thrusting and is shown in Figure 63. It can be noted from Figure 62 that to minimize losses, central burn angles must be relatively small. Studies were made and a 50° burn angle was selected. This resulted in a 3% velocity increase and to maintain orbit altitude, the firing would have to be at 104° instead of 90°, and thus another 3% loss was incurred, resulting in a total loss of 6%.

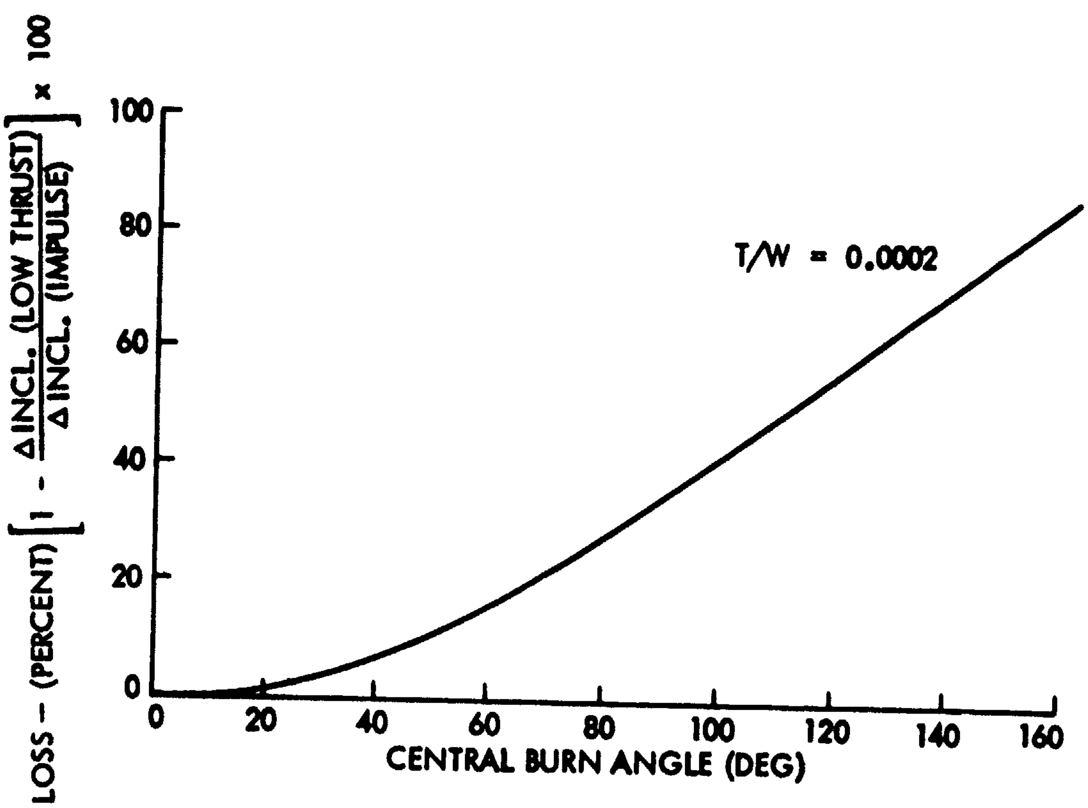


Figure 61. Effect of low thrust on inclination changes low-altitude (160-nmi) circular orbit

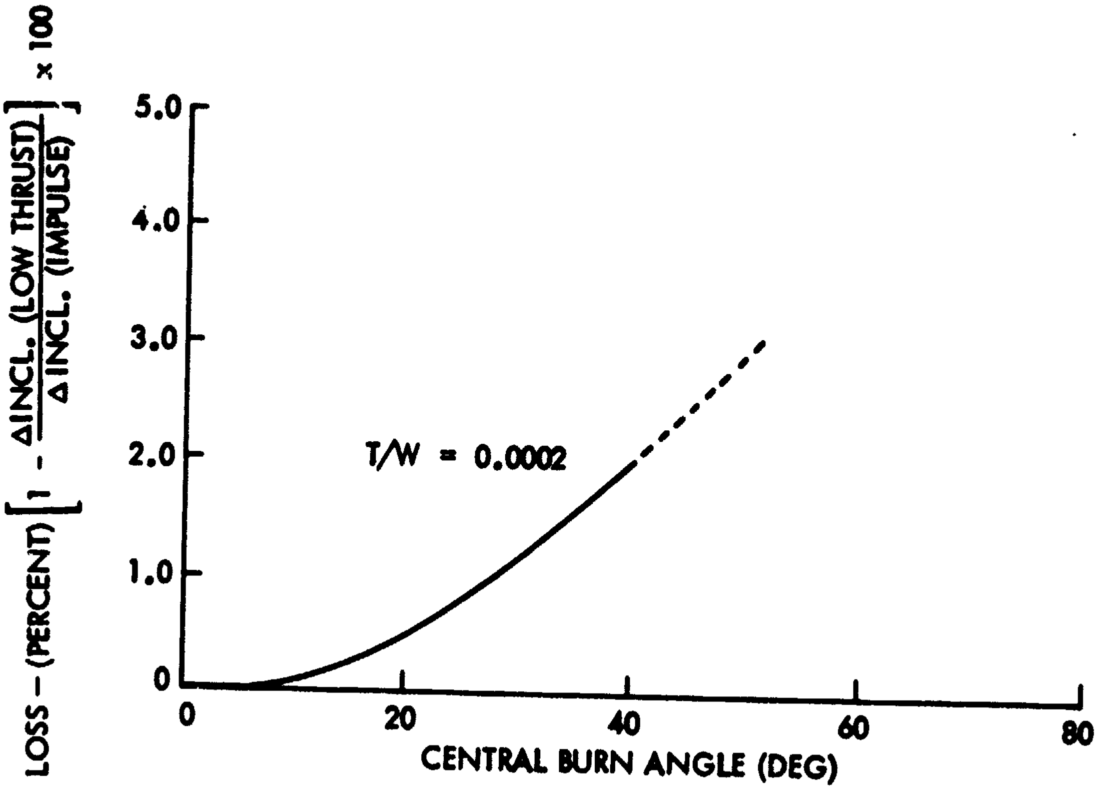
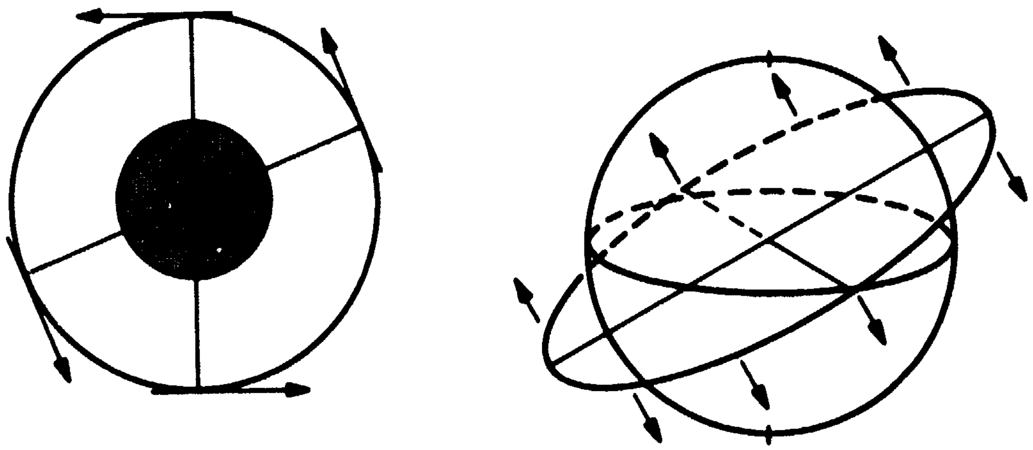


Figure 62. Effect of low thrust on inclination change; geosynchronous orbit



(a) Circumferential for altitude raising.

(b) Normal, switched at orbit antinodes for inclination change alone.

Figure 63. Thrust programs for accomplishing various orbit changes

Subsequently, additional literature search resulted in data which showed that if an optimal yaw thrusting program was used, trip times could be reduced. The approach is illustrated in Figure 64a and consists of continuous thrusting during the climb out with yaw thrusting about the node and the antinode. This reduces transfer time approximately 22% for the range of payloads evaluated in Reference 15 (Figure 64b). While this range of payloads is less than that of the SETVs, it was decided that trip times would be calculated by the sequential thrusting approach and then reduced 22%. This approach was used only for the vehicles which had continuous thrusting capability as the effect of periodic thrusting (~ 60%) on the orbit for sun-only vehicles was not evaluated.

The approach for computing trip times was to calculate the thrust-to-weight for each mission phase, use Figure 55 to estimate the orbit transfer time, and then add the time to change inclination for 28.5° to 0° and then back to 28.5°. For vehicles which did not have thermal storage and could operate in sunlight only (full sun only thrusting), it was assumed that 60% of the orbital period was available for thrusting until the vehicle climbed into full sunlight. The climbout time associated with solstice and equinox departures was calculated and the average of the two was used.

The time to make the inclination change was computed by determining the propellant required for the maneuver and the calculating the thrusting time required. The total time was then determined based on two firings over a 100° total central angle.

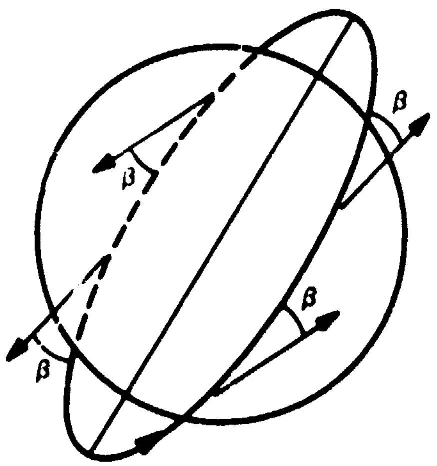


Figure 64a. Thrust program for accomplishing simultaneous altitude and inclination change

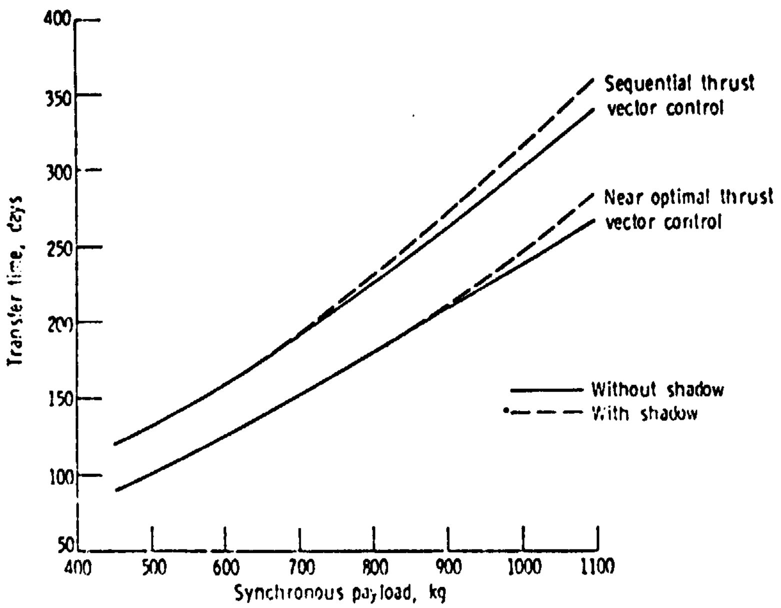


Figure 64b. Capability of solar electric propulsion stage to achieve 5-year mission life with 5 kw of power remaining.

A number of vehicle configurations were evaluated to determine effects of thruster input power, thrust level, and thruster specific impulse on trip time. The results are shown in Table LIII.

The first vehicle configu ation assessed was a Solar Electric Propulsion System (SEPS). Because of the solar array degradation due to the Van Allen belts, the array size was increased by a factor of 3. The low thrust level of this configuration resulted in a low T/W and an unacceptably high trip time. Based on Figure 55 results, the vehicle power level was increased to 203 kWe so thrust level could be increased; however, trip times still were in the order of 400 days. Power level was then doubled but round trip times did not change appreciably.

All the vehicles initially assessed used conventional radiator designs and aluminum solar concentrators. Based on LMSC efforts in the Laser Power Conversion Analysis and other studies, a vehicle was configured using the lightweight Mylar concentrator and a lighter weight radiator design. This vehicle was designated the "Advanced Technology Vehicle" and since it was lighter, resulted in higher T/W ratios and as seen in Table LIII, reduced trip times. It is the selected configuration and was used in all cost studies.

Table LIII shows that for the SSETV that the continuous thrusting configuration at a specific impulse of 7500 s has the shortest trip time. Since this vehicle also requires less than half the propellant that would be requ red with a 3960-s specific impulse, it was the selected configuration. The minimal optimization conducted does not result in a conclusive design point selection, but the calculated data appear to show that a specific impulse of 7500 s is near optimum when the total program is considered. That is, trip times are near minimum and cost of transporting propellant for refueling is near minimum.

TABLE LIII. SETV TRIP TIME COMPARISON

Vehicle	Thruster Power (kWe)	Thrust (n)	Usable Propellant (kg)	ISP (s)	Mission Ave T/W	Vehicle Dry Mass (kg)	(Days) Round Trip Cont. Thrusting	Time Seq. Thrusting	Full Sun Only	Remarks
SSETV	25	1.0	2502	3000	1.5×10^{-5}	4651	–	–	739	SEPS with 3X Solar Array
SSETV	203	6.6	4646	3000	6.5×10^{-5}	8639	–	–	404	SOA* Vehicle
SSETV	203	6.21	3243	4000	6.3×10^{-5}	8639	–	–	407	SOA Vehicle
SSETV	406	13.2	6110	3000	10×10^{-5}	12910	–	–	365	SOA Vehicle
SSETV	406	8.5	2138	7500	6.2×10^{-5}	12910	–	–	417	SOA Vehicle
SSETV	406	8.5	2675	7500	5.5×10^{-5}	14439	328	421	–	SOA Vehicle W/Thermal Storage
SSETV	406	6.8	1569	10000	5.7×10^{-5}	12910	–	–	470	SOA Vehicle
SSETV	406	8.5	1674	7500	8.8×10^{-5}	8692	220	282	–	Adv. Technology Vehicle Thermal Storage
LSETV	9430	203	28720	7500	1.5×10^{-4}	81220	161	204	–	Adv. Technology Vehicle Thermal Storage

*SOA – State-of-the-Art

Note: (1) For SSETV, Vehicle Dry Mass includes a 2268 kg payload
(2) For LSETV payload mass is not included

3.13.2 SETV Concept Cost and Evaluation

Two SETV concepts were defined for which system life-cycle cost were estimated. The two concepts were:

- Small SETV (SSETV) with 2268 kg (5,000 lb) round-trip capability to GEO from LEO orbit in 220 days
- Large SETV (LSETV) with 148,000 kg (326,000 lb) one way to GEO orbit and return empty to LEO orbit in 161 days

These systems correspond in capability to the two Space Based Laser Rocket Systems as discussed previously in Section 3. Also, the SETV life-cycle cost include comparable cost elements to the laser rocket systems to allow relative cost comparisons for a given mission model.

<u>Ground Rules and Assumptions</u>

Most of the same ground rules and assumptions as applied in Task 4 to the space-based laser rocket systems were applied to the SETV system except for the following:

	SSETV	LSETV
• Round-trip time (days)	220	161
• Maximum number of missions/vehicle in 10 years of operations	16	22
• Ion thruster module replacement frequency (number of missions)	4	5
• Vehicle refurbishment frequency (number of missions)	8	10
• Average number of missions performed per vehicle prior to expendable mission	2	2
• Vehicle deployment flights	2 Shuttles	1.2 HLLVs
• On-orbit assembly required	Yes	Yes
• Launch vehicle cost/flight	\$13.5 M	\$6.5 M
• Fuel resupply and refurbishment flights on flight sharing basis	Yes	Yes

The round-trip times include an 11-day allowance on the average for payload attachment, payload deployment, refueling, thruster replacement, and vehicle refurbishment. The thruster replacement frequency is based on an estimated life of 20,000 hr per ion thruster module. The vehicle refurbishment frequency is based on an estimated life of 40,000 hr for the Brayton power units. The maximum number of missions per vehicle is a direct fallout from the number of missions which can be performed in a 10-year operating period with the above given round-trip times. Fractional trips were not allowed (i.e., 3,650 days ÷ 220 days = 16.59 or 16 missions/vehicle).

The SETV fleet size for a given mission model was determined based on the maximum number of missions per SETV and the missions model traffic requirements allowing for expendable vehicle missions. For example: In the Case 3 model, SSETV's are required to perform 450 reusable missions in 10 expendable missions. The fleet size was established as follows:

Missions	Vehicles
10 expendable 20 reusable (2 × 10)	10
430 reusable (430 ÷ 16 = 26.9)	27
Vehicle Fleet Size	37

The vehicle deployment requirements in terms of the number of Shuttle flights or HLLV flights are based on packaging constraints rather than on weight. In the SSETV case, two Shuttle flights were estimated to be required to deploy the SSETV with two 40.6-m-diameter collectors and their associated radiators. Comparable figures for the laser rocket propulsion vehicle was 2/1 for the Shuttle or a factor of 4 for the SSETV. In the LSETV case, the volumetrically undefined HLLV is used, and the assumption was made to apply a factor of 2 to the laser rocket system deployment requirements to arrive at 1.2 HLLV flights per LSETV deployment. Since the HLLV volume will be considerably larger than the Shuttle, packaging constraints should be alleviated even though the LSETV has two 160-m-diameter collectors and associated large radiators. For comparison purposes the same Shuttle fee and HLLV costs per flight are used as was done for laser rocket propulsion case even though they may not represent the latest estimates of absolute cost for delivery to LEO.

Solar Electric Transfer Vehicle Costs

The SETV life-cycle costs were estimated in comparable fashion to the laser rocket propulsion system costs. Parametric costs estimating relationships were used for spacecraft systems.

Gross estimate analogs were used for the ion propulsion systems and the Brayton power systems since there is little actual cost data base. The ion propulsion system development cost was equated to new cryogenic engine development cost and the power systems were based on complex machinery cost and high temperatures/exotic materials production cost factors. Production systems development cost was determined by factoring the unit cost. Technology costs were not included in the development cost. The development cost does include engineering models of the ion propulsion and power units and a complete SETV prototype which is flight-tested.

Tables LIV and LV show the development and first unit cost for the SSETV and the LSETV, respectively. The first unit costs were put on a 90% learning curve to arrive at fleet investment and spares cost.

The SETV refurbishment cost, thruster replacement cost, and fuel resupply cost were estimated as follows:

TABLE LIV. SSETV COST ESTIMATE
(FY '77 $ Millions)

Subsystem	DDT&E Cost ($ M)	First Unit Cost ($ M)
SPACECRAFT:		
Structures	$ 19.40	$ 0.97
Communication and Command	7.13	4.46
Guidance and Navigation	5.83	3.43
Reaction Control	1.86	0.62
Electrical Power	1.63	1.42
Thermal	2.40	0.40
Adapter	2.90	0.29
Subtotal	$ 41.15	$11.59
ION PROPULSION:		
Thrusters	$100.00	$ 0.50
Gimbals, Attach. Str., etc.		1.49
Tanks	1.43	0.41
Subtotal	$101.43	$ 2.40
POWER SUBSYSTEM:		
Braytons, Collectors, and Absorber/Receiver	$ 80.00	$14.48
Structure	3.20	1.70
Subtotal	$ 83.20	$16.18
SUBSYSTEMS TOTAL:	$225.78	$30.17
System Engr. and Intgr.	33.87	4.83
System Test	9.03	2.41
Ground Support Equipment and Elect C/O Equipment	18.06	–
Launch Ops.	5.03	–
Flight Ops.	4.81	–
Grd. C^3 Mods.	25.00	–
Shuttle Feet	27.00	–
Data	11.86	1.12
Program Management	15.43	1.93
TOTAL	$375.87	$40.46

TABLE LV. LSETV COST ESTIMATE
(FY '77 $ MILLIONS)

Subsystem	DDT&E Cost ($ M)	First Unit Cost ($ M)
SPACECRAFT:		
Structures	$ 32.70	$ 2.18
Communication and Command	7.13	4.46
Guidance and Navigation	5.83	3.43
Reaction Control	3.18	1.59
Electrical Power	1.63	1.42
Thermal	3.68	0.92
Adapter	3.36	0.42
Subtotal	$ 57.51	$14.42
ION PROPULSION:		
Thrusters }	$200.00	$ 2.50
Gimbals, Attach. Str., etc. }		4.14
Tanks	3.96	1.32
Subtotal	$203.96	$ 7.96
POWER SUBSYSTEM:		
Braytons, Collectors, and Absorber/Receiver	$170.00	$43.13
Structure	4.32	3.21
Subtotal	$174.32	$46.34
SUBSYSTEMS TOTAL:	$435.79	$68.72
System Engr. and Integr.	65.37	11.00
System Test	17.43	5.50
Ground Support Equipment and Elect C/O Equipment	34.86	–
Launch Ops.	10.18	–
Flight Ops.	11.33	–
Grd. C^3 Mods.	25.00	–
HLLV Fee	7.45	–
Data	23.00	2.56
Program Management	29.90	4.39
TOTAL	$660.31	$92.17

	SSETV	LSETV
• Vehicle refurbishment	$8.906 M	$15.11 M
• Thruster replacement	$3.395 M	$5.33 M
• Fuel resupply/mission	1.561 M	1.925 M

Other initial investment costs and annual operating costs were estimated in comparable investment manner as in the laser rocket propulsion case. These are identified below as a part of the life-cycle cost for each of the mission model cases considered.

Mission Model Life-Cycle Costs

Life-cycle costs were estimated for four mission model cases:

- Case 3-all SSETVs; 460 missions
- Case 6-all LSETVs; 4,514 missions
- Case 8-mix of SSETV and LSETV; 4,485 missions
- Case 11-mix of SSETV and LSETV; 8,439 missions

These cases are described in greater detail in section 3.4.5.

Table LVI presents the estimated life-cycle cost in undiscounted FY '77 dollars.

For the two mixed cases of the SSETV and LSETV in Cases 8 and 11, the assumptions were made that full DDT&E cost for the LSETV will be accrued. However, the SSETV DDT&E cost will be lower by about $100 million since some of the same development is common to both (i.e., spacecraft subsystems and part of the propulsion system). As shown in Table LVI, the life-cycle cost for the four-mission model cases range from $4 billion to $59 billion in undiscounted dollars. The average cost per mission ranges from $7 to $15 million with the latter figure being heavily weighted by the large percentage (85%) of the expendable missions in that case.

SETV Versus Laser Rocket Propulsion Cost Comparison

The comparison of the SETV cost to the laser rocket propulsion system cost on a case basis is presented in Table LVII. The Space-Based Laser Rocket System costs are from Tables XXII through XXVI. On undiscounted bases, the cost ratios range from 1.2 to 0.88 for the four mission model cases indicating that the solar electric system is within + 20 to -12% in cost of the laser rocket propulsion system. These differences are within the estimating accuracy of the life-cycle costs and therefore show the SETV to be competitive with the laser rocket propulsion system.

The spread in the ratios indicate the effect of the large laser rocket propulsion RDT&E cost on the life-cycle cost. In Case 3, the size of the mission model (460 missions) is not sufficient for the laser rocket propulsion system to be able to overcome the heavy contribution of the DDT&E cost and compete with the relatively low SSETV DDT&E cost. The larger mission model cases (4,500 to 8,500 missions) has sufficient missions to

TABLE LVI. MISSION MODEL LCC, UNDISCOUNTED
(FY '77 $ Millions)

Cost Breakdown	Case 3	Case 6	Case 8 SSETV	Case 8 LSETV	Case 8 TOTAL	Case 11 SSETV	Case 11 LSETV	Case 11 TOTAL
Fleet Size	37	217	100	182	282	39	364	403
DDT&E	$ 375.9	$ 660.3	$ 275.0	$ 660.3	$ 935.3	$ 275.0	$ 660.3	$ 935.3
INVESTMENT								
SETV	$1004.4	$10,370.2	$2352.4	$ 8,928.1	$11,280.5	$1050.8	$16,101.0	$17,151.8
Spares	100.4	1,037.0	235.2	892.8	1,128.0	105.1	1,610.1	1,715.2
Deployment	999.0	1,616.7	2700.0	1,355.9	4,055.9	1053.0	2,711.8	3,764.8
Launch Operations	132.2	1,258.7	315.9	1,079.7	1,395.6	138.4	1,975.9	2,114.3
Initial Training	11.1	114.1	25.9	98.2	124.1	11.6	177.1	188.7
Ground Support Equipment	33.1	342.2	77.7	294.6	372.3	34.7	531.3	566.0
C&C Equipment and Facilities	5.0	21.7	10.0	18.2	28.2	5.0	36.4	41.4
Data and Program Management	10.8	103.2	25.9	88.6	114.5	11.3	162.0	173.3
Investment Total	$2296.0	$14,863.8	$5743.0	$12,756.1	$18,499.1	$2409.9	$23,305.6	$25,715.5
OPERATIONS (10 YR)								
Thruster Replacement	$ 183.3	$ 2,164.0	$ 98.5	$ 1,940.1	$ 2,038.6	$ 169.8	$ 3,874.9	$ 4,044.7
Vehicle Refurbishment	240.5	6,134.7	124.7	5,484.9	5,609.6	222.6	10,985.0	11,207.6
Replacement Training	27.6	285.2	64.7	245.5	310.2	28.9	442.8	471.7
GSE Maintenance	6.6	68.4	15.5	58.9	74.4	6.9	106.3	113.2
Facilities Maintenance	6.0	9.3	7.0	8.7	15.7	6.0	12.3	18.3
Flight Operations	108.2	351.6	130.4	343.3	473.7	109.2	389.8	499.0
Data and Program Management	10.6	51.1	15.5	46.8	62.3	10.8	67.7	78.5
Fuel Resupply	718.0	8,689.4	757.1	7,700.0	8,457.1	685.3	15,400.0	16,085.3
Operations Total	$1300.8	$17,756.7	$1213.4	$15,828.6	$17,041.6	$1239.5	$31,278.8	$32,518.3
TOTAL LCC	$3972.7	$33,280.8	$7231.4	$29,244.6	$36,476.0	$3924.4	$55,244.7	$59,169.1
Average Cost/Mission*	$ 8.6	$ 7.4	14.9	$ 7.3	$ 8.1	$ 8.9	$ 6.9	$ 7.0
$/lb R.T. To GEO*	1727.0	—	2982.0	—	—	1788.0	—	—
$/lb TO GEO	—	23.0		22.0	—	—	21.0	—
Payload Weight/Mission (lb)	5,000	326,000	5,000	326,000	Mix	5,000	325,000	Mix
*% Expendable Vehicles	27	6	85	0	30	36	0	3

TABLE LVII. LCC COST COMPARISON, UNDISCOUNTED
(FY '77 $ Millions)

Mission Model Case		Laser Rocket System	SETV System	Undiscounted Cost Ratio
	Fleet Size	16	37	
3	LCC	$ 4,528.1	$ 3,972.7	0.88
	Cost/Mission	$ 9.8	$ 8.6	
	Fleet Size	87	217	
6	LCC	$30,921.5	$33,280.8	1.08
	Cost/Mission	$ 6.9	$ 7.4	
	Fleet Size	152	282	
8	LCC	$31,502.1	$36,476.0	1.16
	Cost/Mission	$ 7.0	$ 8.1	
	Fleet Size	154	364	
11	LCC	$49,463.7	$59,169.1	
	Cost/Mission	$ 5.9	$ 7.0	1.20

*Cost Ratio = SETV LCC/LRP LCC

spread the laser rocket propulsion DDT&E cost. Also, the vehicle refurbishments during the life-cycle (22 missions) as opposed to one refurbishment to the SSETV, tends to favor the SSETV system in the Case 3 comparison as applied to other cases.

Cost Comparison, Solar Electric Transfer Vehicle and Laser Rocket System

The mission model definition which is described in section 3.1 was utilized for the life-cycle cost comparison between the SETV and the Laser Rocket System. The same mission model was used to maintain consistency. If desired, the life-cycle costs of the SETV can be compared to the cryogenic OTV system because the time frame and the number of missions are identical. There are some differences in the reuse and refurb capabilities as indicated in Table LVIII.

The Solar Electric Transfer Vehicle (SETV) Life-Cycle Costs are different from both the cryogenic system and the laser propulsion system. The major cost driver of the SETV system is the slow round-trip time, which necessitates a higher number of vehicles than the other alternative technologies. Additionally, the costs per refurbishment are the highest of the systems compared, and are required more frequently.

While some costs in the SETV System are higher than systems compared in this report, others are lower. The DDT&E costs are lower as well as fuel resupply. There is no laser transmitter so that no laser deployment costs exist for the SETV. The deployment cost of the SETV as a function of weight is more than the laser system but less than the cryogenic. Tables LIX through LXII indicate cost breakdowns for Cases 3, 6, 8, and 11.

TABLE LVIII. REUSE AND REFURBISHMENT CAPABILITIES OF LO_2/LH_2 AND LASER-PROPELLED OTVs

Attribute	5,000-lb Payload Capability		326,000-lb Payload Capability	
	Laser	S. SETV	Laser	L. SETV
Number of Reuses	40	16	60	22
Number of Missions Before Refurbishment	20	4	20	5
Number of Reuse Missions Prior to Use on an Expendable Mission (Refurbishment Prior to Expending)	10	2	10	2

Table LXIII provides a comparison of life-cycle costs between the SETV System and the Laser Rocket System with the laser system as the baseline. In Case 3, the SETV is less costly than the laser system, but in all other cases the SETV is more costly, and quite significantly more.

Two factors contribute to the Case 3 situation. The first is that the laser transmitter in Case 3 is not used intensively enough to drive down the per-flight cost of the transmitter; that is, the transmitter cost is not amortized over many flights, thus making the laser system somewhat costly in Case 3. The second factor is that the same low-intensity mission model does not call for a high number of SETV units as there is ample time within the 10-year horizon to complete the mission-model, thus the SETV system is not well utilized.

In Cases 6, 8, and 11, the laser system is more cost-effective. The basic driving factor is the high number of vehicles needed and their intensive refurbishment schedule. The more demanding the case, the more savings the laser system provides in the classic high-technology tradeoff of high development costs for lower per-unit costs.

Table LXIV depicting various cost parameters suffices to explain the differences from a cost standpoint. In the first column there is a comparison of the average cost per vehicle where the SETV is generally the highest. The next column shows the comparison of total number of flights per vehicle where the laser system is consistently the highest and the SETV system is consistently the lowest. The third column deals with the marginal cost per flight where the SETV consistently has the lowest marginal cost (refurbs and fuel). However, when the average total cost is computed, the laser system is generally the lowest cost because despite lower marginal operating costs, the higher vehicle-related costs of the SETV do not bring about a lower average cost. This is brought out graphically in Figures 65 and 66. Here, the magnitude of the area on the

TABLE LIX. LCC COST COMPARISON 3 – SETV VERSUS LASER SYSTEM

MISSION COMPOSITION: 450 5,000 LB P/L's
10 5,000 LB EXPENDABLE P/L's

NUMBER OF OTV'S =

LASER		SETV
16	SMALL	37

LCC COSTS (IN MILLIONS OF 1977 DOLLARS)

S.SETV	CATEGORY	SPACE-BASED LASER ROCKET SYSTEM
375.87	DDT&E	1377.80
1104.81	INVESTMENT & SPARES	442.99
0.0	LASER SYSTEM DEPLOYMENT	482.00
1350.15	OTV DEPLOYMENT & OPS.	295.77
423.79	REFURBS	302.54
718.03	FUEL RESUPPLY	1627.02
3972.65	TOTAL REAL YEAR LCC	4528.12
2048.49	TOTAL PRESENT VALUE COST (1984)	2456.40

TABLE LX. LCC COST COMPARISON 6 – SETV VERSUS LASER SYSTEM

MISSION COMPOSITION: 4,500 326,000 LB P/L's
14 326,000 LB EXPENDABLE P/L's

NUMBER OF OTV'S =

LASER		SETV
87	LARGE	217

LCC COSTS (IN MILLIONS OF 1977 DOLLARS)

L. SETV	CATEGORY	SPACE-BASED LASER ROCKET SYSTEM
660.31	DDT&E	4204.90
11407.21	INVESTMENT & SPARES	3736.53
0.0	LASER SYSTEM DEPLOYMENT	981.50
4225.23	OTV DEPLOYMENT & OPS.	1304.22
8298.64	REFURBS	2015.38
8689.45	FUEL RESUPPLY	18678.93
33280.84	TOTAL REAL YEAR LCC	30921.46
14590.93	TOTAL PRESENT VALUE COST (1984)	11954.83

TABLE LXI. LCC COST COMPARISON 8 – SETV VERSUS LASER SYSTEM

MISSION COMPOSITION: 4000 326,000 LB P/L's
400 5,000 LB P/L's
85 5,000 EXPENDABLE P/L's

NUMBER OF OTV'S =

LASER		SETV
67	LARGE	182
85	SMALL	100

LCC COSTS (IN MILLIONS OF 1977 DOLLARS)

SETV	CATEGORY	SPACE-BASED LASER ROCKET SYSTEM
935.31	DDT&E	4625.39
12408.54	INVESTMENT & SPARES	4855.28
0.0	LASER SYSTEM DEPLOYMENT	981.50
7026.87	OTV DEPLOYMENT & OPS.	2076.91
7648.19	REFURBS	2266.49
8457.09	FUEL RESUPPLY	16696.53
36476.00	TOTAL REAL YEAR LCC	31502.10
16751.99	TOTAL PRESENT VALUE COST (1984)	12347.29

TABLE LXII. LCC COST COMPARISON 11 – SETV VERSUS LASER SYSTEM

MISSION COMPOSITION: 8000 326,000 LB P/L's
425 5,000 LB P/L's
14 5,000 EXPENDABLE P/L's

NUMBER OF OTV'S =

LASER		SETV
135	LARGE	364
19	SMALL	39

LCC COSTS (IN MILLIONS OF 1977 DOLLARS)

SETV	CATEGORY	SPACE-BASED LASER ROCKET SYSTEM
935.31	DDT&E	4625.39
18866.96	INVESTMENT & SPARES	5510.44
0.0	LASER SYSTEM DEPLOYMENT	981.50
8029.25	OTV DEPLOYMENT & OPS.	2098.50
15252.28	REFURBS	3013.09
16085.28	FUEL RESUPPLY	33234.82
59169.08	TOTAL REAL YEAR LCC	49463.74
25683.40	TOTAL PRESENT VALUE COST (1984)	18016.71

TABLE LXIII. LIFE-CYCLE COST COMPARISON BETWEEN SETV AND LASER ROCKET SYSTEM

CASE	SETV CHANGE IN CONSTANT YEAR DOLLARS	SETV CHANGE IN PRESENT VALUE DOLLARS
CASE 3	12 % LESS	17 % LESS
CASE 6	8 % MORE	22 % MORE
CASE 8	16 % MORE	36 % MORE
CASE 11	20 % MORE	43 % MORE

TABLE LXIV. COST PARAMETERS COMPARISON

CASE	AVERAGE COST PER VEHICLE	AVERAGE FLIGHTS PER VEHICLE	MARGINAL COST PER FLIGHT	AVERAGE TOTAL COST PER FLIGHT
	CRYO LASER SETV	CRYO LASER SETV	CRYO LASER SETV	CRYO LASER SETV
CASE 3	31.6/27.7/29.9	20 / 29 / 12	28.0 / 4.1 / 2.5	32.2 / 9.8 / 8.6
CASE 6	48.7/42.9/52.6	28 / 51 / 21	45.9 / 4.6 / 3.8	49.3 / 6.9 / 7.4
CASE 8	37.1/31.9/44.0	20 / 30 / 16	41.4 / 4.2 / 3.6	45.4 / 7.0 / 8.1
CASE 11	43.4/35.8/46.8	29 / 55 / 21	43.5 / 4.3 / 3.7	46.6 / 5.9 / 7.0

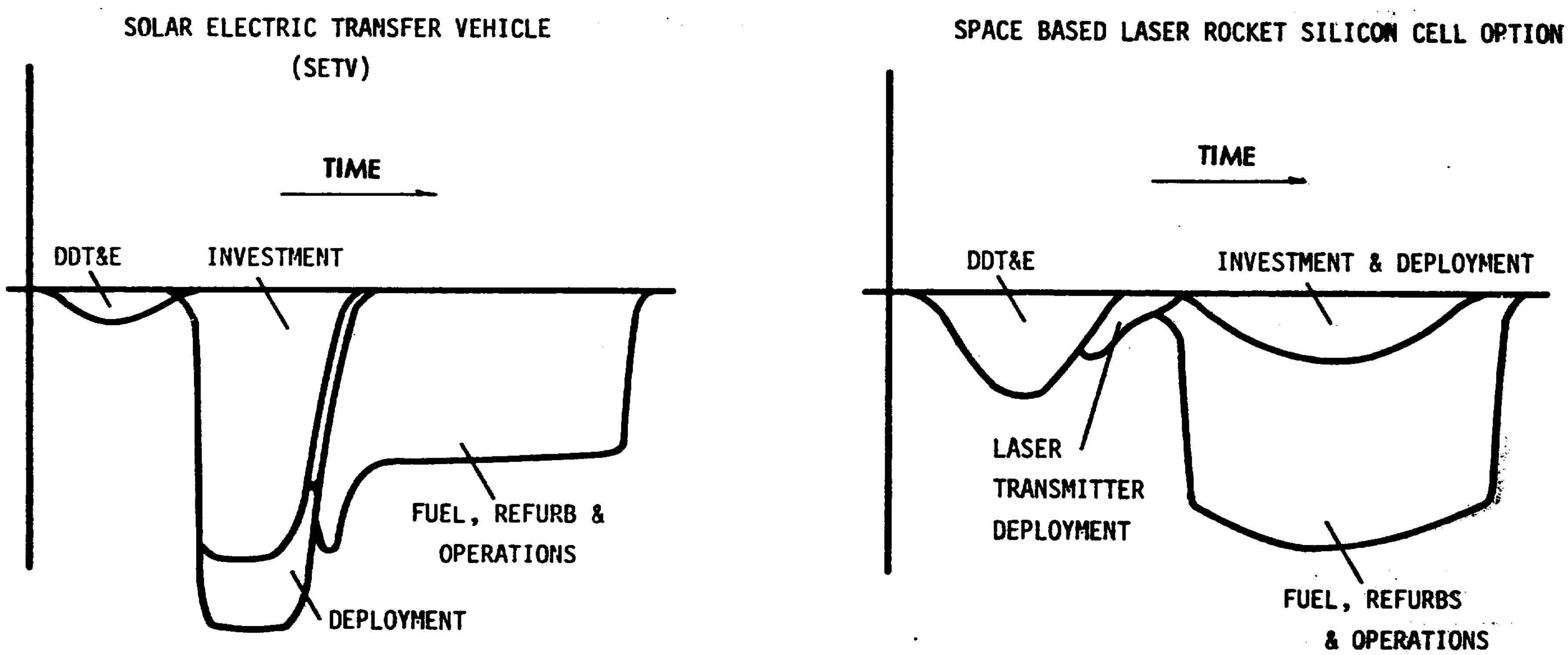


Figure 65. Case 6 Life-Cycle Cost Profiles: Laser and SETV

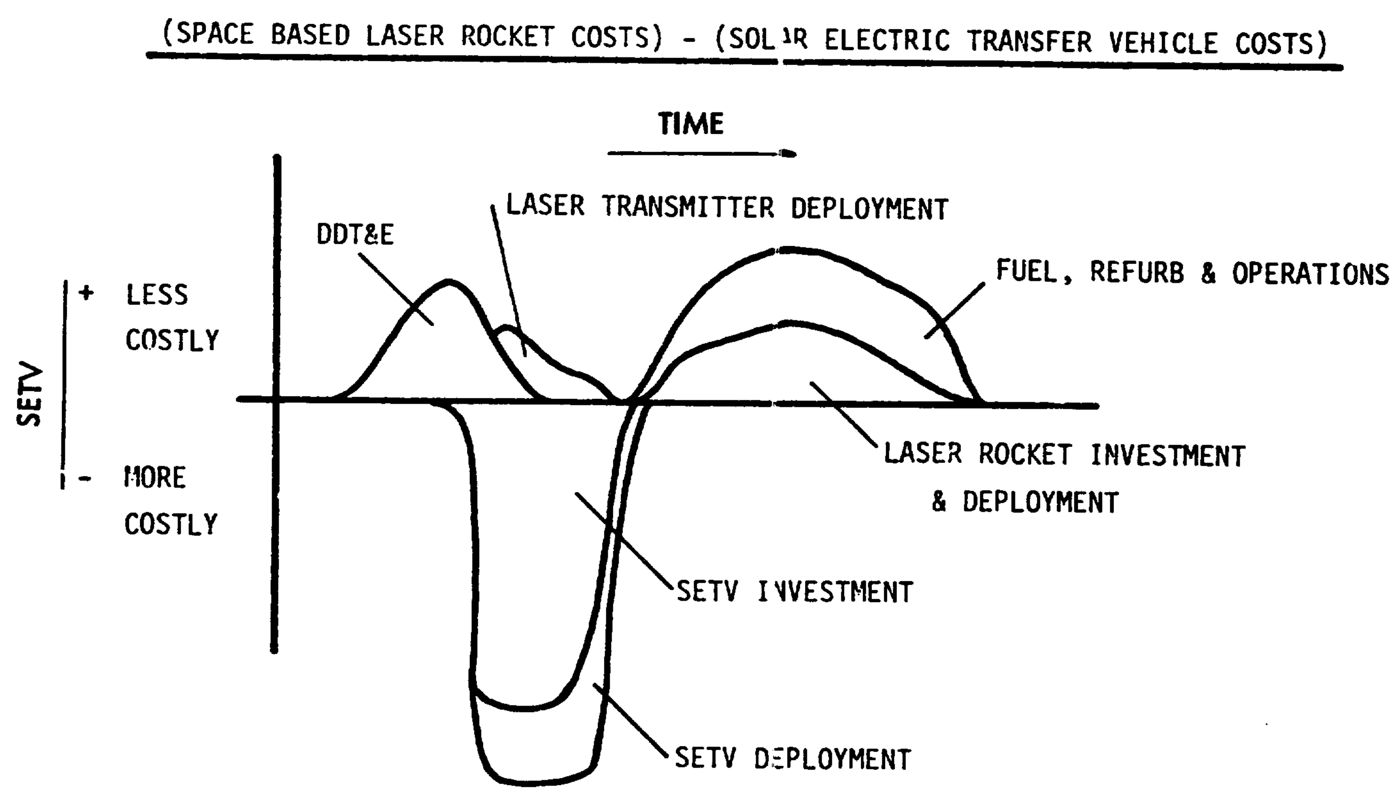


Figure 66. Case 6 Life-Cycle Cost Profile Differences

less costly or more costly side indicate the relative portions over time. Because the number of flights per vehicle is constant, the size of costs for the SETV is linear except for DDT&E and, thus, the more demanding the mission model, the more cost-effective is the laser system.

In summary, the laser system is the best choice from a cost viewpoint, except in the case where the U.S. space flight activity is expected to drop to a low level as in Case 3. Even in Case 3, more research would be necessary into both system alternatives to provide better data for a choice, as the savings are not significant enough to sway the decision on cost alone.

Section 4

DISCUSSION OF RESULTS

The results obtained in this analysis of laser rocket systems clearly show that significant cost savings can be attained with laser rocket systems when compared to cryogenic chemical systems (LO_2-LH_2) performing the same mission model. Twelve cases, varying the mission model, were analyzed for both the space and ground systems. Table LXV summarizes four representative cases for both space- and ground-laser rocket systems. In addition to the cost ratios for a normal 10-year life cycle (includes all development costs), ratios are presented without DDT&E costs. This is done to provide a spread of cost ratios for each case because the probability of all development costs being charged to the laser rocket system is extremely small. Therefore, the true cost savings ratio lie within the ratio spread.

TABLE LXV. COST-EFFECTIVENESS RATIOS (DISCOUNTED CRYOGENIC COSTS)/(DISCOUNTED LASER SYSTEM COSTS) FOR EQUAL CAPABILITY

	Case 3		Case 6		Case 8		Case 11	
	Space	Ground	Space	Ground	Space	Ground	Space	Ground
Normal Life-Cycle Costs	2.37	2.25	5.90	5.77	5.23	5.15	6.91	6.69
Without DDT&E Costs	4.08	3.47	8.02	7.64	7.27	6.98	8.57	8.12

The break-even point, or the point at which the cost savings ratio is 1 lies between 115 and 120 flights of the small payloads over the 10-year life cycle. This is within the range of current activity which will undoubtedly increase when the shuttle becomes operational.

The cost ratios shown in Table LXV are basically a result of the savings of propellant and the cost of transporting the propellant resupply from earth to low-earth orbit. Figure 67 illustrates the point for Case 11. The ratio of propellant required for the chemical system to the laser system is about 10:1 and the resulting cumulative costs show that savings are being accrued within a few months of the initial operating capability as shown in Figure 68.

Obviously, the propellant resupply is the major cost driver but are there other cost drivers that are also significant, or if an error was made in propellant requirements,

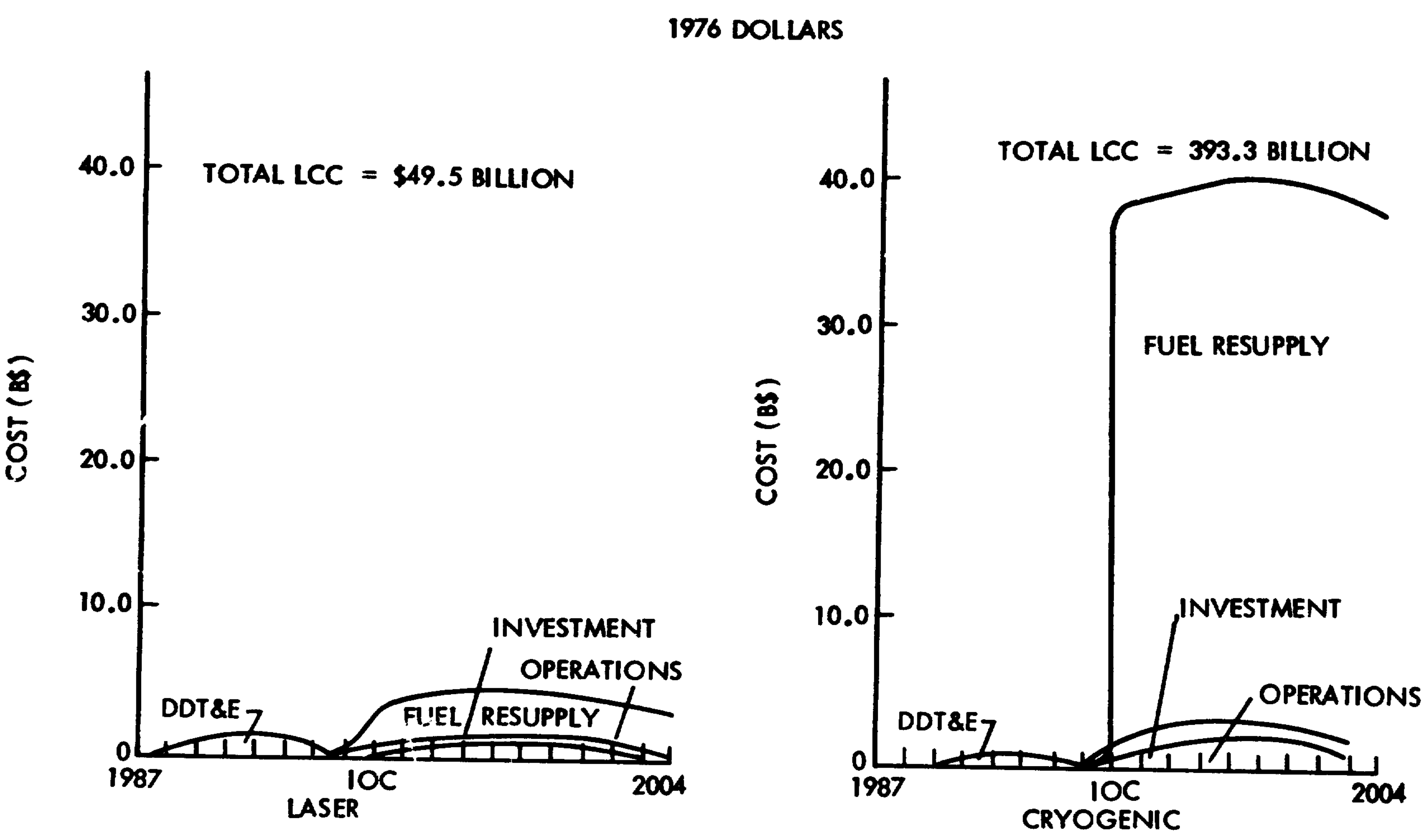


Figure 67. Case 11 costs per year

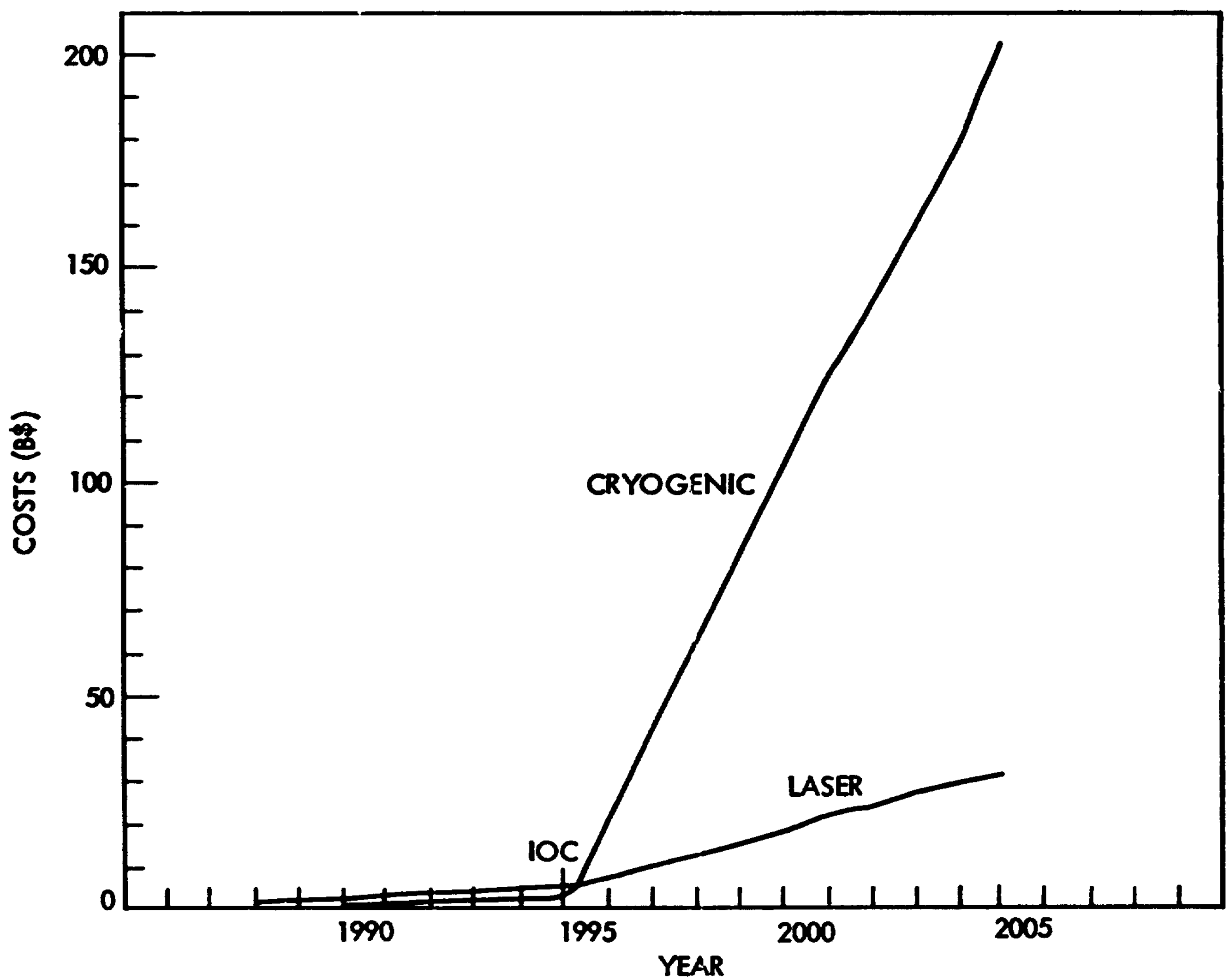


Figure 68. Case 8 cumulative costs

what effect would it have on the cost ratios? To answer these questions various cost components were evaluated including the propellant resupply. Figure 45 shows the sensitivity of RDT&E costs which indicates that these costs are not major drivers. Figure 46 shows that the number of years of operation will not significantly affect the cost ratio. Figure 47 shows the sensitivity of reusability which again is not significant. Figure 48 shows the sensitivity of the fuel resupply cost which was obviously a major cost driver; however, it may also be noted that an error of 50% (from 10:1 to 5:1) in propellant requirements would still result in a substantial cost savings.

In summary, the laser rocket system offers potential cost savings that should not be ignored. The cost ratios developed in this analysis, while preliminary, will not change significantly with more definitive analysis.

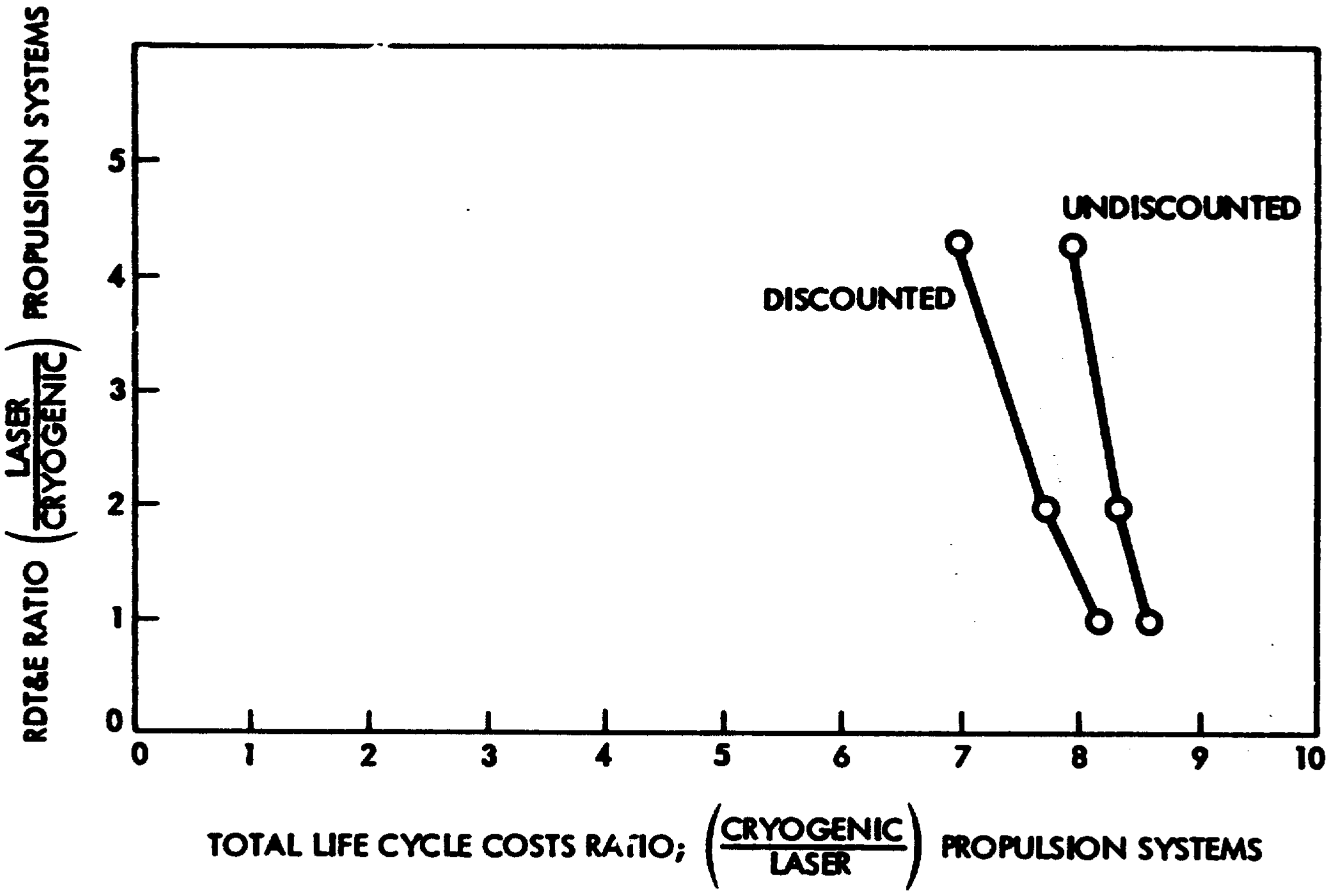


Figure 69. Sensitivity of RDT&E costs

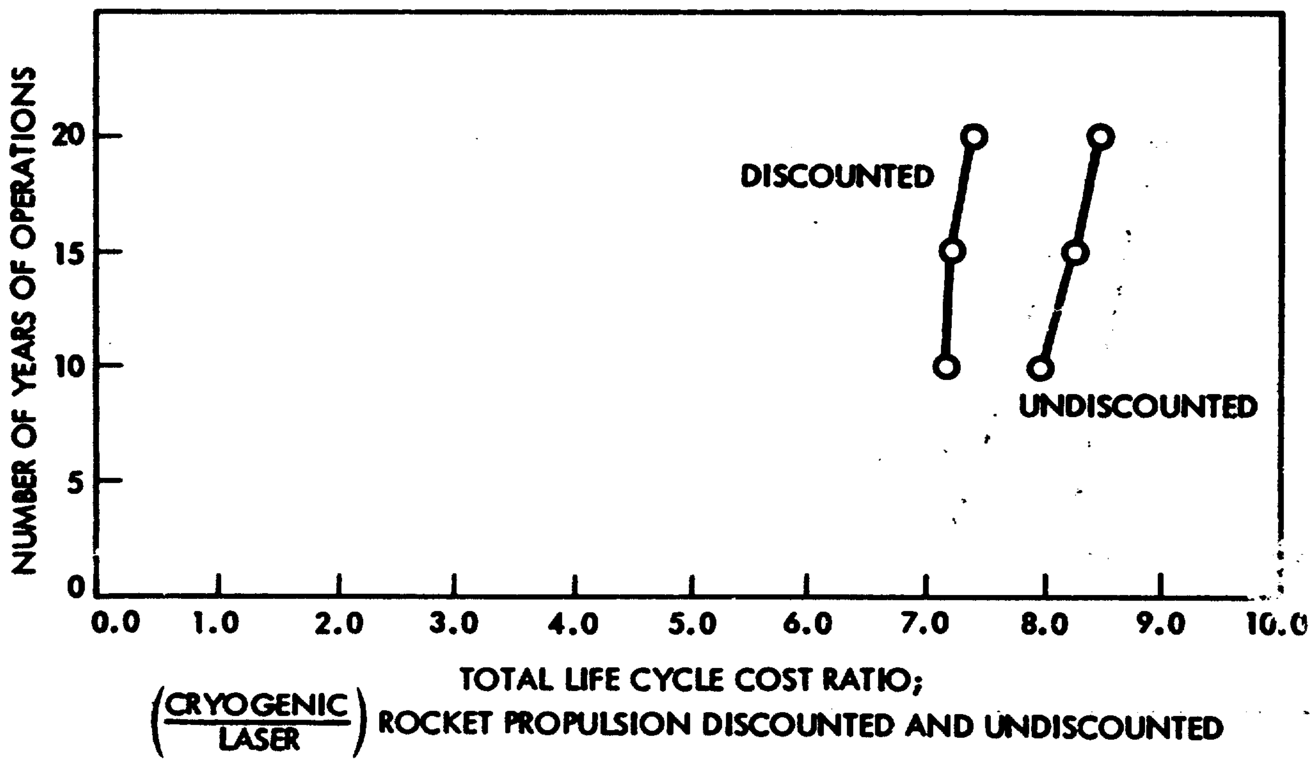


Figure 70. Number of years of operations

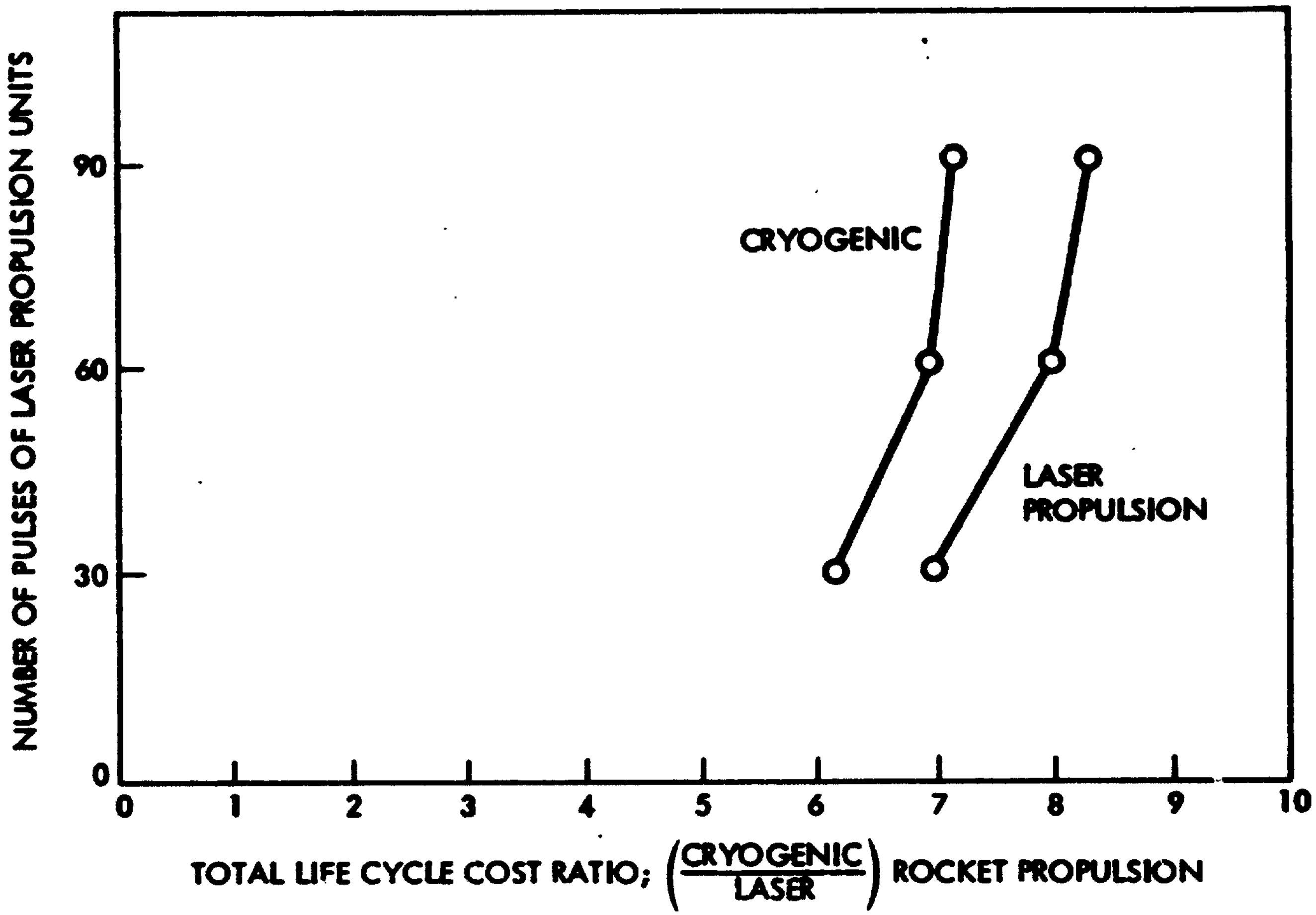


Figure 71. Sensitivity of reusability

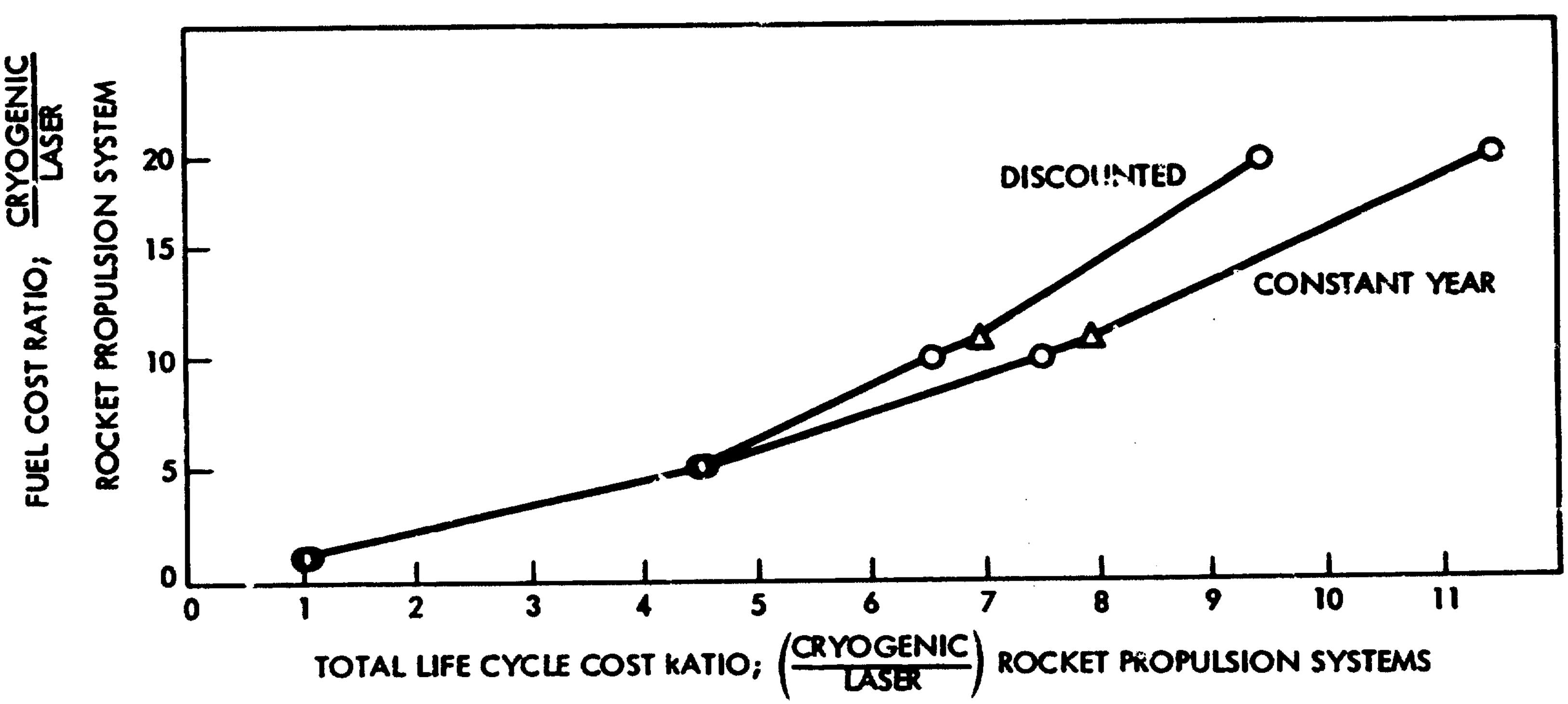


Figure 12. Relative fuel resupply cost sensitivity

Section 5

CONCLUSIONS

Laser rocket systems are very attractive when compared to conventional chemical orbital propulsion systems of equal capability on the basis of 10-year life cycle costs. The development of laser rocket systems will require substantial technology advancements, but the payoff is also substantial. Based on the very encouraging results of the Laser Rocket Systems Analysis, it is the conclusion of the authors that work should continue and specifically recommend:

- Continuation of the CW Rocket Thruster Program
- Initiation of a Laser Rocket System Technology Development and Program Plan Study
- Initiation of a Space-Based Electrical Power Supply Systems Analysis for power levels in the range required by space-based laser rocket systems

C-3

Section 6

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