Space Transfer Concepts and Analyses for Exploration Missions

Contract NAS8-37857

Technical Directive 12

Beamed Power Systems Study

Final Report

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FOREWORD

This report is the final report for Technical Directive 12 of the Space Transfer Concepts and Analyses for Exploration Missions study, MSFC Contract no. NAS8-37857. This Technical Directive was performed during the first half of 1992; the report was written in November 1992. The subject of the Technical Directive was parametric performance and cost/benefit analysis of Earth-based laser powered electric orbit transfer.

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FIGURES (Continued)

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ABBREVIATIONS AND ACRONYMS

ALS	Advanced Launch System
ASRM	Advanced Solid Rocket Motor
BOL	Beginning of Life
CFLOS	Cloud-Free Line of Sight
DDT&E	Design, Development, Test and Evaluation (cost)
DOS	Disk Operating System; as used here refers to an IBM PC-type desktop computer
E-beam	Electron Beam
EOTV	Electric (Propulsion) Orbit Transfer Vehicle
EPS	Electrical Power System
ETO	Earth to Orbit
FEL	Free Electron Laser
GEO	Geosynchronous Earth Orbit
GN&C	Guidance, Navigation and Control
IMLEO	Initial Mass in Low Earth Orbit
ISRU	In-Situ Resource Utilization
JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
LCC	Life Cycle Cost
LEO	Low Earth Orbit
LeRC	Lewis Research Center
MIT	Massachusetts Institute of Technology
MSFC	Marshall Space Flight Center
N	newtons (4.4482N = 1 lbf)
NEP	Nuclear Electric Propulsion
NLS	Notational Launch System
PMAD	Power Management and Distribution
PV or p/v	photovoltaic
RAAN	Right Ascension of the Ascending Node
RF	Radio Frequency
SDIO	Strategic Defense Initiative Office
SEPS	Solar Electric Propulsion System
SSF	Space Station Freedom

ABSTRACT

Parametric models were constructed for Earth-based laser powered electric orbits transfer from low Earth orbit to geosynchronous orbit. These models were used to carry out performance, cost/benefit and sensitivity analyses of laser-powered transfer systems, including end-to-end life cycle cost analyses for complete systems. Comparisons with conventional orbit transfer systems were made, indicating large potential cost savings for laser-powered transfer. Approximate optimization was done to determine best parameter values for the systems.

Orbit transfer flights simulations were conducted to explore effects of parameters not practical to model with a spread-sheet. The simulations considered view factors that determine when power can be transferred from ground stations to an orbit transfer vehicle and conducted sensitivity analyses for numbers of ground stations, Isp including dual-Isp transfers, and plane change profiles. Optimal steering laws were used for simultaneous altitude and plane change. Viewing geometry and low-thrust orbit raising were simultaneously simulated. A very preliminary investigation of relay mirrors was made.

1. INTRODUCTION AND SUMMARY

1.1 PURPOSE

The purpose of this document is to present a final report on results from completion of the technical tasks described in paragraph 4 of Technical Directive 12: Beamed Power Analyses, for NASA contract NAS8-37857.

1.2 BACKGROUND

Power beaming by coherent electromagnetic radiation has attracted technical interest since the early publications of Glaser, et. al. on microwave power beaming. Power beaming has natural advantages for transfer of power over long distance in space. The beam propagation is lossless and the beam conduit, free space, has no mass or cost. In principle, beams of arbitrarily high collimation can be formed, although required apertures may be quite large and there are practical engineering limits to attainable wavefront precision, which must be high in proportion to the collimation needed. Some of the mechanisms available for generating electromagnetic waves involve low-entropy sources for which conversion efficiency should, in principle, be very high. High efficiencies have been achieved in practice at microwave frequencies. Laser generators have to date been low in efficiency but this is a question of engineering state of the art rather than physical limits.

Studies of microwave power beaming were conducted in the 1970s and 1980s. At microwave frequencies, very large apertures (kilometers) are required to achieve the collimation for efficient power transmission over space distances of practical interest. Such large apertures cannot be economic except for very large blocks of power. Since aperture is proportional to wavelength, use of laser frequencies offers the potential of efficient transmission over cislunar distances with apertures on the order of meters. This brings the range of economic blocks of power down to levels of interest for current space projects.

Technology advancements over the past several years, mainly sponsored by the Strategic Defense Initiative, have produced a state of the art that is near capability for directing such blocks of highly collimated laser power from the surface of the Earth to lunar distance and beyond. If the source of power is on the Earth, high efficiency in conversion from electric power to laser power is not economically important and laser efficiencies currently achievable are adequate to realize advantageous systems. Key elements of this emerging laser technology include:

a. Continuous operation with infrared or visible light generated by a free-electron laser, at hundreds of kW up to a few megawatts: High power has been demonstrated for short periods and Boeing is currently under contract with the U. S. Army to demonstrate high duty cycle power generation by a free-electron laser at 100 kW or more. The state of the art presently under development could be uprated to the megawatt range.

b. Adaptive optics capable of forming a highly collimated beam and responding rapidly enough to compensate for atmospheric turbulence, making it possible to direct a near-diffraction-limited beam through the atmosphere into space with acceptable beam degradation. Flight experiments have demonstrated this capability at modest apertures of roughly one meter. Laboratory tests and analyses now in progress are developing technical alternatives suitable for apertures up to ten meters, with many thousands of adjustable elements, as needed for turbulence compensation in apertures of this size.

c. Photovoltaic arrays capable of converting laser light to electricity at greater than 50% efficiency. This has been demonstrated on a laboratory scale at low power with continuous laser light. Planned experiments will develop means of matching photovoltaic response to laser pulse formats so that high-power systems can be operated at high efficiency.

d. Concepts for efficient, very light weight electric propulsion power processors and thrusters, based on electron beam accelerator and related technologies. These offer the potential for unprecedentedly high power-to-weight ratios for electric propulsion systems, which in turn yield relatively short trip times for electric propulsion systems in cislunar space.

These technological potentials led to a number of applications ideas from various investigators, collected by NASA Headquarters Code R under the name "Selene", signifying SpacE Laser Electric ENErgy. The applications include laser electric power for a lunar base, laser-powered electric propulsion transfer systems, and laser power to satellites in cislunar space — particularly laser power to geosynchronous satellites during Sun occultation periods.

During 1992 an analytical working group was formed by NASA Headquarters to assess the technical and economic merit of these ideas. Participants included NASA Headquarters, Boeing, Comsat Corporation, JPL, MIT Lincoln Labs, LeRC, MSFC, Science Research Laboratories, and W. J. Schaefer Associates.

The Boeing contribution to this activity, documented in this report, was funded through the Space Transfer Concepts and Analysis for Exploration Missions Study, Contract NAS8-37857. This contribution included systems/cost/economics analysis and performance simulations for laser-electric orbit transfer systems, for low Earth orbit to geosynchronous orbit and lunar missions. Boeing worked with the other participants in the study to develop an integrated technical and economics assessment, documented in a NASA Headquarters briefing conducted on July 16, 1992. This report presents the results of the Boeing studies in more detail and depth, as a complement to the integrated briefing.

The applications studied by this group by no means exhaust the interesting possibilities for application of electromagnetic power and energy beaming. A survey is included in this report as Section 4.0.

1.3 SCOPE OF WORK AND REPORT ORGANIZATION

This final report for Technical Directive 12 covers work performed from January through August 1992. The Statement of Work tasks were as follows[1]:

a. Integrate JPL and Boeing beamed power computer models and lunar systems data to establish end-to-end power system effectiveness for lunar transportation systems and other uses. Determine sensitivities and parametric effects on beamed power concepts. Optimize power system elements to minimize system costs. Determine benefits of lower transportation costs to other system elements.

b. Identify and quantify advantages of beamed power applications, including electric transfer vehicle GEO performance, and communication satellite life extension and performance enhancement. Review and assess assumptions of Lewis Research Center 1990-91 laser beamed power study.

c. Perform trades/analyses of competing concepts; parametric sensitivity data over ranges of interest of key parameters. Provide data supportive of the overall systems analysis/integration activity.

d. Attend meetings at MSFC and other locations, provide briefings as appropriate. Provide report at end of the work. Data (briefings, reports, etc.) shall be provided in digital form (Mac preferred) and in paper form.

The study was carried out by addressing the tasks though (a) construction and use of an integrated spread-sheet model of the transportation system to perform systems parametrics and cost/economics analyses and trades, and (b) time-dependent performance simulation of laser beam powered orbit transfers to perform systems parametric analyses and trades that require detailed simulation. The report is organized into two primary sections representing these two activities since this leads to a clearer exposition of results than organization according to the statement of work. Also, a survey was conducted to identify all known applications of power beaming; this is included as an Appendix (Appendix A) in this report.

1.4 SUMMARY OF RESULTS

The Boeing studies investigated two important aspects of electric propulsion orbit transfer missions powered by Earth-based lasers: (1) Parametric optimization analyses for integrated systems to find combinations of system parameters that yield economic performance and short trip times, and (2) Detailed performance simulations to define appropriate values for some of the parameters, especially related to line-of-sight occurrence between laser ground sites and orbiting vehicles and resulting integrated duty factors and trip times.

The study found that laser-powered electric orbit transfer has major potential cost and economic advantages for LEO to GEO and LEO to lunar orbit missions. These advantages derive from the high specific impulse that can be used, eliminating most of the propellant that otherwise must be transported to low Earth orbit at high cost, and from the high power density achievable with the power collection and conversion system on the orbit transfer vehicle, yielding unprecedentedly high specific power estimates, and as a result performance, for the electric propulsion system.

Laser powered vehicles exhibit a large number of design-choice parameters which influence performance and cost in complex interrelated ways. These include number and location of ground stations; laser power and wavelength; beam expander aperture; flight system receiver aperture; solar array power handling capacity, efficiency and mass; power processing and thruster efficiency and mass; electric propulsion specific impulse; and life/number of reuses for the flight system. The study performed parametric optimizations and sensitivity analyses. Cost advantages of the laser electric system tend to be insensitive to variation in system parameters.

The general range of system parameters found to yield good cost performance is as follows:

Laser wavelength	0.85 micron
Beam expander aperture	10 meters
Laser power	1 to 5 megawatts
Number of ground sites	4 to 6

Electric specific impulse	2000 to 3000 seconds, with the early part of the transfer accomplished using resistojets at about 800 seconds
P/V array size	10 to 15 meters diameter
Array output	200 to 500 kWe
Zenith angle	60 - 70 degrees
Transfer time (up)	20 to 50 days
Number of vehicle reuses	3 to 10
Payload to GEO orbit	2500 to 5000 kg

Important external influences include space transfer traffic level (the market), cost of space transportation to low Earth orbit, degradation effects of the van Allen radiation belts on phototvoltaic arrays used for the transfer system, cost of competing orbit transfer systems, and shared amortization of investment in laser technology and ground stations.

These interact as described later in the report; an optimization criterion is defined as the least life cycle cost over a practical operating lifetime; optimization analysis selected optimal values of the design-choice parameters and investigated their sensitivity as well as the bottom-line life cycle cost to variations in the external parameters. Since the global optimum trip time is often uncomfortably long, optimum values can also be determined as a function of trip time. A judgmental selection of trip time, in view of the life cycle cost penalty, can then be made.

The laser-electric system suffers from poor duty factors when the space vehicle is in low Earth orbit. The duty factor for one ground station is less than 1% when the vehicle is at 500 km. altitude. This low duty factor includes typical orbit inclination and ground station latitude effects. As the vehicle altitude increases, the duty factor increases rapidly, reaching a value on the order of 20% (from a ground station) at geosynchronous altitude. The geometry is somewhat complex; a flight simulation approach was determined to be the most practical approach to investigating parameters dependent on view factors. A software package designed to solve a similar problem was available and was adapted to suit this purpose. Simulations were performed in conjunction with the systems analyses to develop overall results. These simulations helped develop several insights in how to configure and operate the systems as described later in the report.

A significant proportion of the Boeing activity was directed to integration with other investigators. A portion of our work statement included "integration of models".

As the work progressed, it became clear that integration of results of models was more efficient than literal computational integration, and the former is what we did. Boeing worked closely with MSFC, and participated in several meetings and teleconferences. These included in-depth discussions of technical and economic aspects of design operation of laser-electric space power systems. Results of these deliberations were incorporated into the integrated briefings prepared by NASA and its contractors.

2.0 PARAMETRIC AND COST/ECONOMIC ANALYSES

This section describes the systems concept, construction of a spread-sheet model of a laser-powered orbit transfer system, and results of analyses performed using the model.

2.1 OVERALL SYSTEM DESCRIPTION

The reference laser-powered orbit transfer concept is depicted in figures 2.1 and 2.2. Four or more laser ground sites, located in remote, high-altitude, clear-weather areas are used. Each ground site has one or more high-power lasers with beam expanders and directors. The system may have multiple targets: satellites needing laser power, orbit transfer vehicles, a lunar base. (A global analysis of the best numbers of ground sites and number of lasers at each, for various target configurations and requirements, has not been performed.) A system of contention resolution and target allocation determines which lasers serve which users.

Lasers. Free-electron lasers are indicated as the best technology to use. Present technology programs are developing RF drive and induction drive lasers. (The drive system produces the electron beam which in turn generates the laser beam.) While the laser is constantly operating when illuminating a target, free-electron lasers are pulsed devices, producing a continuous stream of short pulses of light. The RF-drive laser is more technologically advanced and has a pulse format better suited to photovoltaics.

- Laser power beams from Earth to Moon.
- FEL with adaptive optics for tight beam.
- ~1 micron wavelength matched to receiver photovoltacis.
- Four sites on Earth for continuous view to Moon.
- Two sites are available for electric orbit transfer power beaming.
- Power per unit array area ~ ten times natural sunlight.

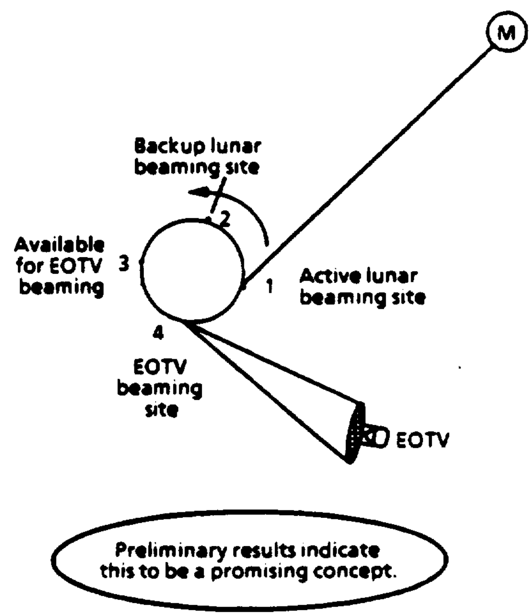

Figure 2.1. Laser Power Beaming Concept

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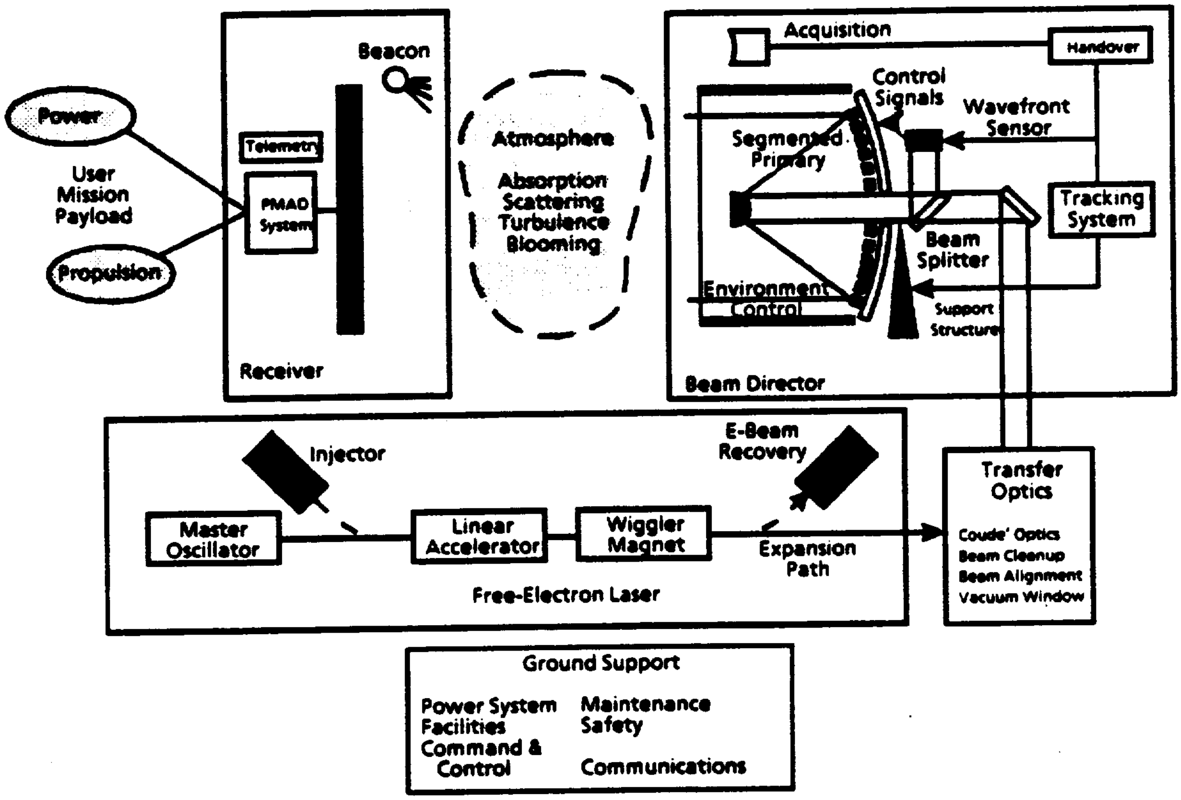

Figure 2.2. Laser Power Beaming Concept Elements

The induction laser produces larger pulses which saturate the photovoltaics, resulting in low efficiency. Unless this problem can be solved, the RF drive laser will be the best technology for this application.

The laser frequency is selected for a good atmosphere window and the photovoltaic material of the receiver. The present baseline uses gallium arsenide photovoltaics and a laser wavelength of 0.85 microns. Greater than 50% conversion efficiency from laser light falling on the array to electricity is estimated, and has been demonstrated with low-power continuous-wave lasers.

Beam Director. The laser beam is expanded to fill a beam director aperture of about 10 meters. A diffraction-limited 10-meter aperture can project a parallel beam to approximately geosynchronous altitude. Atmospheric turbulence, if uncompensated, will cause the beam to dissipate so as to be unusable. Therefore, the baseline includes adaptive optics to compensate for turbulence. The baseline concept segments the primary beam expander reflector into hexagonal elements about 5 cm. in size; there are about 50,000 elements. These are controlled by a massively parallel computing system, based on signals derived from a laser beacon at the target (or a synthetic beacon generated by illuminating sodium in the upper atmosphere), and a wavefront sensor. The target beacon is displaced from the target in the direction of flight so that the beacon signal traverses the same patch of atmosphere the main beam traverses on its way to the target. The light travel time, beacon down to main beam up, through the lower 10 to 15 km of atmosphere where the turbulence is, is about 0.1 msec. This is short enough to "freeze" the turbulence motion so that the beam is well-corrected. Turbulence compensation has been experimentally demonstrated.

Electric Orbit Transfer Vehicle. The vehicle has an array size of about 13 meters. Based on maximum useful intensity, this array can generate up to about 450 kWe; less may be cost-optimal. The photovoltaics are passively-cooled by thermal radiation from both sides. The array specific power available is so high, over 1 kWe/kg, that electric propulsion technology is challenged to come up with a compatible thruster technology. Ion thrusters, while very efficient, are heavy and require much power processing and conditioning. The specific power for the power processing and thruster system for the NASA SEPS design circa 1980 was about 0.07 kWe/kg. Higher power systems with more modern technology might reach 0.1 to 0.2, still much less than the laser-powered array. The most promising new thruster technology for high specific power is the pulsed plasma thruster, with the Russian Hall thruster also a possible contender. While the pulsed plasma technology is speculative at this time, it was baselined with estimated efficiency of 50% and power processor/thruster mass of 1 kg/kWe, comparable to the array. With addition of structure and other systems needed to complete the EOTV, a specific mass of about 5 kg/kWe (not including payload) is estimated. The selected baseline Isp was 3300 seconds; close to optimal based on the parametric studies conducted. Baseline design parameters are presented in detail in Table 2.1.

2.2 DISCUSSION OF COST LEVERAGES

Typical lunar scenarios, such as the ones in the synthesis group report [2] require annual cargo to Earth orbit in the range of 1.4Mlb per year. This gives a net payload to the lunar surface of 220klb, or 100 tons per year. Of this about 80% is cryogenic propellant for delivery of the useful payload to the Moon. Traffic projections for GEO are less, but cost trends are analogous. An electric-based transportation system with a specific impulse ten times higher then chemical would, in the ideal case, reduce the 80% propellant to more like 3%, reducing the Earth Launch transportation requirement to 23% of 1.4 Mlb, or 322klb. From the equation above, at 1.4 Mlb/year we would expect launch to Earth orbit costs of $2.88 billion per year. At 322klb/year we would expect costs of $1.08 billion per year, providing a launch cost savings of $1.8 billion per year.

Table 2.1. Parameters of Reference Laser Power Beaming Concept

• Reference EOTV obtained by optimizing life cycle cost for GEO mission			
• <u>Thruster</u>			
Type			SRL plasma thruster
Specific impulse			3300s
Efficiency			50%
Mass flow rate			0.42 g/s
Thrust			13.6N
Input power			440kW
Specific mass			0.5kg/kW
• <u>Electrical power system</u>			
Busbar and power processing efficiency			95%
Specific mass			0.5kg/kW
• <u>Photovoltaic Array</u>			
Type			Planar GaAs
Cell efficiency @ 300K, 850nm, 12.75 kW/m^2			58.2%
Efficiency change with temperature			-0.13%/K
Equilibrium temperature			527K
Operating efficiency			28.7%
Solar array output power			463 kW
Area fill factor (cell area/array area)			90%
Array output/area			3300W/m^2
Array area			141m^2
Array diameter			13.4m
Array mass/area			4kg/m^2
Array mass			562kg
• <u>Vehicle Mass Parameters</u>			
Structure and other systems overhead			10%
Propellant tank inert mass overhead			20%
Propulsive mass ratio one way			1.1889
• <u>Vehicle Mass Statement</u>			
Solar array	*562 kg*		
Power processing	*232*		
Thrusters	*220*		
Propellant tank	*309*		
Structure & other	*132*		
Vehicle inert mass		1455kg	
Propellant mass		1546	
Payload mass		5000	
Initial mass in Earth Orbit			8002kg
• <u>Program Mass Statement</u>			
Fleet inert mass			10,200kg
Propellant			309,000
Payload			1,000,000
Total mass to Earth Orbit			1,319,000
• <u>Mission</u>			
Description			Deliver 5000kg spacecraft to GEO
Type			Vehicle returns intact to LEO
Number of missions per year			20
Low thrust mission velocity			5600m/s
Engine run time			42.6 days
Mission flight time			113.7 days
Refurbish/mission window time			14 days

Table 2.1. Parameters of Reference Laser Power Beaming Concept (Continued)

• Program	
Fleet size	7 vehicles
Design life	10 years
Annual cargo delivered	100 mt
Operations duration	10 years
• Power Beam	
Incident beam intensity at spacecraft	12.75 kw/M^2
Beam energy intercepted by array	85%
Wavelength	850 nm
Total beam power	2.1 MW
Maximum target range	42000 km
Target angular size	320 nrad
Theoretical aperture required	1.33 m
Laser to space transmission efficiency	0.50
• Ground Station	
Quantity	4
Design zenith angle	60°
Spacecraft within useable view of 1 station	0.185
Probability of cloud-free line-of-sight	0.9
On-line probability	0.9
Average duty cycle	0.583
Laser output power	4.2 MW
Laser efficiency	0.10
Laser electric power in	42 MW
Operations staff	40
• Cost Parameters	
Photovoltaics development	$0
Laser development	$0
Spacecraft design cost	$130,000/kg
Photovoltaic array production cost	$70,000/kg
Spacecraft production cost excl. PV array	$20,000/kg
Ground station fixed unit cost	$300,000,000
Ground station marginal cost	$20/W/station
Power cost	$0.3/kWh
Technician cost	$50,000/person/yr
Maintenance burden	5% of acquisition/year
Earth-to-orbit launch cost	$5,000/kg
Payload transit time cost	18% of cost/year
• Life Cycle Costs (in million $)	
Spacecraft development	189
Spacecraft production	401
Ground station construction	1,537
Earth-to-orbit launch cost	6,597
Ground station power	259
Ground station maintenance	769
Personnel	80
Payload transit time cost	1,137
Life Cycle Cost	10,969

Over a typical 10-20 year exploration program, the cumulative savings would be in the range of $18-36 billion.

In addition to the direct launch cost savings, a secondary effect will be to lower the cost of the space hardware itself. If laser-powered transfer vehicles lower the delivery cost of payload, then the hardware can be designed less expensively. The optimal design is where the marginal cost of removing weight from the hardware is equal to the delivery cost of the hardware. If the delivery cost is reduced, then less expensive but heavier solutions can be chosen. In some cases, it will no longer be worthwhile to design new hardware for the space application simply to save launch costs. An off-the-shelf item will avoid the development costs of a new item. Other sources of savings would come from use of less expensive materials, and less analysis and modeling work during development to squeeze the last gram out the system weight.

The magnitude of the cost savings for the space hardware is difficult to estimate without a fairly detailed design optimized for each of the transportation costs. One approach is a study of general trends in structures cost with stress level. The specific cost of structure ranging from concrete dams (at the low end) to communications satellites (at the high end) was found to vary with the 5/2 power of the stress. Since structural weight for a given load varies inversely with design stress, then the structure cost goes as the 3/2 power of stress level. Thus for structure, at least, an overall reduction in transportation costs of a factor of 2 should induce a significant change in structure cost. The effects on other subsystems is not as clear at present, but this one example indicates that the response of the payload hardware cost to transportation costs is a potentially major addition to the benefits from beamed power.

2.3 ISSUES FOR ANALYSIS

2.3.1 Traffic Model and Utilization

The traffic model used for these analyses was a nominal 20 deliveries annually from LEO to GEO of 2500-kg spacecraft. This was intended to represent a future world market for commercial, NASA, and military deliveries. The mass represented is typical of the heaviest spacecraft presently delivered to this orbit; most are smaller. The number is about equal to the present worldwide market, including Delta, Atlas, Ariane, and other launchers. Whether a laser-powered system could accomplish so complete a market capture is of course questionable, but if the cost advantages estimated in this report are upheld by more detailed study and development, the potential for near-total capture exists since no other delivery system is cost-competitive. Cost reduction, in addition to capturing most of the market, would increase traffic because the commercial market demand, in particular, is believed to be elastic with respect to price such that a 50% reduction in total GEO satellite cost would lead to more than 3 times the present traffic. Only part of the cost of a GEO satellite is contributed by transportation cost. Significant reduction in transportation cost could, however, enable some reduction in satellite cost because satellites could be re-optimized for the reduction in delivery cost.

2.3.2 Comparative Transportation Costs

The purpose of the parametric analysis was threefold: first, to determine the operating parameters that maximize the economic benefit from laser-powered orbit transfer; second, to perform sensitivity studies on these parameters; and third, to compare the costs and economics of laser-powered operations to more conventional solutions including cryogenic orbit transfer stages and solar-electric propulsion.

A major part of space program cost can be attributed to launch costs to Earth orbit. For example, in a conventional (chemically-propelled) mission to the Moon or to GEO, the majority of the weight launched is propellant, inherently not very expensive, but very costly to deliver to orbit. The great importance placed on the performance (Isp) and weight of space systems is due to the fact that these drive launch weight, which in turn drives cost.

The fundamental leverages of the laser-powered system are (1) versus cryogenic propulsion, a much higher specific impulse, resulting in much lower cost for launch of propellants to low Earth orbit, and (2) versus solar electric propulsion (which also has high specific impulse), much higher output from a given area of photovoltaic array resulting in a combination of lower photovoltaic investment cost, lower cost for launch of the (lighter) orbit transfer vehicle, and reduced electric transfer trip time.

Since launch of propellants to low Earth orbit is a primary factor in the cost and economics comparisons, considerable attention was given to projections of transportation costs to low Earth orbit. Table 2.2 compiles data on launch vehicles, their payloads, and cost at a given flight rate. The specific cost is calculated from these.

Figure 2.3 plots total payload mass (payload x flight rate, in lb/year) vs specific launch cost ($/lb). A clear trend is evident, whose lower bound is marked by the arrow. The outlier point for the proposed ALS is way off the trend line. More recent estimates from the NLS program more closely match the trend. The trend can be modeled as $\$/lb=(2x10^5)/(M/a)^{1/3}$, where M/a is mass per year in lbs. The total annual cost is then $\$2x10^5(M/a)^{2/3}$ per year. Adjusting to 1992$ makes the equation $\$2.3x10^5(M/a)^{2/3}$.

Based on these data, a LEO transportation cost of $5000/lb was used as the reference for the parametric model and for comparison with conventional transportation to GEO.

Table 2.2 Launch Vehicle Characteristics and Costs

Launch Vehicle	Payload (lb)	Flight Rate (per year)	Launch Cost (1988M$)	Specific Cost ($/lb)
Scout	500	4	9	18000
Titan II	5000	5	47	9400
Saturn IB	31000	2	270	8710
Delta 3920	7400	4	44	5948
Atlas Centaur	13200	4	69	5227
Saturn V	249000	2	1090	4370
Shuttle Orbiter	45000	14	190	4222
Titan IV	38000	10	158	4158
Delta II	8700	12	34	3908
Titan III	28800	5	101	3507
Shuttle/ASRM	56000	14	190	3393
Unmanned Orbiter	70000	14	190	2714
Shuttle-C (2-engine)	114000	3	280	2456
Shuttle-C (3-engine)	150000	3	295	1967
ALS (goal)	150000	20	73	487

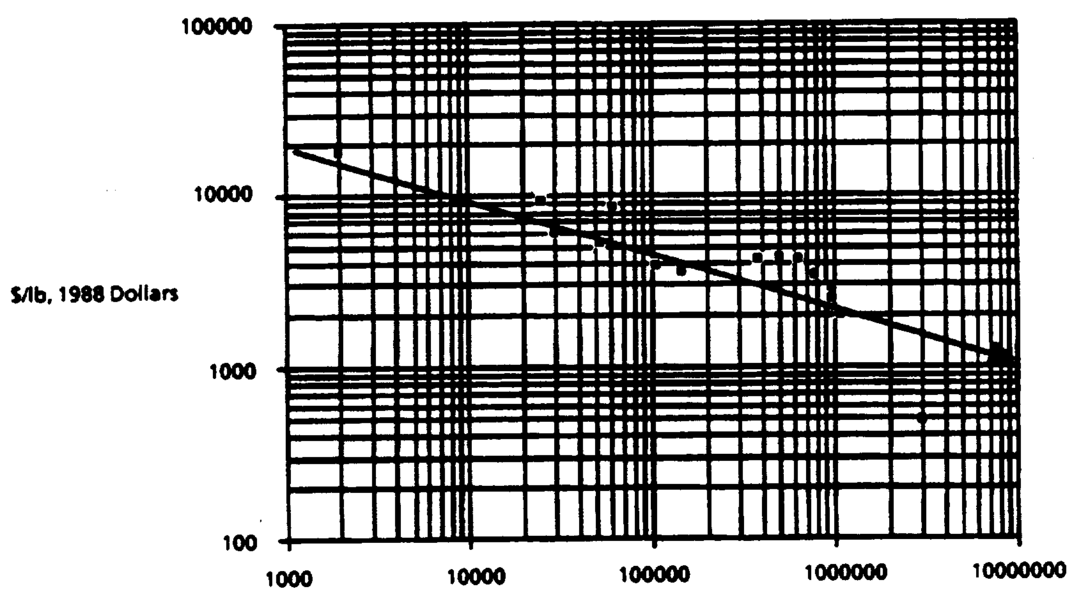

Figure 2.3. Launch System Cost Trends

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2.3.3 Radiation and Array Degradation

Low-thrust orbit transfer operations between low Earth orbit and geosynchronous Earth orbit (LEO and GEO) will experience significant radiation exposure in the vanAllen radiation belts. High-thrust transfers also pass through the belts but spend two hours or less in regions of high radiation dose rates. Low-thrust systems spend much longer, about 1/3 of the total transfer time for continuous thrust, and 1/2 or more for a laser system which has intermittent thrusting periods at low altitudes because a line of sight for power transmission between a ground site and the vehicle does not always exist at low altitudes (see section on simulation results). Typical high-dose times are 15 days for a laser system with total trip time about 30 days, and 60 days for a solar electric system with total trip time about 180 days.

For a payload, which experiences this exposure once, the dose is similar to or less than the typical lifetime dose in geosynchronous orbit. The penalty of additional radiation dose must be traded against the cost savings obtained by laser-powered transfer. In most cases this trade is expected to favor the laser system.

The problem is more severe for the orbit transfer vehicle which may make ten or more round trips through the belts during its operational life. Solar arrays are subject to moderate to severe degradation as a result of this radiation exposure. The transfer vehicle solar arrays should be lightly shielded for maximum power to weight ratio. There is a shielding mass tradeoff, where too little shielding results in severe degradation and too much penalizes performance excessively. The laser system enjoys a significant advantage because (1) its exposure time is less; (2) the high power density permits more shielding; (3) high light intensity raises the operating temperature to a value that is expected to produce self-annealing in gallium arsenide, a likely cell candidate. The annealing temperature for silicon is higher than for gallium arsenide and is in fact so high that silicon cells cannot be operated at a self-annealing temperature.

Figure 2.4 shows experimental data for the degradation of gallium arsenide solar cells as a function of total electron fluence. A fluence of 10^{15} is approximately equivalent to a 60 day transfer from LEO to GEO with continuous thrust. The Boeing test data show greater degradation than the Lewis Research Center data. It is well-known that degradation of photovoltaics increases as performance increases. The Boeing data are from a later time period than the Lewis data and it is surmised that higher-performance cells were tested, explaining the greater degradation.

Self-annealing was assumed in the results presented here whenever the calculated cell operating temperature was at or above the annealing temperature. All the calculations presented are for gallium arsenide photovoltaics. Silicon appears to be a poor choice for laser-powered electric propulsion because its performance decreases more rapidly with temperature increase and it offers little expectation of self-annealing.

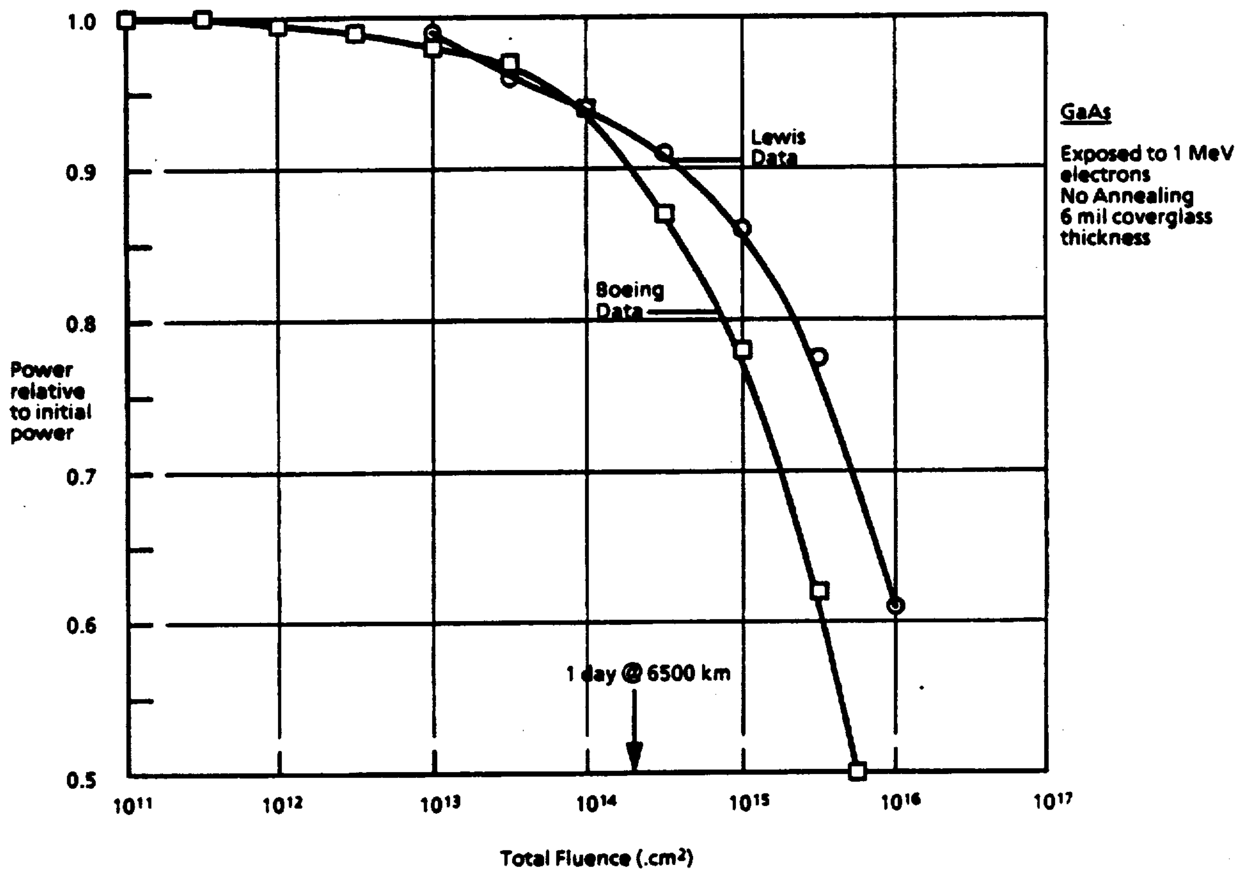


Figure 2.4. Solar Array Power Degradation Due to Radiation AC5048

2.3.4 Duty Factor

The fraction of time that an electric orbit transfer vehicle in a low Earth orbit can be illuminated by a single ground station is less than 1%. With four to six ground stations the initial duty factor is still less than 5%. Figure 2.5 shows cumulative duty cycle for a simple transfer using four ground stations with zenith angle 60 degrees. The cumulative duty factor reaches only about 20% at the end of the missions. (This is the cumulative average. The instantaneous duty factor approaches 100% at GEO altitude but the cumulative average is weighted down by the long duration spent with low duty factor.) A significant part of the present study, as described in Section 3, addressed this issue. By using a greater zenith angle of 70 degrees, more ground sites (6 vs. 4), and a dual Isp strategy, the cumulative average was roughly doubled. With these improvements, the laser system yields attractive LEO-GEO trip times on the order of 30 days with good payload performance. Further improvement is possible through use of relay mirrors.

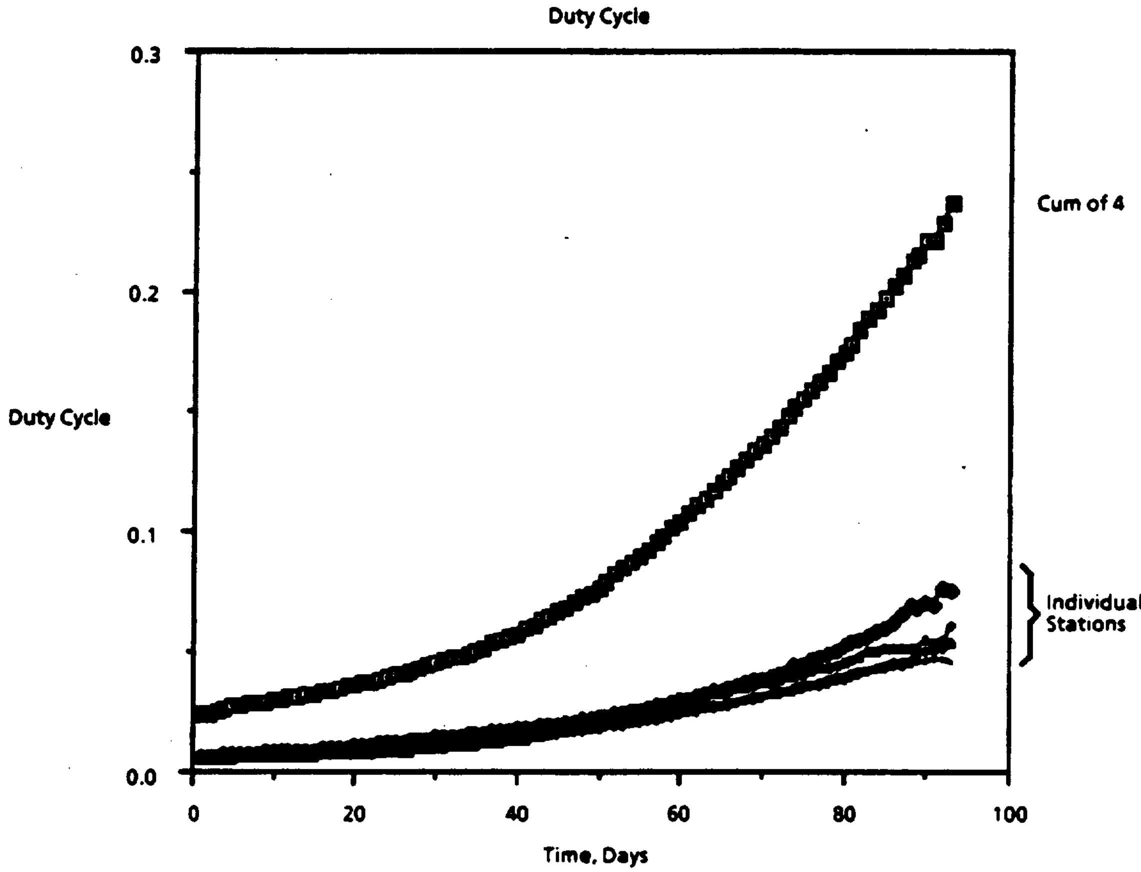


Figure 2.5. Cumulative Duty Factors, 4 Ground Stations and 60° Zenith Angle AC5049

2.4 INTEGRATION OF MODELS

Prior to initiation of this technical directive, JPL had developed a spread-sheet model of the cost and economics of alternative means of supply of electrical power to a lunar base. This model was a relatively detailed simulation of a "lunar electric utility", including the usual utility considerations of peak and average power, multiple sources and uses of power, availability and outage, and investment and operating costs. The Boeing models were similarly detailed simulations of transportation system operations, as described below. Investigation of the model interrelationships revealed that the JPL model considered transportation only as a portion of investment cost, based on cost per unit mass transported, while the Boeing models considered the lunar base only in terms of the mass to be transported. Because of this weak interface, it was more practical to simply integrate results obtained from the models rather than linking or integrating the spread sheets.

A useful integration could be performed in the future. If laser beam power were in use both to supply a lunar base and to power orbit transfer systems, the Earth-based laser beam utility system would be time-shared between the two applications. As illustrated in figure 2.1, some of the ground-based laser sites will be freely available for transportation service, while for others, there may be a contention problem. An integrated simulation which includes both applications, weather outage statistics, and evaluates various methods of resolving contention, would be useful in determining (a) integrated economic benefits, (b) the economic trade between more ground sites and electric power storage at the lunar base, and (c) the most economic and reliable algorithms for resolving contention. This needs to be a time-dependent simulation and probably should not be implemented on a spread sheet, but instead as an extension of the Boeing beam power orbit transfer simulation implemented in C++.

2.5 APPROACH; CONSTRUCTION OF MODELS

The general structure of the model used for parametric analysis of orbit-to-orbit transportation using laser beam power is illustrated in figure 2.6. The model builds up performance beginning at the electric thruster, adds in spacecraft power system factors, then estimates spacecraft mass, then estimates duty cycle and mission duration, calculates laser performance, and finally builds up to program performance and cost factors.

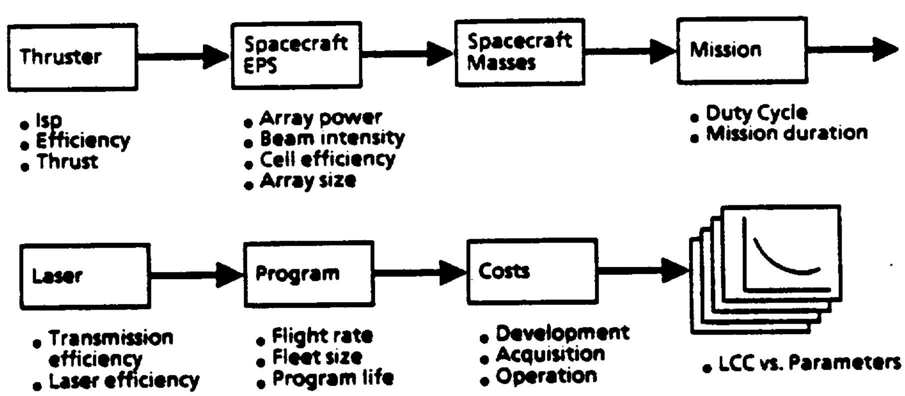


Figure 2.6. Electric Orbit Transfer Vehicle Parametric Analysis Model

The model was implemented on an Excel (TM) spread sheet operating on a Macintosh (TM) computer. An extensive array of parameters are available for user choice; parametric studies were performed by varying the user-choice parameters. Table 2.3 presents a listing of the model and the user inputs. Parameters alterable by the user for parametric studies are outlined in the table. Not all of these were varied for the present studies; those that were are listed in Table 2.4. Expressions used for calculations are also shown in Table 2.3 in the usual spread-sheet notation. A general summary follows:

The model sizes the laser systems by user selection of illumination intensity on the photovoltaic array, then determining the laser sizing and performance needed to achieve that intensity. The overall flow of the model is as follows: (1) User designates Isp, efficiency, and mass flow rate. Model calculates thrust and thruster busbar power. (2) User designates laser beam intensity (W/sq m). Model calculates array unit output based on thermal balance and p/v temperature degradation. Model calculates array size based on thruster power and unit output. (3) User designates mass estimating factors such as array mass per unit area, and mission factors such as payload and mission delta V. Model calculates vehicle size and initial mass in Earth orbit. Model calculates thruster run time, estimates duty factor based on number of ground stations and some contention factors, calculates trip time and average number of ground stations operating. (4) User inputs laser parameters. Model calculates laser aperture and power. At this point the model is ready to calculate programmatic and evaluation parameters.

2.6 Selection of Parameters and Ranges of Values

A few of the more important reference parameter selections are as follows: Wavelength, thruster selection and overall power level and system sizing, and reference LEO transportation cost were described above. Gallium arsenide photovoltaics were selected because of availability of a suitable atmosphere transmission window, high performance and maturity of the photovoltaic technology, and its ability to perform at elevated temperatures. Payload of 5000 kg was selected as a baseline for parameter studies, but a traffic model of twenty 2500 kg payloads per year was later selected for programmatics analysis. The baseline trip time was 114 days, but parametric studies emphasized shorter trip times. Although these are not rigorously cost-optimal, payload owners are expected to select trip times shorter than the rigorous optima because the incremental cost is small and risk is reduced with shorter trips.

2.7 Results

Parametric results presented here are variations on parameters about an initially optimized point. Each parameter is varied individually. The resulting parametrics are not optimal, since whenever a particular variable such as annual traffic is varied, other variables could be readjusted to give an optimal cost. Thus what is presented here is a sensitivity study, without each point being globally optimized. Also, there were certain differences between the parametric results presented here and the simulation studies:

Table 2.3. Parametric Model Equations

	A	B
1		
2	**Spacecraft**	
3	Thruster Performance	
4	Isp(s)	3300
5	Exhaust Velocity (m/s)	= 9.80665*B4
6	Efficiency	0.5
7	Mass flow rate (kg/s)	0.00042
8	Thrust (n)	=B7*B5
9	Jet Power (W)	=0.5*B7*B5*B5
10	Thruster Busbar Power (W)	=B9/B6
11		
12	Spacecraft EPS	
13	Busbar & Power Proc. Efficiency	0.95
14	Array Output Power (W) (avg)	=B10/B13
15	Incident Beam intensity (W/m²)	12750
16	Cell Type	Planar GaAs
17	Efficiency at 300K	=0.026*(20 + 2.5*(LOG(B15/1400)))
18	Equilibrium Temp (K)	= 53*(B15*(1-B20) + 700)^0.25
19	Efficiency loss/K	0.0013
20	Cell Efficiency (BOL)	=B17-B19*(B18-300)
21	Array Output (Avg/BOL) (3)	1
22	Array BOL Power (W)	=B14/B21
23	Cell/Array Area Fill Factor	0.9
24	Fraction of Beam Captured	0.85
25	Total Beam Power at Spacecraft (W)	=B14/(B20*B21*B23*B24)
26	Array Power/Area (W/m²)	=B15*B20*B23
27	Area Area (m²)	=B22/B26
28	Array Diameter (m)	=((B27/3.14159)^0.5)*2
29		
30	Masses	
31	Array Areal Mass (kg/m²) (5)	4
32	Power Proc. Specific Mass (kg/W)	0.0005
33	Thruster Mass (kg/W)	0.0005
34	Structure & Other Factor	0.1
35	Tank Factor	0.2
36	Mission Delta V (m/s one way)	5600
37	Mass Ratio One Way	= EXP(B36/B5)
38	(masses in kg)	
39	Solar Array	=B31*B27
40	Power Processing	=B32*B14
41	Thrusters	=B33*B10
42	Tanks	=B35*(B45 + B48)
43	Structure & Other	=B34*SUM(B39:B42)
44	*Dry Weight*	*=SUM(B39:B43)*
45	Return Propellant	=(B37-1)*B44

Table 2.3. Parametric Model Equations (Continued)

	A	B
46	Payload	5000
47	***Mass @ Payload Delivery***	***=SUM(B44:B46)***
48	Delivery Propellant	=(B37-1)*B47
49	*IMLEO*	=B48+B47
50		
51	**Mission**	
52	Propellant Consumed (kg)	=B48+B45
53	Engine Run Time (days)	=(B52/B7)/86400
54	Duty Cycle	
55	>=1 Spacecraft Above Min Elevation	=1-(0.815^B77)
56	>=1 Cloud-Free Line-of-Sight	=1-(0.1^((MAX(5,B80))/5))
57	>=1 Laser On-Line	=1-(0.1^((MAX(5,B80))/5))
58	Laser Target Contention	=MIN(1,5/B80)
59	Ground Station Useful Duty Cycle	=B58*B57*B56*B55
60	Mission Duration (days)	
61	Flight Time	=B53*MAX((B77/B62),1)
62	Average # ground stations running	=B59*B80
63		
64	**Laser**	
65	Max Range to Target (km)	42000
66	Target Angular size (rad)	=B28/(1000*B65)
67	Wavelength (m)	0.00000085
68	Minimum Aperture (m)	=B67/(2*B66)
69	Transmission Efficiencies (4)	0.5
70	Laser Output Power (W)	=B25/B69
71	Laser Efficiency	0.1
72	Laser input Power (W)	=B70/B71
73		
74	**Program**	
75	Annual Cargo (kg/yr)	100000
76	Missions/Year	=B75/B46
77	In-Transit Fleet Size	=B76/(365.25/B61)
78	Refurb/Attach Payload Time (days)	14
79	Actual Fleet Size	=INT(B77*((B78+B61)/B61))+1
80	Ground Stations	4
81	Program Years of Operation	10
82	Vehicle life (yrs)	10
83	Vehicle Production Run	=B79*(B81/B82)
84		
85	**Costs**	
86	PV Array (M$/kg)	0.07
87	Spacecraft DDT&E (M$/kg)	0.13
88	Spacecraft Production excl. PV (M$/kg)	0.02
89	Laser Station @ zero power ($M)	300
90	Laser Station ($/W)	20

Table 2.3. Parametric Model Equations (Continued)

	A	B
91	Launch Cost ($/kg)	5000
92	Busbar Power ($/MJ)	0.03/3.6
93	Personnel Cost/Station/Year (M$)	2
94	Development (M$)	
95	*Spacecraft*	=B87*B44
96	*Cell Development*	0
97	*FEL Development*	0
98	Acquisition	
99	Spacecraft Unit Cost	=(B88*(B44-B39))+(B39*B86)
100	*Fleet Cost*	=B83*B99
101	Laser Station Unit Cost	=B89+(B90*B70)/1000000
102	*Ground Stations Total*	=B101*B80
103	Operations	
104	Fleet Mass, Dry (kg)	=B83*B44
105	Propellant (kg)	=B52*B76*B81
106	Payload Mass (kg)	=B75*B81
107	Total ETO Mass	=SUM(B104:B106)
108	*ETO Launch Cost (\$M)*	=(B107*B91)/1000000
109	Station Duty Cycle	=B62/B80
110	Energy/Station/Year (MJ)	=B72/1000000)*B109*31556925
111	Power Cost/Station/Year (M$)	=(B110*B92)/1000000
112	*Power Cost Total*	=B111*B80*B81
113	*Maintenance Cost*	=0.05*B102*B81
114	*Personnel Cost*	=B93*B80*B81
115	*Payload Transit Time Cost*	=0.0005*B61*B88*B106
116	Life Cycle Cost ($M)	=B95+B96+B97+B100+B102+B108+B112+B113+B114+B11
117	Life Cycle Cost ($/kg)	=B116/(B106/1000000)
118		
119	Notes:	
120	(1) Ignores non-uniform illumination effects	
121	on cell efficiencies (2) Ignores temperature	
122	changes in cells due to Earth's shadow and	
123	passing in and out of beams (3) Cell	
124	temperature high enough for continuous	
125	annealing in LPB case, annealing after trip in	
126	solar case (4) Includes losses from Atmospheric	
127	Transmission and beam director optical train	
128	(5) Includes concentrator at 1 kg/m²	

(1) the parametric studies used a 60 degree zenith angle; simulations showed advantages to 70 degrees; (2) The parametric model was not set up to represent the dual Isp strategy described below; and (3) the parametric model somewhat overestimated the cumulative average duty factor, tending to compensate for not including some of the performance enhancements of the simulations.

Table 2.4. Laser Electric Orbit Transfer Vehicle List of Sensitivity Analyses

Parameters Investigated	Purposes
• Mission - Traffic Level - Payload Size	• Identify which parameters have significant impact on life cycle cost
• Vehicle - Thruster Efficiency - Array Specific Mass - Tank Mass/Propellant Mass - Vehicle Life	• Provide information on value of technology improvements
• Transmission - Transmission Efficiency - CFLOS - Beam Capture - Beam Intensity	• Guide optimization of reference concept
• Ground - ETO Launch Cost - Number of Ground Stations - Laser Efficiency	

2.7.1 Payload Traffic

The model was centered on 100,000 kg per year, 20 trips at 5000 kg. The sensitivity to reduced traffic is shown in figure 2.7. For the economics analyses performed by NASA, 20 trips at 2500 kg were used.

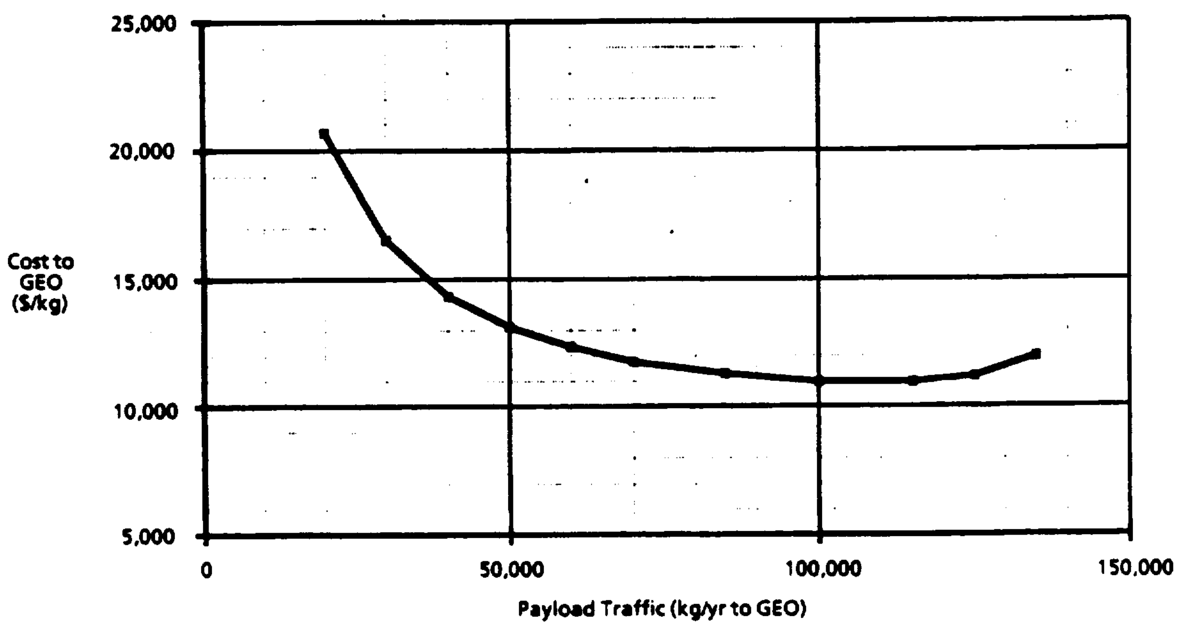


Figure 2.7. Life Cycle Cost vs. Payload Traffic AC5050

2.7.2 Payload per Flight

The system is insensitive to this parameter under the assumption used here which is that total payload delivered annually is constant. The sensitivity is shown in figure 2.8.

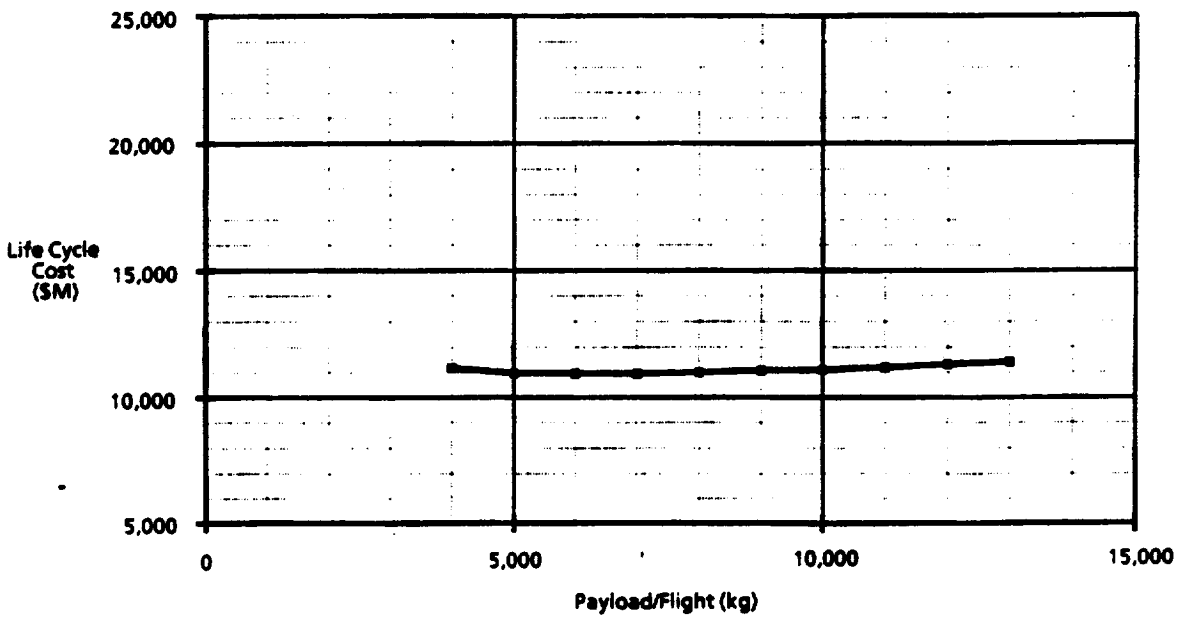


Figure 2.8. Life Cycle Cost vs. Payload Size AC5051

2.7.3 Thruster Efficiency

The benefits of improving thruster efficiency above 50% are modest, as illustrated in figure 2.9, but costs increase rapidly as efficiency drops below 50%. This means that achieving the nominal 50% estimate is important. Experiments with continuous plasmadynamic thrusters have tended to indicate about 30% efficiency. Advocates of the pulsed designs claim higher efficiency but there is not much experimental data. Experimental demonstration of plasma thrusters should receive high priority.

2.7.4 Array Specific Mass

As shown in figure 2.10, the system is not highly sensitive to this parameter because the power per unit area is so high.

2.7.5 Tank Factor

As shown in figure 2.11, the system tolerates relatively poor tank factors, but if hydrogen is the propellant, it must be liquid or supercritical to achieve tank factors in the range of interest.

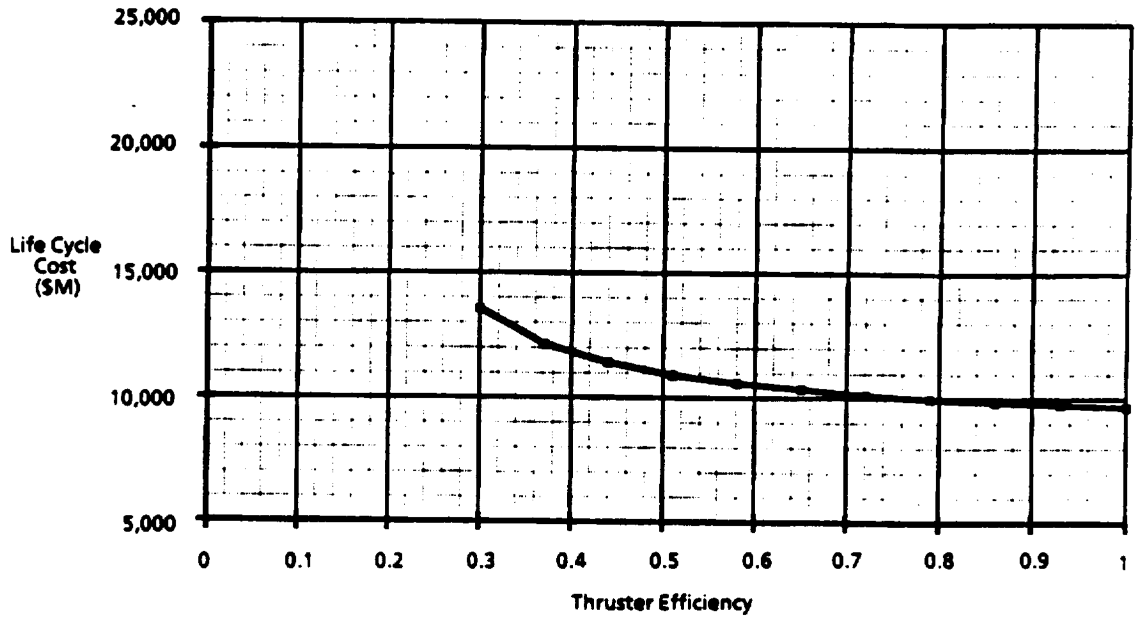


Figure 2.9. Life Cycle Cost vs. Thruster Efficiency. ACS052

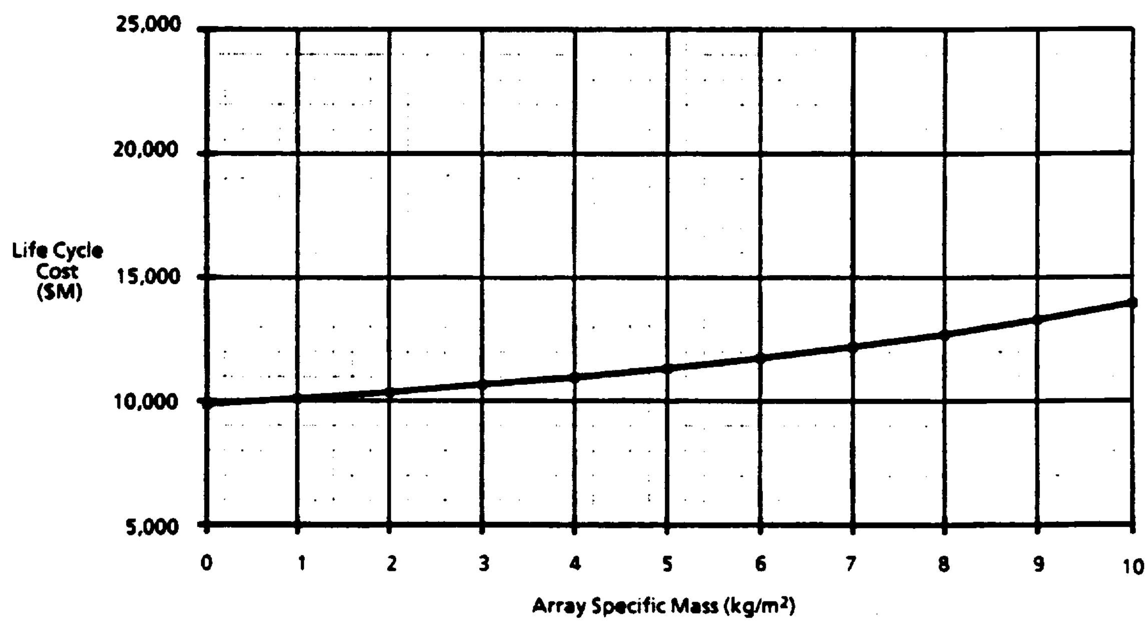


Figure 2.10. Life Cycle Cost vs. Array Specific Mass ACS053

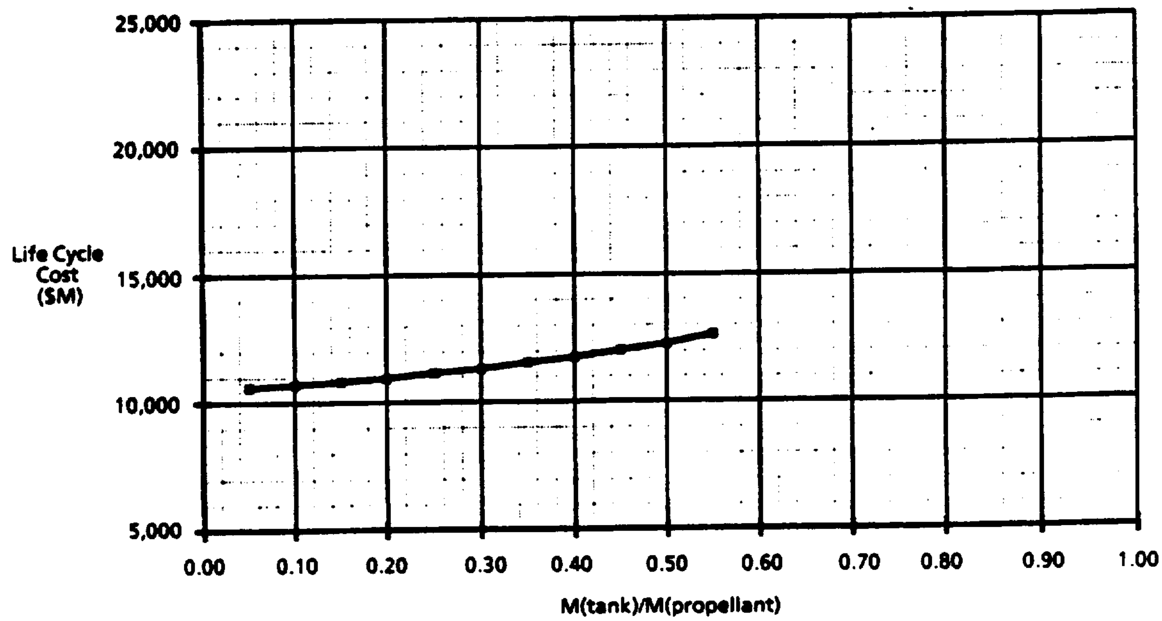


Figure 2.11. Life Cycle Cost vs. Tank Factor AC5054

2.7.6 Vehicle Life

This study used a nominal value of 5 years. There is not much benefit to improving this, but short lifetimes are costly, as shown in figure 2.12. In particular, a single-use vehicle appears only marginally economic. Trades of laser electric versus solar electric systems, not shown here, indicated solar electric systems to be much less economic than laser electric for a single-use system. While the laser system was moderately less costly than a conventional chemical rocket system for single use, the solar system was more costly.

2.7.7 Laser to Space Transmission Efficiency

This parameter is often called Strehl ratio or factor; the system was not sensitive over the range investigated, as shown in figure 2.13. A value of 0.4 to 0.5 is expected.

2.7.8 Probability of Cloud-Free Line of Sight

A nominal value of 90% for any one ground station was used in the parametric analyses. Simulations described in Section 3 did not consider cloud obscuration of the line of sight. It is clear that this is a sensitive parameter, as shown in figure 2.14. Because of the high cost of a ground station, it is economically important to locate ground stations in areas with high probability of cloud-free line of sight. Ground station locations for the simulation studies were selected with this in mind.

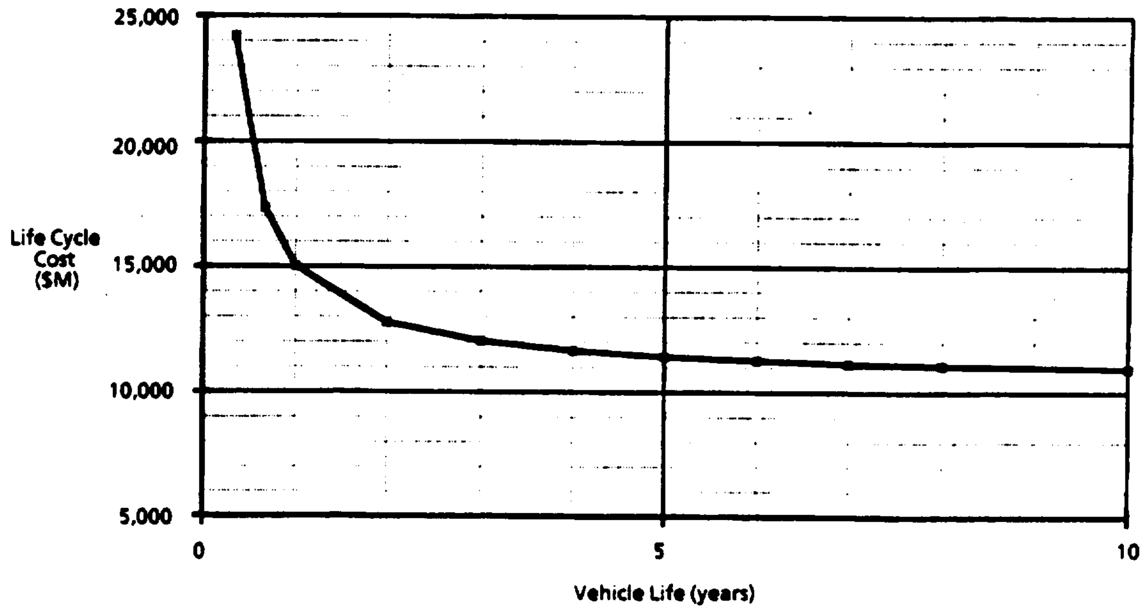


Figure 2.12. Life Cycle Cost vs. Vehicle Life

ACS055

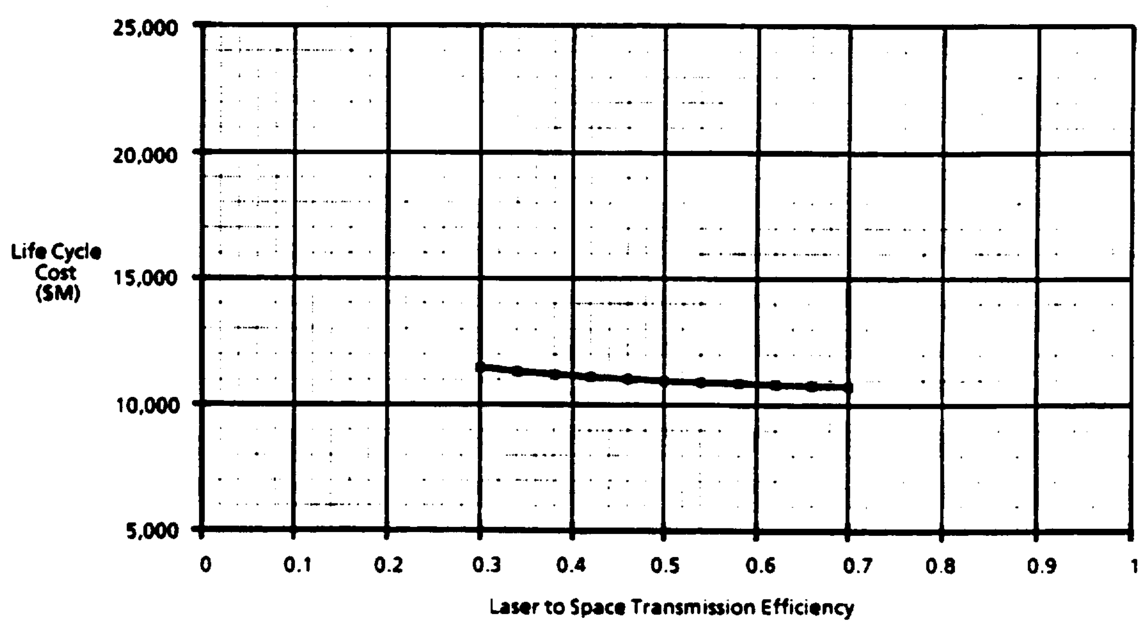


Figure 2.13. Life Cycle Cost vs. Transmission Efficiency

ACS056

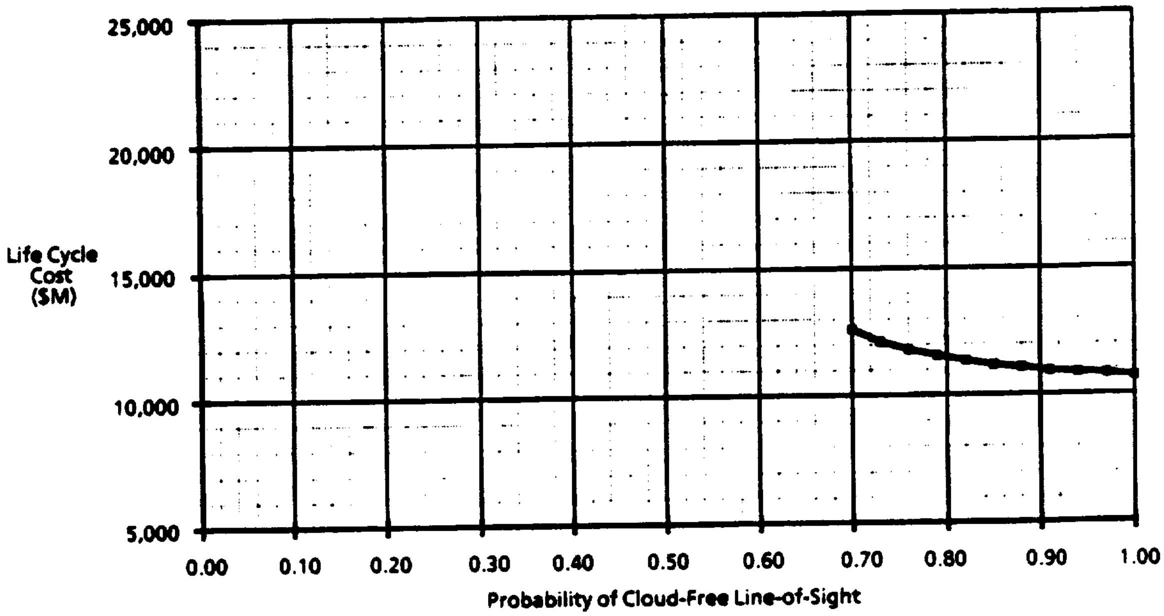


Figure 2.14. Life Cycle Cost vs. Cloud-Free Line-of-Sight

AC5057

2.7.9 Beam Interception

Power not intercepted by the array is lost, but the system sensitivity to this parameter is low, as shown in figure 2.15. This is apparently because the cost of additional power is small compared to the cost of ground station installations and the EOTV spacecraft.

2.7.10 Beam Intensity

The nominal beam intensity was about 10 suns, i.e. 13,500 W/m^2. As shown in figure 2.16, the system is insensitive to this parameter until the beam intensity drops below that needed to operate the gallium arsenide solar array above its radiation damage annealing temperature. The parametric analysis included degradation of array output if beam intensity was below this value. This result indicates the importance of self-annealing of radiation damage. While annealing of radiation damage in gallium arsenide cells has been demonstrated, and the annealing temperature is reasonably well known, laboratory demonstration of self-annealing operation of a gallium arsenide array under radiation fluence with high light intensity should be accomplished.

2.7.11 ETO Launch Cost

As discussed earlier in this report, the high cost of ETO launch is a primary reason for the cost benefit of laser-powered electric orbit transfer. A reference ETO cost of $2270/lb ($5000/kg) was used for the parametric analysis. This is in the "hoped for"

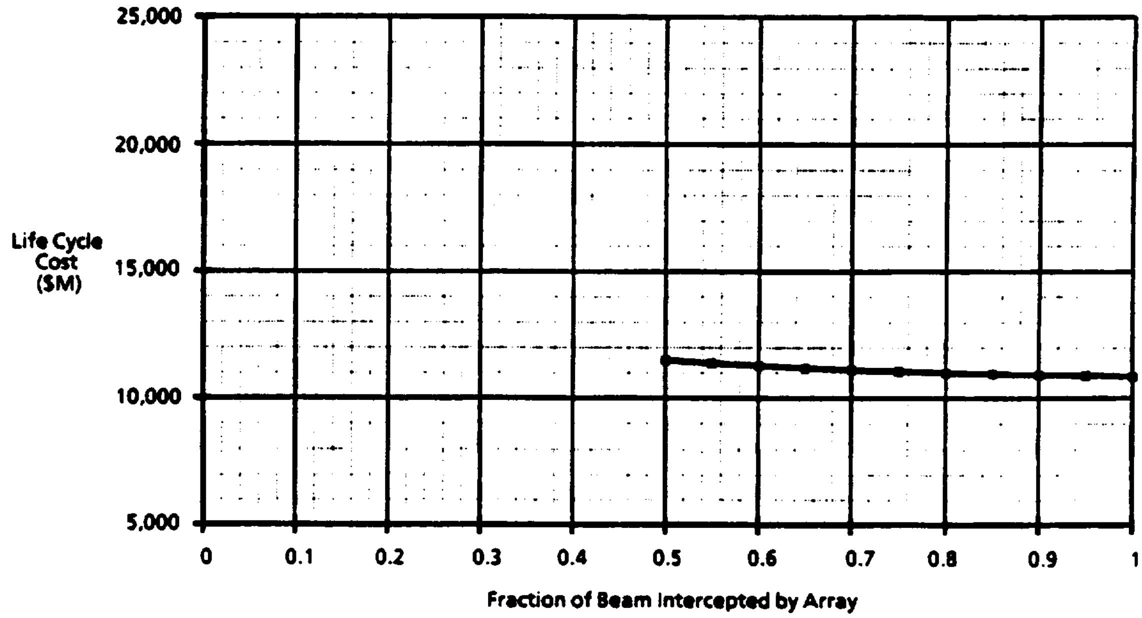


Figure 2.15. Life Cycle Cost vs. Beam Interception ACS058

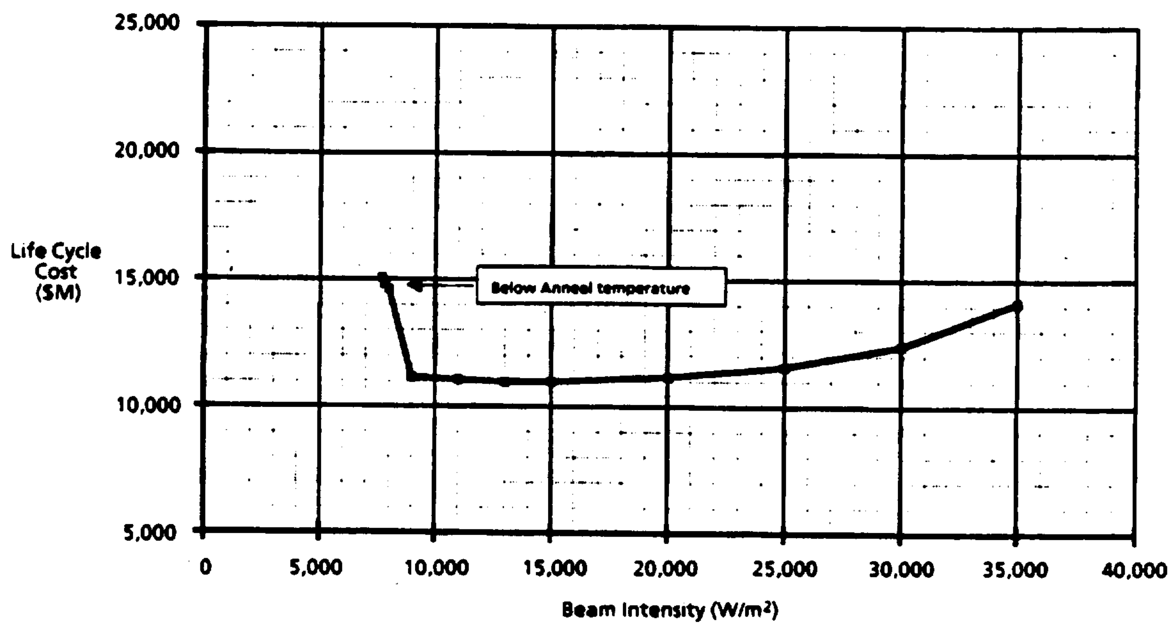


Figure 2.16. Life Cycle Cost vs. Beam Intensity ACS059

range for future launch systems. Today's ETO launch systems operate at $4000 to $5000/lb. The integrated economic analyses performed by MSFC and supported by this study used $5000/lb as a basis for economic comparisons.

As launch cost comes down, the relative contribution of laser ground stations to the life cycle cost of transportation increases. The total ETO delivery mass for conventional cryogenic propulsion orbit transfer is about 3 times that for laser electric propulsion. If ETO delivery mass becomes inexpensive enough, the cost advantage of laser electric propulsion may disappear, depending on the relative cost of space operations for electric versus cryogenic vehicles. This sensitivity curve in figure 2.17 shows the cost advantage disappearing at ETO costs between $1000 and $2000 per kg. Where operations costs may be a large contributor, caution is warranted in interpreting results of this analysis since operations costs were not well modeled.

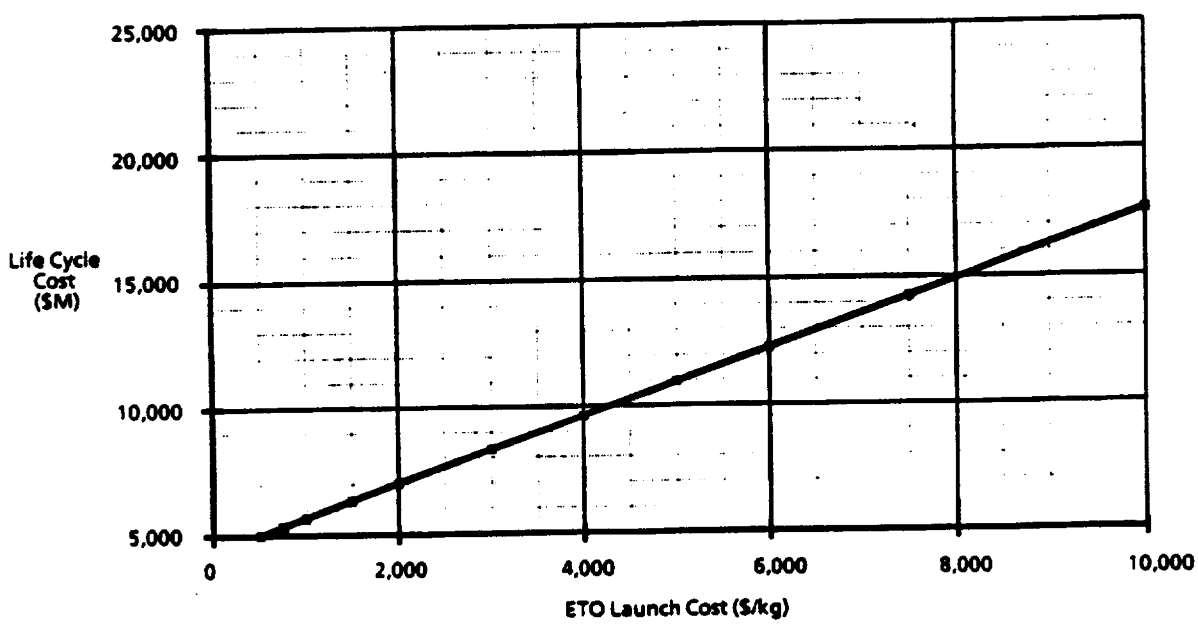


Figure 2.17. Life Cycle Cost vs. Earth to Orbit Launch Cost ACS060

2.7.12 Number of Ground Stations

The number of ground stations is shown in figure 2.18 as optimum for the baseline number. This result is probably an artifact of optimizing the system for that number of ground stations. The question of the best number of ground stations is complicated by the potential use of relay mirrors, alternate uses of the laser power such as power for a lunar base, and with cloud-free line-of-sight statistics.

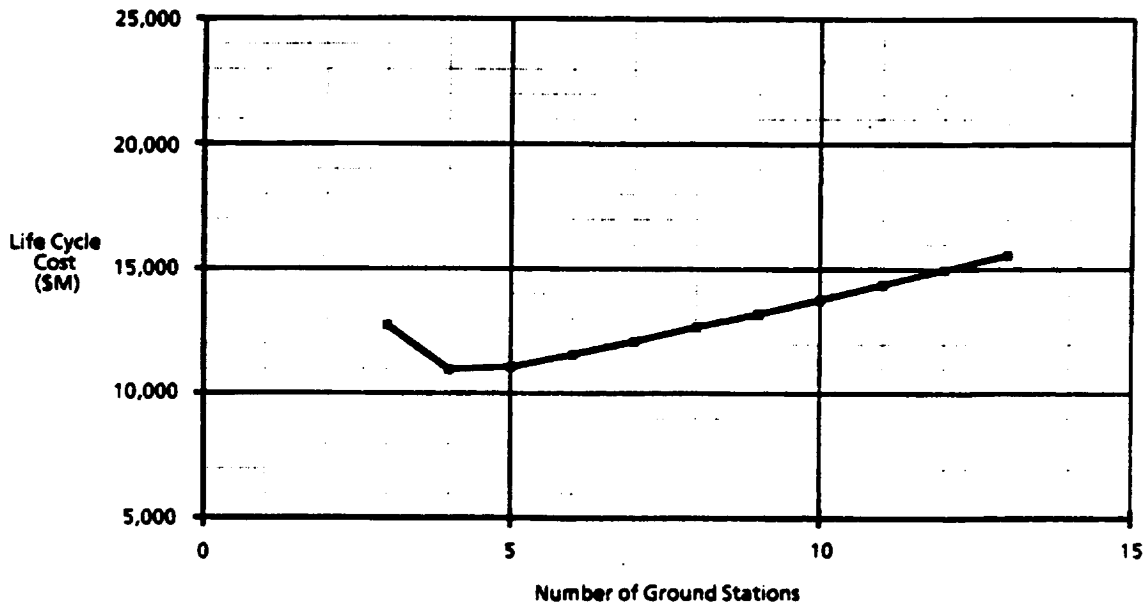


Figure 2.18. Life Cycle Cost vs. Number of Ground Stations AC5061

2.7.13 Laser Efficiency

As shown in figure 2.19, the system is insensitive to this parameter until the laser efficiency falls to very low values, because the cost of electric power is not a major contributor to system life cycle cost if laser efficiency is 5% or better.

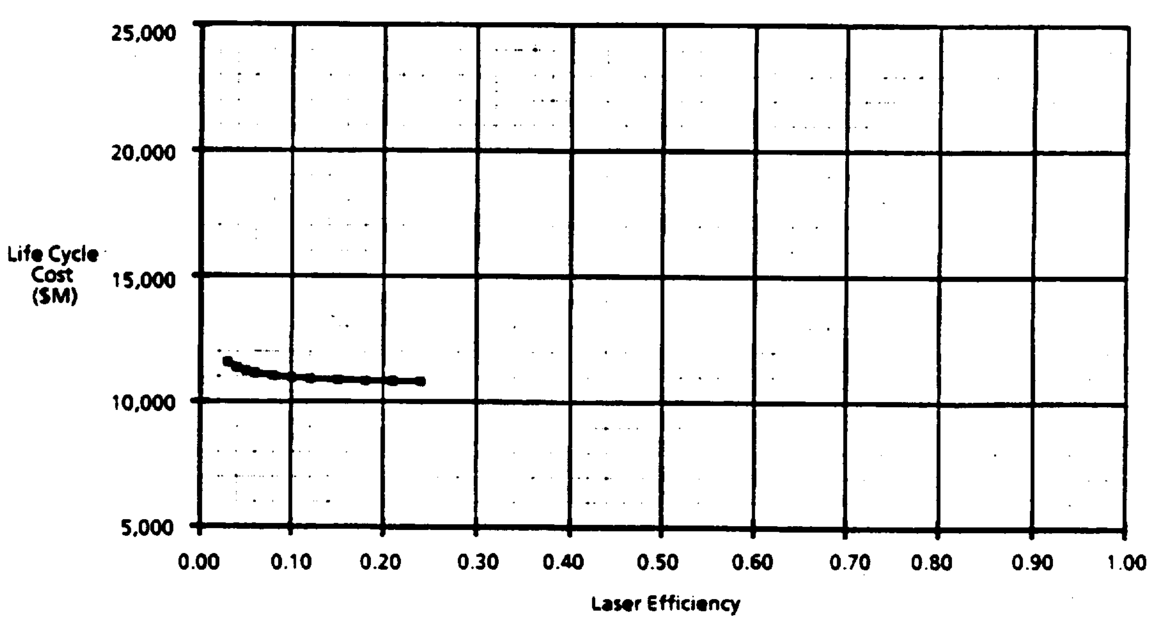


Figure 2.19. Life Cycle Cost vs. Laser Efficiency AC5062

2.7.14 Laser - Solar EOTV Comparisons

Figures 2.20 and 2.21 compare laser and solar EOTVs as a function of trip time and Isp. Tables 2.5 and 2.6 are spread-sheet outputs for some of these cases. The laser system can accommodate much shorter trip times without severe cost penalty. The dual Isp strategy described below enables trip times as short as 30 days at reasonable cost. The long trip times necessary for economical solar electric EOTV operation are seen as a major advantage for laser electric operation.

The laser system optimizes at somewhat higher Isp, but neither system is particularly sensitive to Isp.

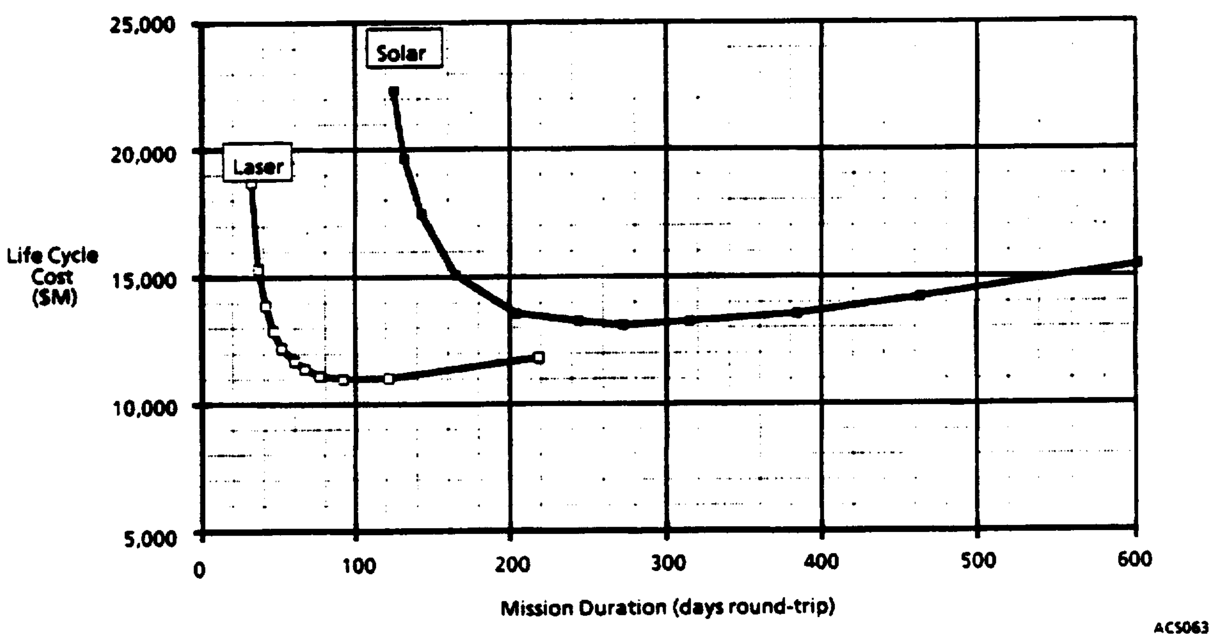


Figure 2.20. Life Cycle Cost vs. Trip Time for Laser and Solar Electric Orbit Transfer Vehicles

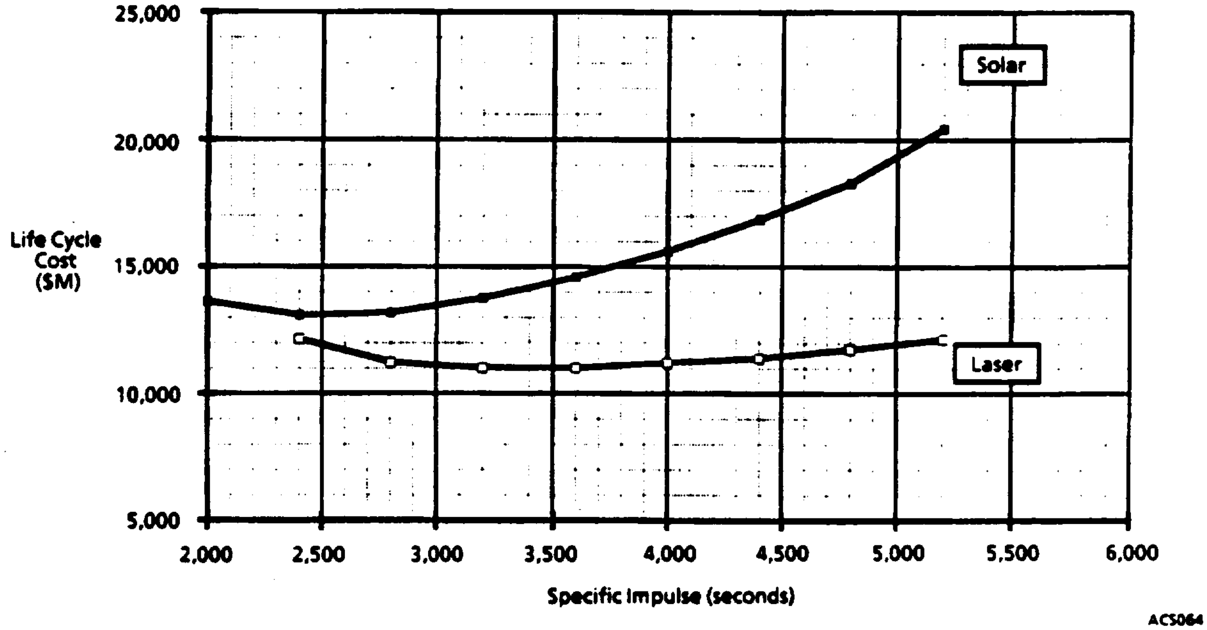


Figure 2.21. Life Cycle Cost vs. Specific Impulse for Laser and Solar Electric Orbit Transfer Vehicles

Table 2.5. Tabulated Sensitivity Data

Sensitivities of Life Cycle Cost To:											
Specific Impulse (sec)	1,200	1,600	2,000	2,400	2,800	3,200	3,600	4,000	4,400	4,800	5,200
1. Solar Electric, other parameters constant at optimum values for 2400s	20,220	15,134	13,626	13,085	13,197	13,769	14,598	15,644	16,887	18,315	20,455
2. Laser Electric, ditto for 2600s				12,155	11,237	11,032	11,029	11,222	11,404	11,751	12,165
Trip Time (days)											
1. Solar Electric Trip Time:	603.76	463.14	384.36	314.83	272.73	244.45	204.47	165.35	143.26	132.06	125.33
LCC	15,448	14,207	13,541	13,255	13,097	13,260	13,575	15,075	17,451	19,643	22,328
2. Laser Trip Time	218.81	121.79	91.90	76.40	66.77	60.17	51.69	46.45	41.50	38.81	32.36
LCC	11,830	11,032	10,993	11,132	11,405	11,709	12,217	12,876	13,893	15,338	18,708
Laser Sensitivities											
Thruster efficiency	0.3	0.37	0.44	0.51	0.58	0.65	0.72	0.79	0.86	0.93	1
LCC	13,550	12,142	11,416	10,917	10,607	10,370	10,137	9,989	9,867	9,764	9,677
Beam Intensity W/m²	7,700	7,800	8,000	9,000	11,000	13,000	15,000	20,000	25,000	30,000	35,000
LCC	15,083	14,821	14,621	11,147	11,058	10,969	10,988	11,177	11,578	12,402	14,134
Array Mass (kg/m²)	0	1	2	3	4	5	6	7	8	9	10
LCC	9,874	10,120	10,373	10,687	10,969	11,339	11,748	12,200	12,705	13,276	13,945
Tank factor (tank mass/propellant mass)	0.05	0.10	0.15	0.20	0.25	0.30	0.35	0.40	0.45	0.50	0.55
LCC	10,616	10,726	10,843	10,969	11,173	11,324	11,562	11,748	12,034	12,274	12,638
Cloud-Free Line-of-Sight probability	0.70	0.73	0.76	0.79	0.82	0.85	0.88	0.91	0.94	0.97	1.00
LCC	12,608	12,109	11,801	11,559	11,356	11,180	11,088	10,944	70,876	10,817	10,700
Transmission Efficiency (laser to space)	0.3	0.34	0.38	0.42	0.46	0.5	0.54	0.58	0.62	0.66	0.7
LCC	11,479	11,329	11,211	11,115	11,036	10,969	10,913	10,864	10,821	10,784	10,751
Fraction of Beam Captured in Space	0.5	0.55	0.6	0.65	0.7	0.75	0.8	0.85	0.9	0.95	1
LCC	11,505	11,387	11,288	11,205	11,133	11,071	11,017	10,969	10,927	10,889	10,855
Laser Efficiency	0.03	0.04	0.05	0.06	0.08	0.10	0.12	0.15	0.18	0.21	0.24
LCC	11,573	11,358	11,228	11,142	11,034	10,969	10,926	10,883	10,854	10,834	10,818
Payload /Year (kg)	20,000	30,000	40,000	50,000	60,000	70,000	85,000	100,000	115,000	125,000	135,000
$/kg	20,699	16,544	14,325	13,140	12,370	11,748	11,285	10,969	10,973	11,183	11,992
Number of ground stations	3	4	5	6	7	8	9	10	11	12	13
LCC	12,761	10,969	11,066	11,573	12,123	12,692	13,208	13,795	14,386	14,978	15,573
Vehicle Life (yrs)	10	10	10	10	10	10	10	10	10	10	10
LCC	24,201	17,359	15,033	12,775	12,023	11,647	11,421	11,270	11,163	11,082	10,969
Lunch Cost	500	750	1,000	1,500	2,000	3,000	4,000	5,000	6,000	7,500	10,000
LCC	5,032	5,362	5,691	6,351	7,011	8,330	9,650	10,969	12,289	14,268	17,567

Table 2.6. Selected Comparison Points, Laser-Electric, Solar-Electric, and Cryogenic Orbit Transfer Vehicles

	Laser vs. Solar vs. Cryo			Trip Time &
Spacecraft	Case 1	Case 2	Case 3	Case 4
Thruster Performance	Laser	Solar	Cryo	Ion Thruster
Isp(s)	3,300	2,400	480	5,200
Exhaust Velocity (m/s)	32,362	23,536	4,707	50,995
Efficiency	0.5	0.5	1	0.698677141
Mass Flow Rate (kg/s)	0.00042	0.000132	10	0.000325
Thrust (N)	13.59	3.11	47,071.92	16.57
Jet Power (W)	219,932	36,560	110,788,283	422,573
Thruster Busbar Power (W)	439,864	73,120		604,818
Spacecraft EPS				
Busbar & Power Proc. Efficiency	0.95	0.95		0.95
Array Output Power (W) (avg)	463,015	76,969		636,651
Incident Beam Intensity (W/m^2)	12,750	4,200		13,750
Cell Type	Planar GaAs	GaAs Conc.		Planar GaAs
Efficiency at 300K3	0.582	0.194		0.584
Equilibrium Temp (K)	527	431		539
Efficiency loss/K	0.0013	0.0005		0.0013
Cell Efficiency (BOL)	0.287	0.129		0.274
Array Output (Avg/BOL) (3)	1.000	0.473		1.000
Array BOL Power (W)	463,015	162,796		636,651
Cell/Array Area Fill Factor	0.90	0.90		0.90
Fraction of Beam Captured	0.85	1		0.85
Total Beam Power at Spacecraft (W)	2,108,947	0		3,036,957
Array Power/Area (W/m^2)	3,293	486		3,391
Area Area (m^2)	140.60	334.95		187.74
Array Diameter (m)	13.38	20.65		15.46
Masses				
Array Areal Mass (kg/m^2) (5)	4	4.2		4
Power Proc. Specific Mass (kg/W)	0.0005	0.0005		0
Thruster Mass (kg/W)	0.0005	0.0005		0.004973844
Structure & Other Factor	0.10	0.10	0.10	0.10
Tank Factor	0.20	0.20	0.05	0.20
Mission Delta V (m/s one way)	5,600	5,600	4,328	5,600
Mass Ratio One Way	1.1889	1.2686	2.5080	1.1161
(masses in kg)				
Solar Array	562	1,407	0	751
Power Processing	232	38	0	0
Thrusters	220	37	941	3,008
Tanks	309	540	918	534
Structure & Other	132	202	186	429
Dry Weight	*1,455*	*2,224*	*2,045*	*4,722*
Return Propellant	275	597	3,084	548
Payload	5,000	5,000	5,000	13,000

Table 2.6. Selected Comparison Points, Laser-Electric, Solar-Electric, and Cryogenic Orbit Transfer Vehicles (Continued)

Mass @ Payload Delivery	*6,730*	*7,821*	*10,130*	18,270
Delivery Propellant	1,271	2,101	15,275	2,121
IMLEO	**8,002**	**9,922**	**25,405**	**20,391**
Mission				
Propellant Consumed (kg)	1,546	2,698	18,360	2,669
Engine Run Time (days)	42.62	236.59	0.02	95.04
Duty Cycle				
> = 1 Spacecraft Above Min Elevation	0.72			0.62
> = 1 Cloud-Free Line-of-Sight	0.90			0.90
> = 1 Laser On-line	0.90			0.90
Laser target contention	1.00			1.00
Ground Station Useful Duty Cycle	0.583			0.500
Mission Duration (days)				
Flight Time	**113.71**	**295.73**	**052**	**223.24**
Average # ground stations running	2.33			2.00
Laser				
Max Range to Target (km)	42,000			42,000
Target Angular size (rad)	3.186E-07			3.186E-07
Wavelength (m)	8.50E-07			8.50E-07
Minimum Aperture (m)	1.33			1.15
Transmission Efficiencies (4)	0.5			0.5
Laser Output Power (W)	4,217,893			6,073,914
Laser Efficiency	0.10			0.10
Laser input Power (W)	4.22E+07			6.07E+07
Program				
Annual Cargo (kg/yr)	100,000	100,000	100,000	100,000
Missions/Year	20.00	20.00	20.00	7.69
In-Transit Fleet Size	6.23	16.19	0.03	4.70
Refurb/Attach Payload Time (days)	14.00	14.00	14.00	14.00
Actual Fleet Size	7.00	17.00	1.00	5.00
Ground Stations	4	0	0	4
Program Years of Operation	10	10	10	10
Vehicle Life (yrs)	10	10	0.50	10
Vehicle Production Run	7	17	20	5
Costs				
PV Array (M$/kg)	0.070	0.070	0.070	0.070
Spacecraft DDT&E (M$/kg)	0.130	0.130	0.130	0.130
Spacecraft Production excl. PV (M$/kg)	0.020	0.020	0.020	0.020
Laser Station @ zero power ($M)	300	0	0	300
Laser Station ($/W)	20	0	0	20
Launch Cost ($/kg)	5,000	5,000	5,000	5,000
Busbar Power ($/MJ)	0.0083	0.0083	0.0083	0.0083

Table 2.6. Selected Comparison Points, Laser-Electric, Solar-Electric, and Cryogenic Orbit Transfer Vehicles (Continued)

Personnel Cost/Station/Year (M$)	2	0	0	2
Development (M$)				
Spacecraft	189	289	266	614
Cell Development	0	0	0	0
FEL Development	0	0	0	0
Acquisition				
Spacecraft Unit Cost	57	115	41	132
Fleet Cost	401	1,952	818	660
Laser Station Unit Cost	384	0	0	421
Ground Stations Total	1,537	0	0	1,686
Operations				
Fleet Mass, Dry (kg)	10,188	37,802	40,908	23,611
Propellant (kg)	309,287	539,648	3,671,976	205,293
Payload Mass (kg)	1,000,000	1,000,000	1,000,000	1,000,000
Total ETO Mass	1,319,475	1,577,450	4,712,884	1,228,905
ETO Launch Cost ($M)	6,597	7,887	23,564	6,145
Station Duty Cycle	0.583	0.000	0.000	0.500
Energy/Station/Year (MJ)	7.76E+08	0.00E+00	0.00E+00	9.59E+08
Power Cost/Station/Year (M$)	6	0	0	8
Power Cost Total	259	0	0	320
Maintenance Cost	769	0	0	843
Personnel Cost	80	0	0	80
Payload Transit Time Cost	1,137	2,957	5	2,232
Life Cycle Cost ($M)	10,969	13,085	24,654	12,579
Life Cycle Cost ($/kg)	10,969	13,085	24,654	12,579

Notes:

(1) Ignores non-uniform illumination effects on cell efficiencies
(2) Ignores temperature changes in cells due to Earth's shadow and passing in and out of beams
(3) Cell temperature high enough for continuous annealing in LPB case, annealing after trip in
(4) Includes losses from Atmospheric Transmission and beam director optical train
(5) Includes concentrator at 1 kg/m²

3.0 FLIGHT SIMULATIONS OF LASER-ELECTRIC TRANSFER VEHICLES

3.1 SIMULATION APPROACH

The general approach was to simulate the motion of a space vehicle in orbit around the Earth, compute its visibility from selected ground site locations to determine when laser power can be transmitted to the space vehicle, and to simulate powered operation of the vehicle by integration of the motion as affected by electric thrust during those periods when line-of-sight visibility permits power transfer to occur. The simulation includes ground track, pictorial, and parameter trend graphics. These greatly aided engineering understanding of the vehicle/system operation and were central to the insights gained by performing the simulations.

3.2 SIMULATION MODEL DEVELOPMENT

The simulation model was developed on Boeing IR&D. The model was derived from a satellite overflight visibility prediction code by translation into C++ and addition of orbit raising algorithms and graphics. The overflight code contained the orbital motion and ground station line-of-sight algorithms needed to calculate unpowered orbital motion and ground tracks and line-of-sight vectors (direction, elevation angle, distance). Multiple ground sites could be entered. The code had been used successfully to predict visible overflights, so the algorithms were known to be valid. The orbital motion routines used orbital elements rather than Cartesian coordinates. A set of analytical algorithms was available for integrating changes in orbital elements resulting from thrust. These had been used in an earlier electric propulsion orbit raising code no longer in existence. The only remaining item was a thrusting direction law for executing simultaneous plane change and altitude change in an optimal manner. This also existed from another earlier investigation and was incorporated. Code checkout was mainly accomplished by watching the run-time graphics, noting anomalies in the motion or power transfer depictions, and using a debugger to track down the problems. Also, some special cases of duty factors were compared with analytical results, and delta V to geosynchronous orbit from the simulation was compared to the known optimal value for the same altitude and plane change. The code runs on a DOS-type desktop machine. A LEO-to-GEO simulation, including graphics, requires about 10 minutes on a 25 MHz 386 with math coprocessor.

3.3 SIMULATION RESULTS

A typical raw result from simulation is shown in figures 3.1 to 3.7. The simulation code writes plot files compatible with the Macintosh (TM) program "Cricket Graph"

(TM), which was used to produce these graphs. The great proportion of time spent at low altitude is evident. Figures 3.8 and 3.9 show photographs of the on-screen graphics during a typical run.

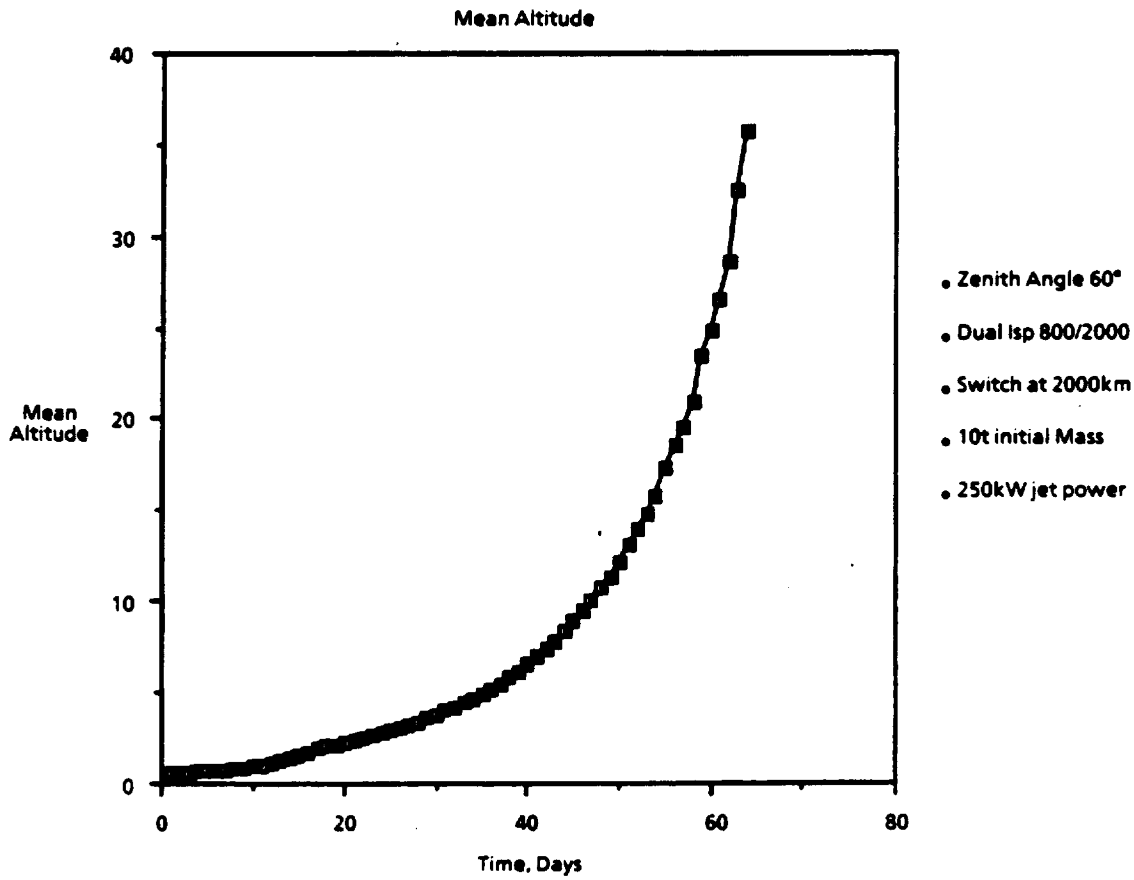


Figure 3.1. Typical Simulation Results - Mean Altitude vs. Time

ACS065

3.3.1 Site Locations and Number

Site locations were selected by NASA on the basis of known low cloud cover climate. Sites included China Lake (in California), Maui, Johnson Island, Bangalore, Alice Springs (Australia), and Morocco. Simulations were run with four and six ground sites. Some of these sites are at about 29 degrees north latitude. While one could obtain a starting orbit at inclination as low as 28.5 degrees launching from KSC, the northerly location of some of the sites resulted in a reduced duty factor at this initial inclination.

Low-thrust orbit transfers expend most of the delta V at lower altitude. Also, the duty factor for line-of-sight power transfer is least at low altitude for geometric reasons. Therefore it is important to not exacerbate the geometry problem at low altitude. Therefore, an initial inclination of 32 degrees was selected. Also, plane change

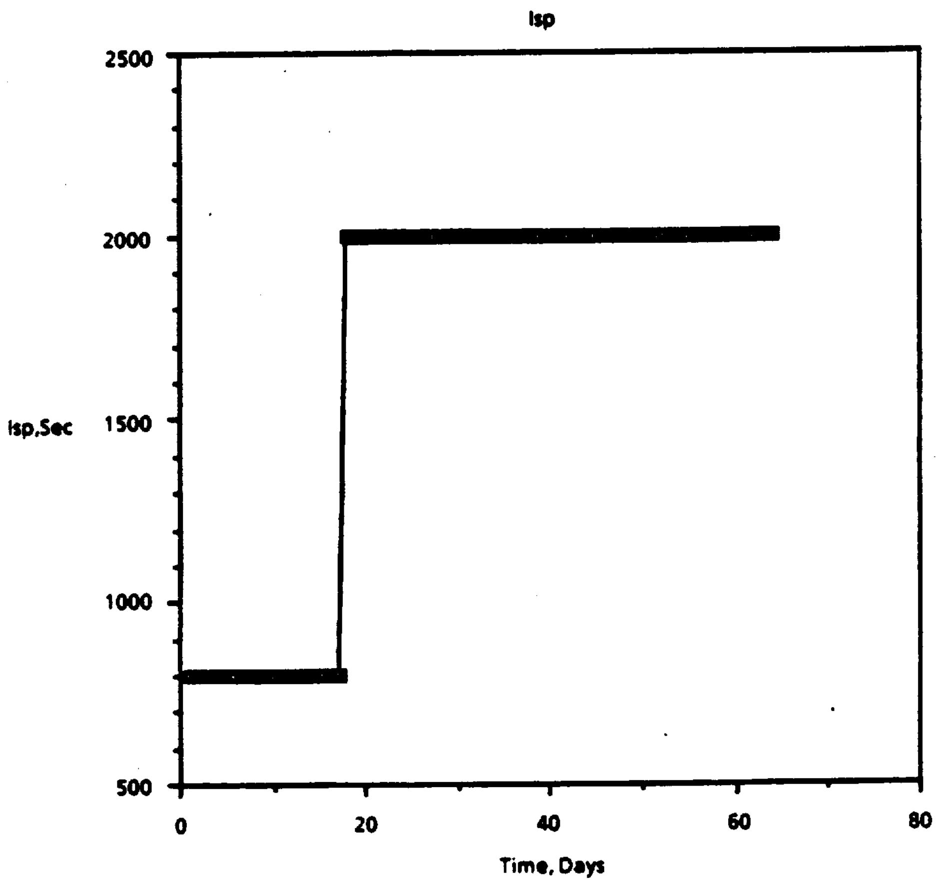


Figure 3.2. Typical Simulation Results - Isp vs. Time ACS066

was not initiated until reaching 3000 km altitude, because otherwise inclination decreased rapidly enough to compromise duty factor, and delta V was spent changing inclination that would otherwise be spent raising altitude and improving duty factor. These choices were not rigorously optimized. They were made by watching the run-time graphics and selecting reasonable values that appeared not to compromise duty factor. The delta V penalties accepted are quite modest in view of the high specific impulse of electric propulsion.

3.3.2 Plane Change

The optimal thrust steering law results in an almost linear decrease in inclination with altitude, as illustrated in figure 3.4 taken from one of the simulations. The actual law used is a simple power law combined with optimal yaw steering around the orbit. It was devised by comparison with a steering law derived by calculus of variations (COV) and selecting power law parameters for best fit; the delta V difference between the power law and COV is about 0.1%. The power law is given in the figure.

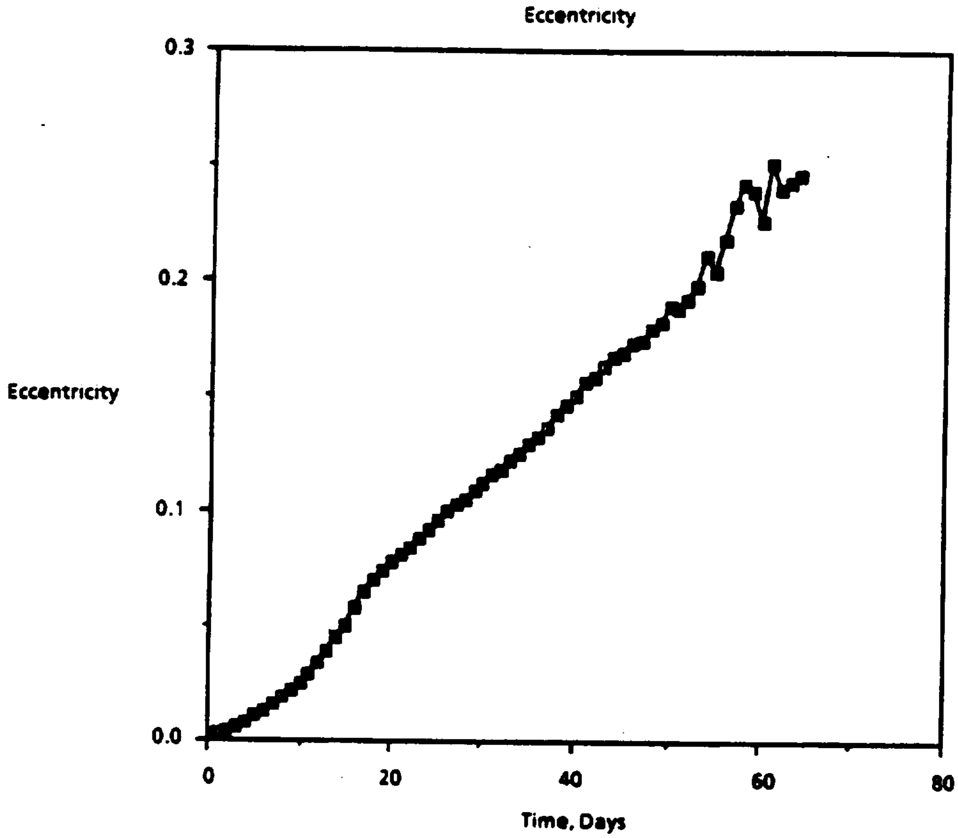


Figure 3.3. Typical Simulation Results - Eccentricity vs. Time

AC5067

3.3.3 Dual Isp Approach

The thrust produced by an electric propulsion system is inversely proportional to the jet velocity if power and efficiency are constant. Since the laser-powered orbit transfer system spends much of its transfer time at low altitudes where the power duty factor is low, it was logical to investigate using a lower Isp at low altitude and switch at some point in the transfer. Use of a resistojet thrust system was chosen for the low Isp portion with an electrodeless plasma thruster for the high Isp portion. This was in part motivated by the fact that a resistojet should deliver high efficiency, about 80%, in conversion of electrical power to jet power, thus producing more thrust per unit power than an option with less efficiency.

The Isp delivered by the plasma thruster and the switch altitude were varied parametrically. The result is that for a particular plasma thruster Isp, a net mass fraction (Mf/Mo) is achieved when the entire transfer is performed at its Isp, and as more and more of the transfer is done at the lower Isp, the mass fraction (mass left at the end of the simulation) decreases, but the trip time does also. A different plasma thruster Isp yields a different mass fraction and trip time for starting point (all thrust at

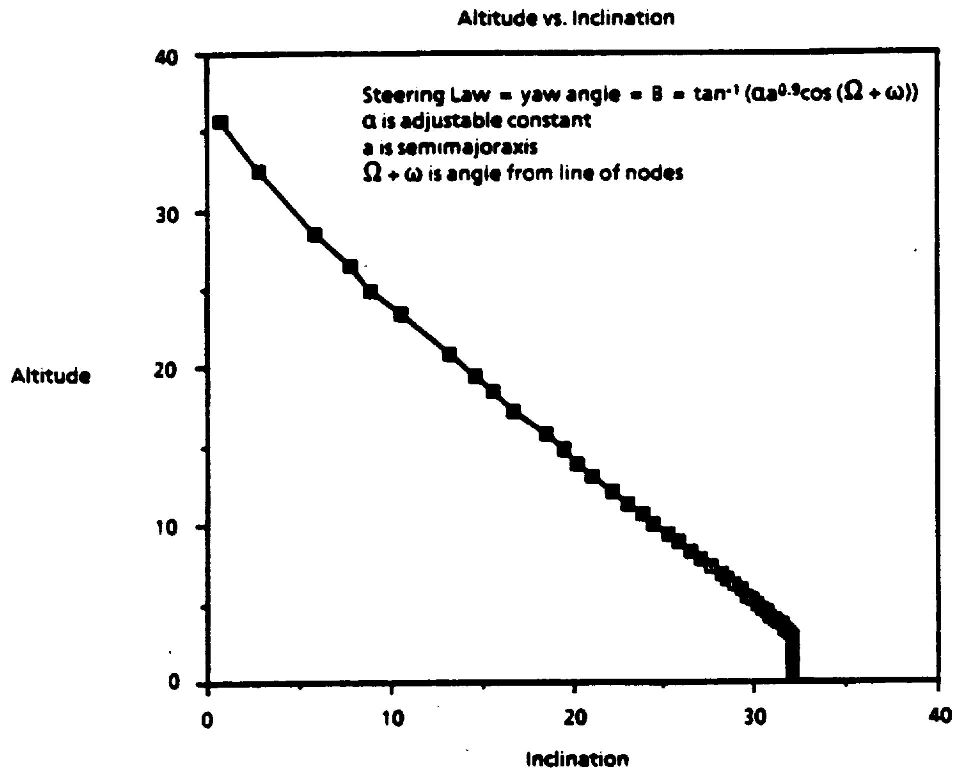


Figure 3.4. Typical Simulation Results - Inclination vs. Mean Radius AC5068

the high Isp), and a different curve of trip time versus mass fraction. The overall result is that for any trip time, there will be an optimal value for plasma Isp and switch altitude that yields a better mass fraction than the best achievable at that trip time by varying plasma thruster Isp without a low-Isp period. This is illustrated in figure 3.10, which also shows effects described in the next section.

This could be carried further: variation in the low Isp value was not investigated; it also is reasonably apparent that the true optimum strategy would have Isp continuously variable; for short trips it may be that the best low-Isp strategy would include a chemical rocket thrusting period which could be continuous. The dual Isp investigation showed that dual Isp is significantly better than a fixed Isp, especially where reducing trip time is concerned.

3.3.4 Integrated Performance Tradeoffs

Figure 3.10 also shows the results of tradeoffs on dual Isp, number of ground sites (four or six) and maximum zenith angle for laser power transmission (60 or 70 degrees). The improvements due to the dual Isp strategy were described above. The effect of number of ground sites is about as expected. Most of the improvement comes from

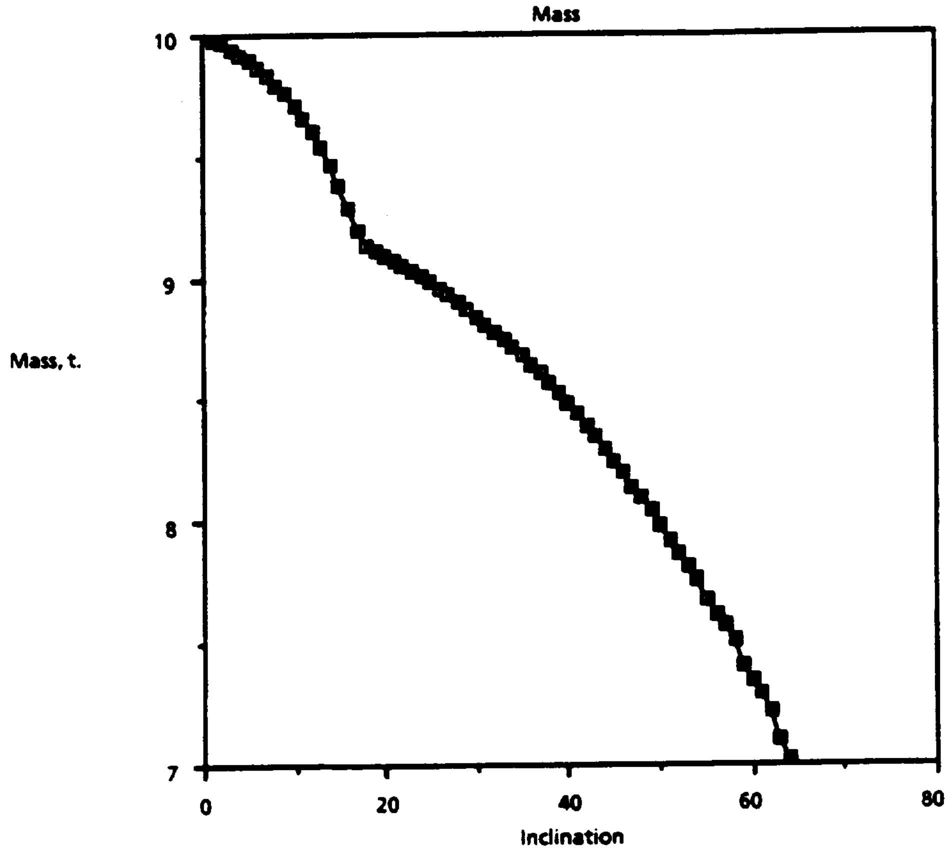


Figure 3.5. Typical Simulation Results - Mass vs. Time

ACS069

increased total duty factor at low altitudes. At high altitudes the coverage circles for the ground sites overlap even with four, and adding more does not increase coverage very much. The low altitude effect is dominant insofar as trip time is concerned, and the trip time with six sites is roughly 1/3 less. The effect of increased zenith angle is striking considering that the increase was only ten degrees. The simulation as presently coded does not include Strehl degradation with zenith angle and therefore slightly overestimates the performance improvement available with higher zenith angle. The improvement shown indicates that use of the higher zenith angle should be implemented even with significant Strehl losses.

3.3.5 Relay Mirrors

The low duty factor for laser power to low Earth orbit vehicles or systems may be improved through use of relay mirrors. Relay mirror analysis was not required by the present statement of work. Relay mirror capability was added to the simulation code as an IR&D task, anticipating a possible need for the analysis capability in the future. Checkout of the simulation provided some initial results, reported here.

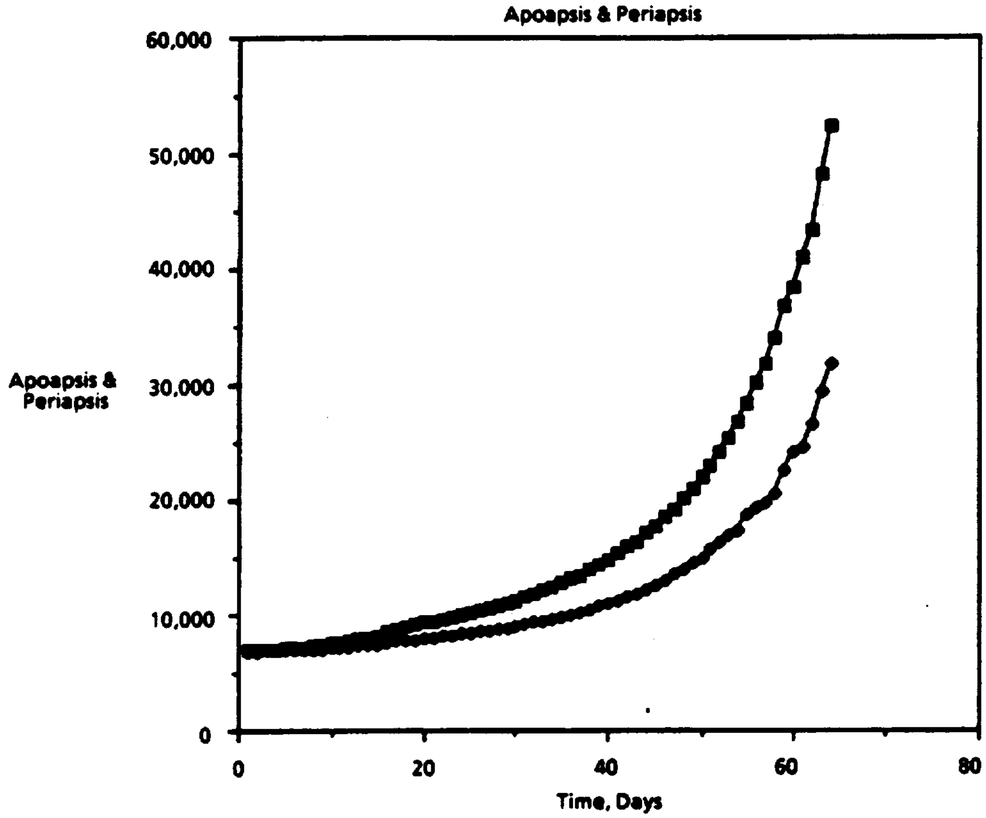


Figure 3.6. Typical Simulation Results - Apoapsis and Periapsis vs. Time ACS070

The relay mirror concept was a part of the SDIO scenarios for use of ground-based lasers against space targets in low Earth orbits, and some technology work was done. As illustrated in figure 3.11, a typical relay mirror operates in an orbit at several thousand kilometers altitude and reflects the laser beam from the ground back towards a receiver in low Earth orbit. The relay mirror may be an optical flat, or may have the capability to focus or defocus the beam slightly. To keep the relay mirror small, it is conceived that the laser beam expander on Earth will focus the beam to about 4 m diameter at the relay mirror, and if the beam at the target needs to be larger than simple geometry dictates, the relay mirror will despread the beam. The required distortions of the relay mirror surface from flatness are only a few microns. The relay mirror is not involved in the adaptive optics process for correction of atmospheric turbulence.

Pointing and tracking is made more complex by use of relay mirrors. Assuming a relay mirror altitude of 10,000 km, its orbital velocity is about 5 km/sec. The light time of flight from the ground is about 0.03 seconds, so the beam director must lead the relay mirror by about 150 m. The relay mirror must be attitude controlled to exactly split the line of sight angles to the ground station and to a point ahead of the receiver travel by about 200 meters (Snell's law). The required attitude accuracy is about 0.01 second of arc.

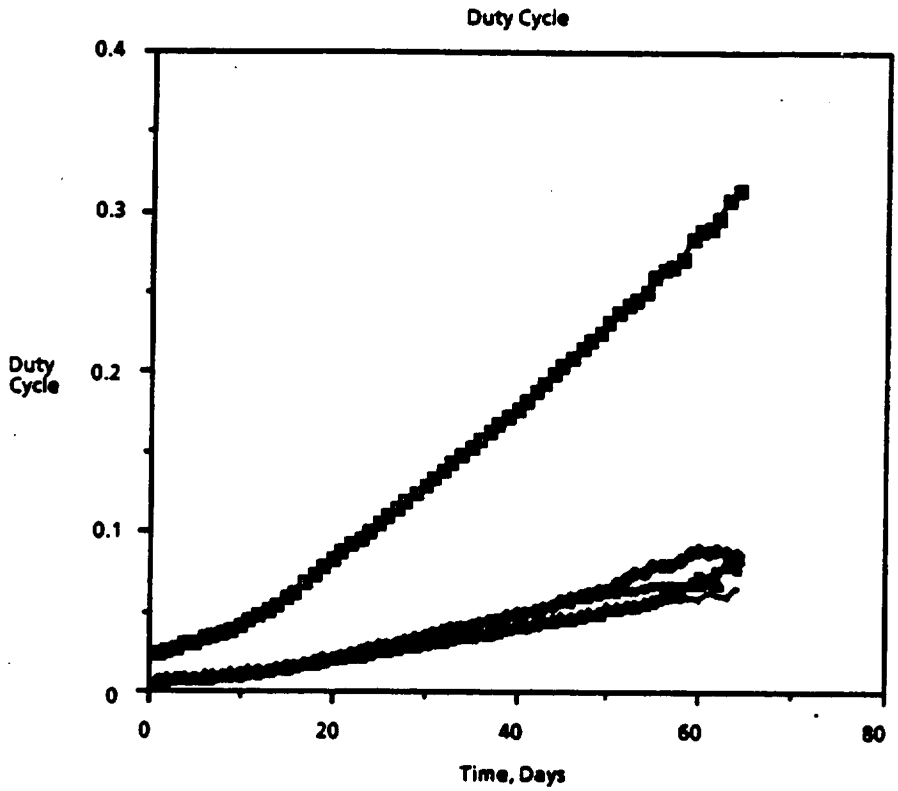


Figure 3.7. Typical Simulation Results - Cumulative Duty Factors vs. Time ACS070

Simulation Approach. The approach is a straightforward extension of the direct laser beaming approach. Instead of generating an orbit path for a single electric orbit transfer vehicle, the simulation generates paths for a chosen number of relay mirrors plus the orbit transfer vehicle. The relay mirrors are assumed to be unpowered insofar as thrust. The simulation tests each of the relay mirrors for visibility from the set of ground sites. If a line-of-sight connect is found, the simulation then tests for line-of-sight connect from the relay mirror to the orbit transfer vehicle. If a connect is found, the beam is "turned on" and the search stops. Otherwise, the search continues until all ground site, relay mirror and vehicle combinations have been tested. If no connect is found, the beam is "turned off". When the beam is "on", the thrusting algorithm is applied to the transfer vehicle. The simulation graphs the positions of the mirrors and vehicle in a selected one of several displays and draws in the beam when it is on, as shown in figure 3.12 and 3.13. It is possible to restrict the "beam on" condition to occur only when the beam returned from the relay mirror will not strike the Earth. This permits a comparison of this operating mode with an unrestricted mode. The restricted mode may be required to prevent reflection of highly collimated laser beams to Earth where the beam might cause eye damage to anyone who stared into it.

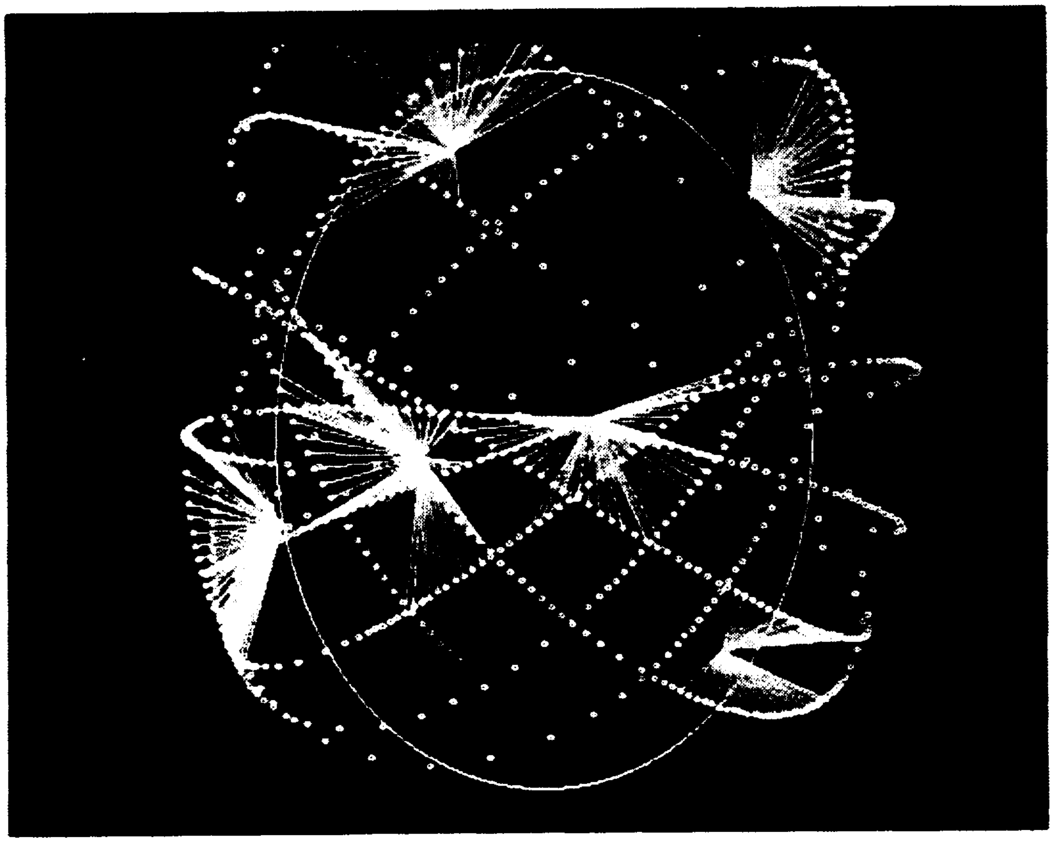


Figure 3.8. Computer Screen Photo of Simulation Graphics

At some orbit transfer vehicle altitude the duty cycle will be better in direct mode than in relay mirror mode. The simulation enables the user to specify a switchover altitude for switching from relay mirror mode to direct mode to test this.

Preliminary Results. Two checkout cases were run: (1) relay-to-direct switchover altitude for a relay mirror altitude of 5000 km with 3 relay mirrors at 4000 km altitude and 6 ground sites, and (2) duty cycle for laser power to Space Station Freedom orbit as a function of relay mirror altitude. Both cases used restricted "beam on" such that the laser beam was not returned to Earth. Results are shown in figures 3.14 and 3.15. The relay mirror case improved orbit transfer trip time by about 10% over the non-relay-mirror case (i.e. the switchover altitude is 500 km, that is, immediate). This is significant in that a 100% duty factor would reduce the trip time by less than half. The Space Station Freedom case illustrates that the best relay mirror altitude is above 10,000 km. These results indicate the capability of the simulation to explore the effects of the many variables involved in relay mirrors operating in conjunction with laser beaming.

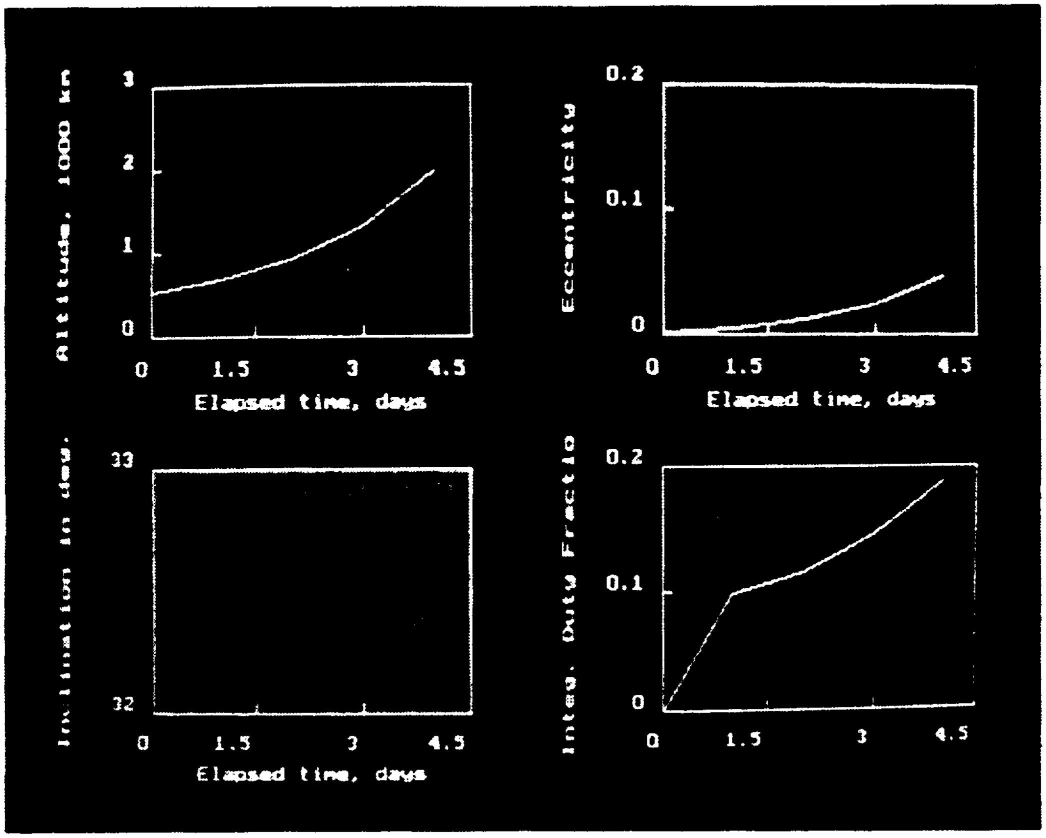


Figure 3.9. Computer Screen Photo of Simulation Status

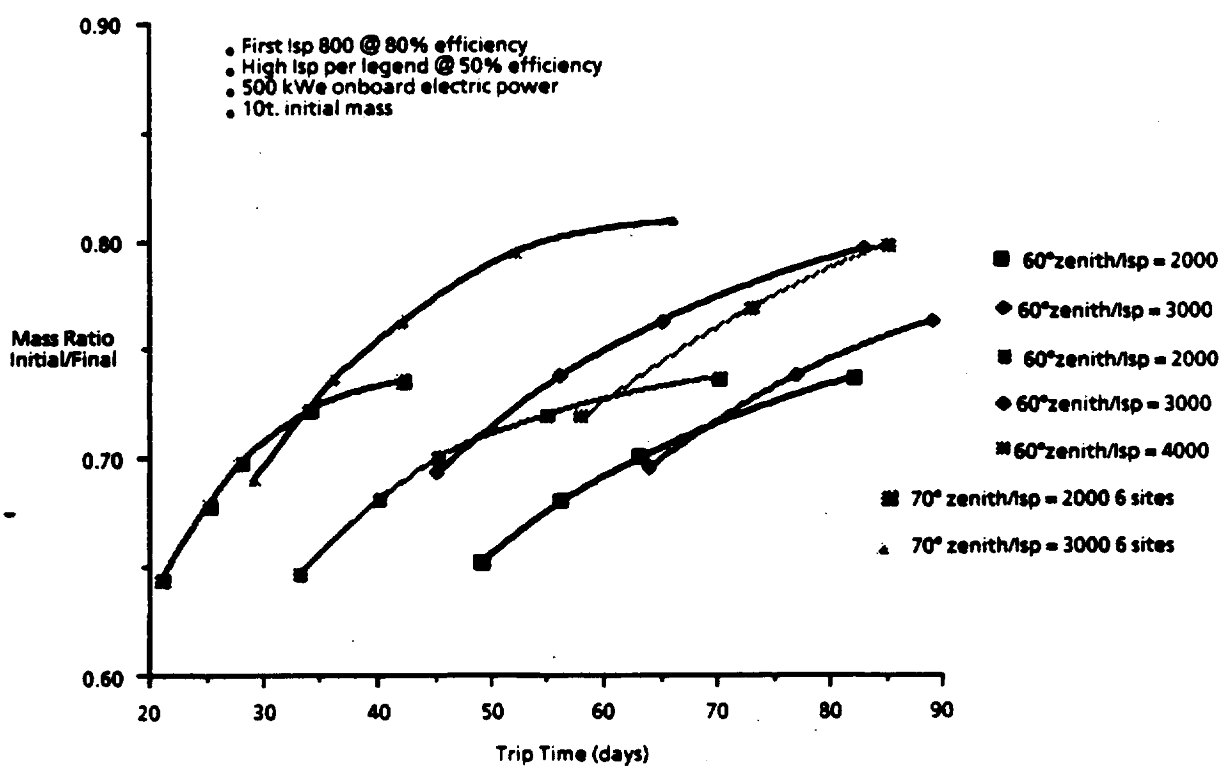
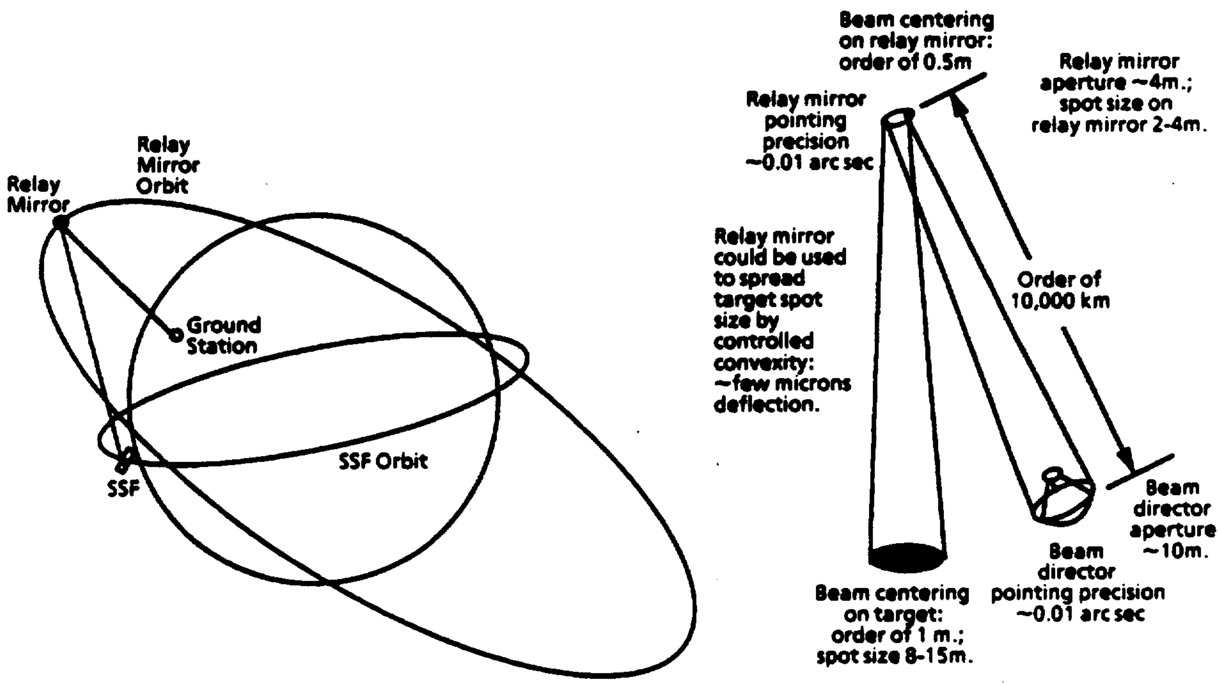


Figure 3.10. Effects of Isp, Zenith Angle, and Number of Ground Stations

AC5072



Beam director uses fine steering mirror and beacon from relay mirror to achieve pointing precision. The relay mirror must use signals from the beam director and the target.

Figure 3.11. Relay Mirror Diagram

AC5073

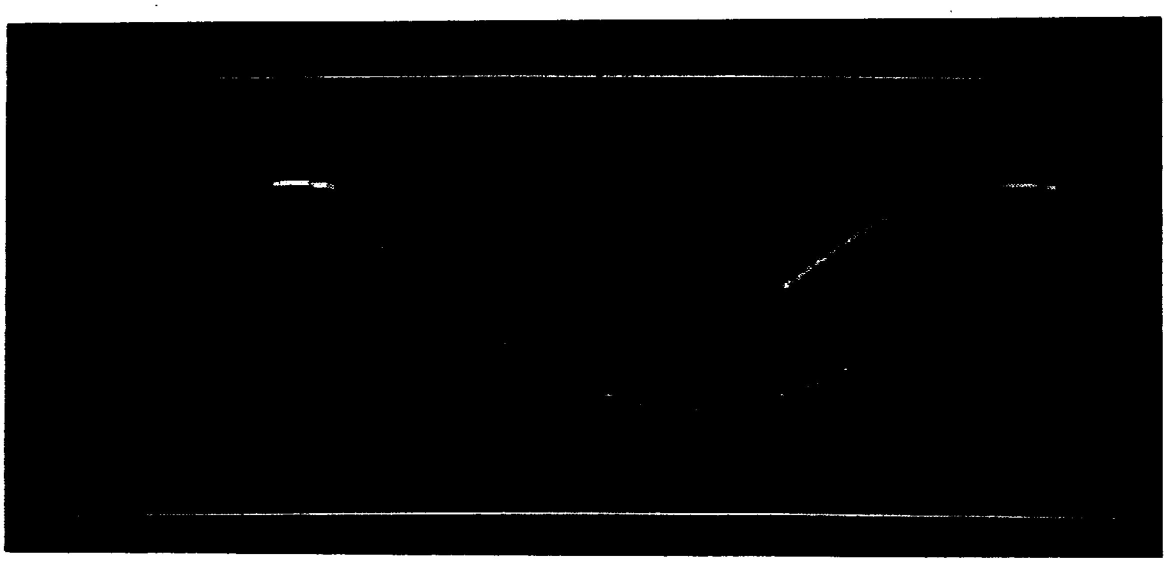


Figure 3.12. Computer Screen Photo of Relay Mirror Simulation - Map Display

This map display shows two relay mirror ground tracks and the EOTV ground track (long arm). The small circles show the location of ground stations and their range re direct transmission to the EOTV.

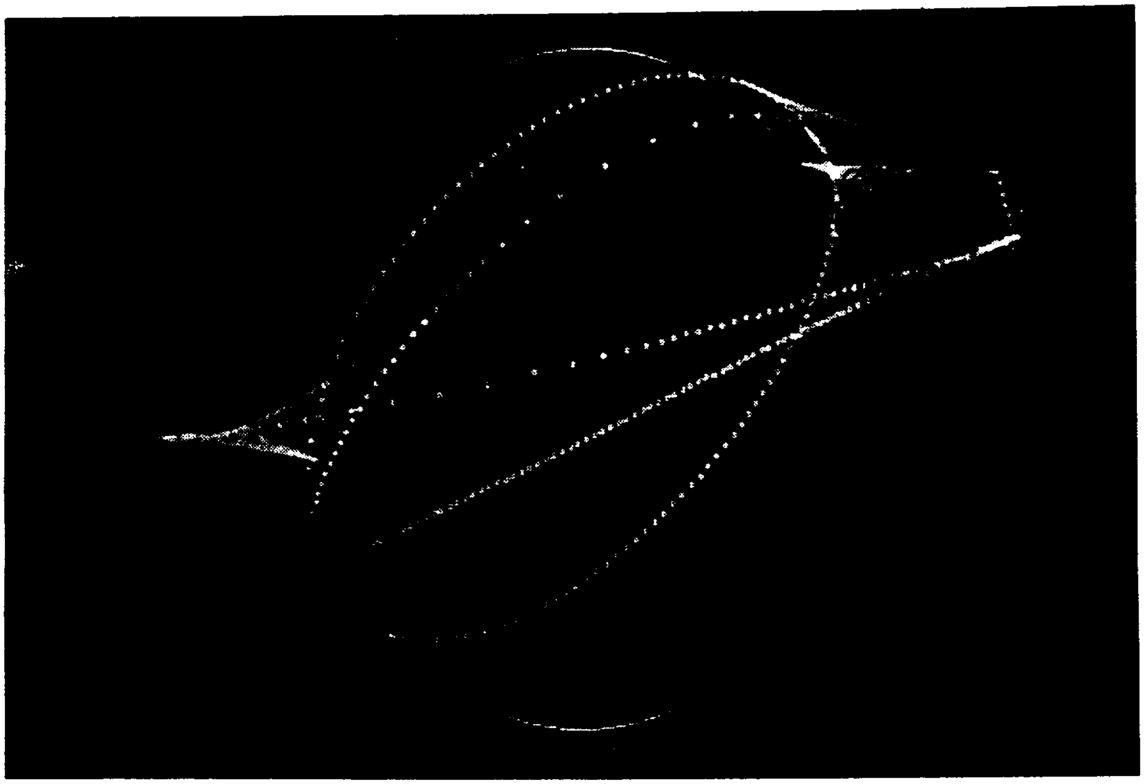


Figure 3.13. Computer Screen Photo of Relay Mirror Simulation - Pictorial Display

This pictorial display shows power transmission links from two ground stations via relay mirrors at higher altitude. The "twist" of the orbit paths occurs because this is a rotating coordinate system display.

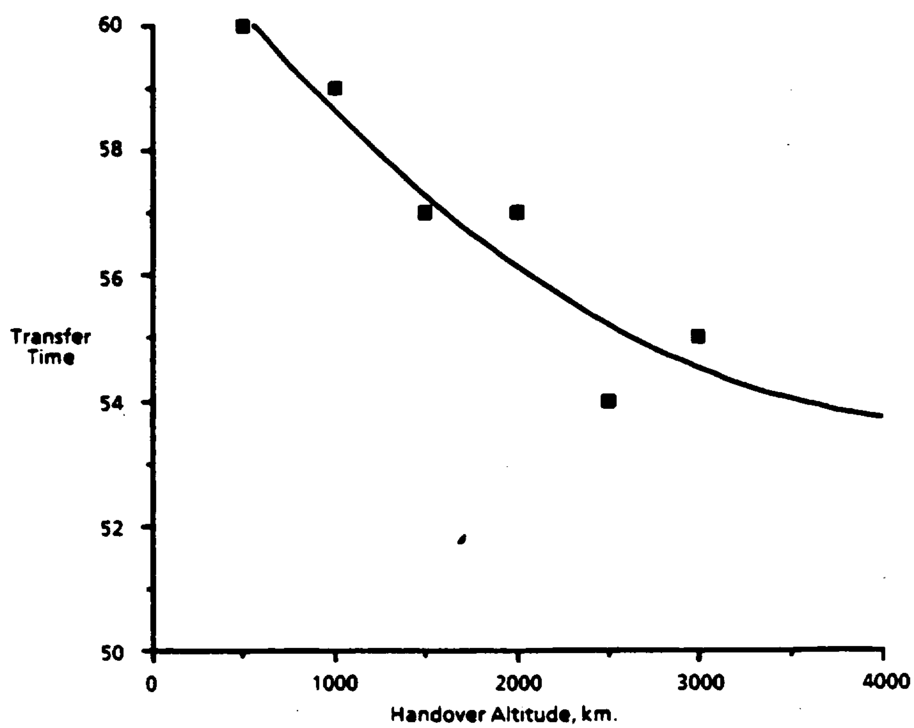


Figure 3.14. Trip Time Effects of Switchover Altitude

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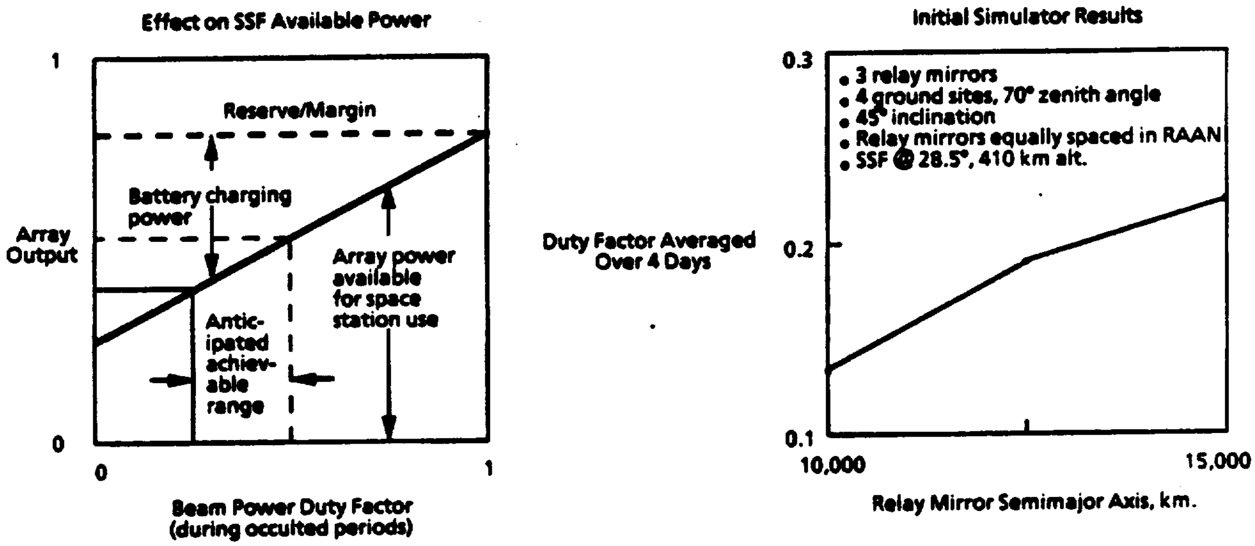

Issues/Questions Addressable by Simulation

- How does illumination duty factor vary with .. Number of relay mirrors? Altitude? Inclination? RAAN spacing? Number & location of ground sites?
- What is the short-period time history for illumination duty factor? How does it vary? How does source direction vary?
- Are there "special" synchronized orbits that make illumination available during part of every shadow period? What is involved in the control thereof?
- What is the spot size time-history and how does it vary? What is the effect of alternative spot size control schemes?

Figure 3.15. Duty Factor vs. Relay Mirror Altitude for Space Station Freedom Laser Power Augmentation

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4.0 CONCLUSIONS FROM THE STUDY

Laser-powered electric orbit transfer has major potential cost and economic advantages for LEO to GEO and LEO to lunar orbit missions. These advantages derive from the high specific impulse that can be used, eliminating most of the propellant that otherwise must be transported to low Earth orbit at high cost, and from the high power density achievable with the power collection and conversion system on the orbit transfer vehicle, yielding unprecedentedly high specific power estimates, and as a result performance, for the electric propulsion system.

The critical technologies needed to make such a system work have been demonstrated on a modest scale by the SDI technology program. The most important demonstration to be achieved is high power continuous operation of a free-electron laser. Boeing is presently under contract to the U.S. Army to achieve such a demonstration. The most significant cost driver technology is large-aperture adaptive optics. Important technical issues needing resolution are marrying the laser pulse format to the receiver photovoltaic system and selecting, based on experiments yet to be accomplished, the best electric thruster technology.

Estimated cost advantages for this technology tend to be insensitive to system parameters. The most sensitive parameters are the electric-to-jet power conversion efficiency of the thrusters, specific power of the thrusters, and trip time.

The general range of system parameters found to yield good cost performance is as follows:

Laser wavelength	0.85 micron
Beam expander aperture	10 meters
Laser power	1 to 5 megawatts
Number of ground sites	4 to 6
Electric specific impulse	2000 to 3000 seconds, with the early part of the transfer accomplished using resistojets at about 800 seconds
P/V array size	10 to 15 meters diameter
Array output	200 to 500 kWe
Zenith angle	60 - 70 degrees
Transfer time (up)	20 to 50 days
Number of vehicle reuses	3 to 10
Payload to GEO orbit	2500 to 5000 kg

Lunar cargo missions were briefly examined. Somewhat higher laser power and vehicle size were preferred. Payloads to lunar vicinity were on the order of 50 metric tons; trip times were longer than for GEO orbit missions. Lunar missions can be flown (a) with the electric vehicle spiraling into a low lunar orbit, with chemical propulsion for lunar descent, or (b) merely to the lunar vicinity with a gravity-assisted return to trans-Earth transfer orbit, and with chemical propulsion for lunar descent from the approach condition as for the high-thrust direct lunar mode. Very little lunar mission analysis was done; detailed profile analysis and trades remain to be accomplished.

The low duty factor attainable for ground-based laser sites illuminating vehicles in low Earth orbit is a very important issue for laser-powered electric propulsion. Rudimentary calculations show that energy storage on the vehicle is not a useful option in view of the great mass penalty. Significant performance improvements were obtained from "tricks" like the dual-Isp strategy described in this report. Use of relay mirrors (the laser beam goes from the ground to a relay mirror orbiting at circa 10,000 km altitude to the vehicle) can significantly improve low altitude duty factor. Enough "tricks" were found to make the low duty factor problem manageable in the sense of achieving good cost characteristics and acceptable trip times from the system; additional simulation studies hold out the promise of further improving the situation.

High priorities for future systems analysis include (1) more rigorous system optimization, (2) simulation and development of flight profiles for lunar missions and GN&C algorithms for LEO-GEO and lunar missions, (3) relay mirrors, using Taguchi methods to investigate effects of the large number of design parameters inherent in a relay mirror system, and (4) investigation of additional mission applications including power boost for Space Station Freedom using relay mirrors.

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Appendix A

Task 2

Survey of Beamed Power Applications

Energy transmission is required in most manmade systems, since the power source and power user are seldom the same. A wide variety of means are used in short range (under 100 meters) transmission, including mechanical (belts, shafts, etc.), hydraulic, compressed gases, and electrical. The options for long range energy transmission have been much more restricted, being mostly via wires for electricity, and pipelines/ships/trucks for fossil fuels. Power beaming represents a new technology for long range energy transmission. As such, it can be considered as a replacement for existing long range transmission systems, and for new applications that are not possible by existing methods.

Aeronautical and space-related applications of power beaming are of most interest to NASA. But, because energy transmission is such an integral part of an industrial economy, the benefits of power beaming technology spread much further. This would augment the rationale for developing the technology. We have collected potential applications for power beaming in this section. Where possible we identify or quantify the advantage of beamed power over other solutions. The list should be considered only preliminary, as our task did not permit in-depth investigation of non-space applications.

Applications for Beamed Power. The following list has been organized by type of application. The list is inclusive in the sense that applications which are yet to be examined for costs vs. benefits are listed.

Some set of applications will together form a mission scenario for beamed power. The mission scenario can then be used to evaluate how to provide the beamed power, and the costs and benefits of doing so.

A.1 Commercial Applications

a. Communications & Other Satellite Support

1. Potential Benefits:
   - (a) Extending the operational life of existing and future comsats by bridging GEO comsat eclipse period.
   - (b) Reduce or replace the weight of batteries, solar arrays, and stationkeeping propellants (by using electric propulsion). This increases the useful payload fraction.
   - (c) Reduce launch costs by using an EOTV, which uses 10 times less propellant than current vehicles due to higher thruster performance (Isp).
   - (d) Enable faster EOTV delivery missions by increasing power output from a given array area.

(e) Provide more power for a given array area, enabling higher-power applications (such as space-based radar)

2. Specific Applications

(a) Relatively low power (a few kW delivered on the target) near-IR or visible laser for eclipse bridging of satellites. When satellites go into the EarthUs shadow, most of them depend on limited cycle life batteries to continue operating during the eclipse.

(b) At 10 kW level, enable higher communications satellite operating power levels (enabling high power direct broadcast).

(c) At 50-100 kW level, power orbital radar satellites.

(d) At 1 MW level, support EOTV propulsion.

(e) Power augmentation for Space Station Freedom.

NOTE: For some of these applications the power in the beam will be much greater than the power on target because of geometry effects, i.e. the target solar array is smaller than the beam or is not circular.

b. Power to Remote Fixed Sites

1. Ground Point-to-Point Relay

(a) Line-of-sight to remote locations (i.e. mountaintops)

(b) Aerostat or Beam-self-powered airborne relay

(c) Orbiting Relay

c. Power Terrestrial Aircraft and Other Vehicles

1. Potential Benefits:

(a) Reduced emissions from electric or directly heated engines

(b) Range/endurance increase relative to on-board fuel limited systems

2. Candidate Implementations:

(a) Marine Applications

(1) PV receiver

(2) Heat engine receiver

(b) Aeronautical Applications

(1) Electric engines

(2) Heat Engines

(c) Surface Transportation

(1) Microwave emitters buried under road triggered when cars pass over them.

Based on construction cost of I-565 ($10M/mile), one could spend $100 per meter per lane and only add 10% to total road cost. This could buy and install a microwave emitter per meter for a six lane highway. Emitters only operate when cars are over them (low duty cycle) so they may be high peak power devices.

The state of California has legislated a requirement for 10% pollution-free vehicles by the late 1990's, but electric vehicles have had a limited range. Beamed power can add to electric car range by supplying power to electric vehicles while they are in motion on the highways. The cars would still use internal battery power when off the major highways, but these drive distances are typically short. Beamed power for cars could be safer than electrified rail. Power for cars using electrified rail or overhead catenaries involves exposed conductors. With the microwave option the power equipment can be encapsulated.

Conceptually, the receivers would be placed on the bottom of the vehicles, and shielded from the occupants.

A study comparing buried microwave transmitters to other electric options would be useful.

d. Supply Raw materials to Earth

1. Potential Benefits:

(a) Strategic metals are available in quantity (nickel, cobalt, platinum group) in Iron-Nickel type asteroids.

(b) Metals are in free-state this means lower recovery energy and reduced environmental impact.

(c) Higher grade steel from asteroids can reduce corrosion losses to domestic economy (Approx. $60 billion per year).

(d) Large market for metals to justify development costs.

2. Specific Applications. These applications will require significant transportation and processing power:

(a) Asteroidal precious metals extraction.

(b) Steel (i.e. the natural 9% Ni, 1% Co ferrous alloy that occurs in asteroids.

(1) Lunar Implanted Meteoritic Iron

- Look at benefits of using already reduced metal from energy standpoint.

(2) Iron-Nickel Asteroids

e. Supply Energy to Earth

1. Potential Benefits:

(a) If fossil fuels are displaced by beamed power, then less carbon dioxide is added to the atmosphere. This reduces the potential for greenhouse warming.

(b) Power supplied from outside the Earth can be a renewable, low environmental impact energy source.

(c) Power relay systems can move large blocks of power across oceans from producers in remote areas to users in populated areas.

(d) Previous studies have shown the potential for low cost power production.

(e) Energy is a large enough (multi-hundred billion $/year) market to justify significant development costs.

2. Specific Applications

(a) Orbiting Solar Power Satellites

(b) Power Relay Satellite

(c) Lunar Surface Solar Power Station

(d) Nuclear power in orbit.

A.2 Environmental Applications

a. Protect Biosphere

1. Asteroid/Comet Interception/orbit adjustment

(a) Power for propulsion. There are two possibilities for propulsion applications under this heading. The first is to provide power for direct laser-thermal propulsion. The propulsion would accelerate an outgoing vehicle carrying an intercept nuclear device. The second possibility is for sending large amounts of power (100's of MW) to a large collector at the target asteroid or comet, and powering a propulsion system that over time deflects the orbit of the object.

(b) Direct deflection by vaporizing material.

2. Send power to high altitude to make ozone or remove chlorine

(a) Drone aircraft power

(b) Direct reaction stimulation.

3. Orbital Debris Cleanup

(a) Reflected light deceleration, ablative deceleration, or total vaporization (no vehicles in these options, just ground beam source).

(b) Maneuverable electric vehicle to recover, de-orbit, consolidate large debris items to reduce the debris source population.

A.3 Science Applications

a. Astronomy with improved ground telescopes

   1. Potential Benefits:

      (a) Technology spin-off of (adaptive optics, optical interferometry) to pure astronomy telescopes.

   2. Specific Applications:

      (a) Use of beam power transmitter as a high performance bistatic "telescope" (works like synthetic aperture radar).

b. Planetary Science. These apply to Moon, near Earth asteroids:

   1 Mapping of bodies with improved ground telescopes

      (a) High resolution imaging

      (b) Laser altimetry

   2. Surface response experiments to beam illumination

      (a) Narrow band spectra from tunable FEL

      (b) Heat capacity measurements

      (c) Melting point determination by reflectance change

      (d) Vaporization by focussing relay mirror for spectrometry

      (e) Mass spectrometer analysis of resulting vapor (at close range) or atmosphere (Moon, long range) from one place while sampling a variety of other places with beam

c. Power for Scientific Satellites.

A.4 Exploration Initiative Applications

a. Transportation Support

   1. Power for Transfer Vehicles (Lunar & Mars)

   2. Power for Transportation Nodes

   3. Power for Planet to Orbit Systems

b. Earth-to-Orbit Propulsion

   1. Orbital Bridge Propulsion Unit

      (a) Beamed Power relay satellite

      (b) Electrodynamic engine.

c. Robotic Exploration Support

   1. Powering of the following:

      (a) Orbiters;

      (b) Landers;

      (c) Rovers;

(d) Sample Return;
(e) Robotic Mining;
(f) Robotic Construction.

d. Surface Installation Support

1. Central Receiver Utility. The concept begins by decoupling the receiver diameter from the arriving beam diameter. In an early lunar scenario, there may be an energetic beam (Megawatt class) powering a transfer vehicle. The initial lunar installation may not need this power level. In this case a simple fixed position photovolatic array is placed at the center of the beam and the rest of the power in the beam discarded.

   As surface power requirements increase, an incremental growth path can be pursued to higher power levels. The options are:

   (a) Deliver additional PV arrays to intercept a larger fraction of the beam,
   (b) Deliver reflectors to concentrate lower intensity part of beam onto PV arrays,
   (c) Manufacture reflectors from local materials.

   Simply melting surface material and allowing it to cool should make a reasonably flat surface. This surface can be coated with aluminum from a simple evaporating wire. These would not have to be brought from Earth.

   (d) Use part of the beam directly as a source, without conversion to electricity

   Either mirrors pick off parts of incoming beam and focuses/directs it to surrounding users, or equipment that requires process heat is directly in the beam area. Power levels ranging from support of outpost (intermittent occupation by about 5 or less crew), to support of base (full time occupation by up to 500 crew). By avoiding a conversion step the efficiency chain improves significantly.

2. Electric Power
   (a) Integrated Lander/Rover
   (b) Pressurized Rovers
   (c) Regolith-moving machinery

3. Process Heat

(a) ISRU Glass Production
(b) Waste sterilization and reduction to safe forms
(c) Extraction of Volatiles by heating
(d) Extraction of Oxygen by Pyrolysis
(e) Paving and tunneling via BP source heating
(f) Evolution of refractories by boiling everything else away
(g) Iron separation from agglutinates by melting
(h) Structures build-up by vapor deposition
(i) Pressurized module warmth during lunar light

4. Lighting
   (a) General Illumination during Lunar night vs.night vision goggles or Earthlight
   (b) Tuned light for plant growth.
5. NEP Vehicle sending power down to surface. An NEP vehicle will have a surplus of power available when in parking orbit. The concept is to then send power down to the surface via a power beam.
6. Relay mirrors for surface vehicles. Surface vehicles operating in the line-of-sight of a tower can receive illumination and power by means of a relay mirror. The mirror will pick off part of the beam being sent from the Earth to the Moon, and re-aim it towards the vehicle. The use for this is the same use as for the main base — to provide power during the lunar night.

   Using a relay mirror would simplify the requirements for a pilot beam - the beam would be aimed at a fixed spot on the lunar surface, and the pilot beacon could also be fixed. The surface vehicles, however would now be fixed to the line-of-sight range of the tower (18 km for a 100 m tall tower).

   Line-of-sight surface mirror relay network for powering vehicles (rovers, mining). Can ease problem of beam focus with 2km pilot beam separation by only having a fixed pilot beam/receiver location.
7. Chemical Production/Regeneration
   (a) Catalysis by specific wavelengths
   (b) Regeneration of reagents required for in-situ resource use
   (c) Plant nutrient extraction from local materials.

A.5 Technology Demonstration Applications

a. Terrestrial Technology Demonstrations.

1. Antarctic remote site power demonstration. Demonstrate short-to-intermediate range power beaming via an aerostat or power beam powered aircraft, with power relayed from a base site to a remote site (order of kW, order of 100 km).
2. South pole base power demonstration. The intent of the demonstration is to deliver substantial amounts of power to a South Pole Base. Currently all power for the base comes from diesel generators. Supplying the fuel to run the generators for the entire winter is a major logistical challenge. (power relay to South Pole Base as long range demonstration; offloads base resupply)

b. Orbital Technology Demonstrations.

c. Lunar Technology Demonstrations. ISRU mirror fabrication (melt regolith + aluminize).