Final Report

In-Space Transportation for GEO Space Solar Power Satellites

Contract NAS8-98244, Modification No. 3
Boeing Reusable Space Systems
December 22, 1999

This document represents the final report of Contract NAS8-98244, Modification Number 3. It provides the results of an effort to evaluate options for in-space transfer of satellites for generating power in space from solar energy. The "Sun Tower" satellite design was considered, with in-space transportation systems which carry relatively small segments (~27,000 kg) of the large "Sun Tower" to the geostationary operational orbit of the satellite.

Several individuals contributed to this activity. James A. Martin of Boeing Reusable Space Systems in Huntington Beach, CA, provided technical project direction and is the primary author of this report. Benjamin B. Donahue of Boeing Space Sytems in Huntsville, AL, provided concept analysis and design. James A. McClanahan of Boeing Rocketdyne in Canoga Park, CA, provided trajectory and propulsion parameters. Schuyler Lawrence of Reusable Space Sytems provided cost analyses. Mark Henley of Reusable Space Systems and Paul Gill of Rocketdyne provided programmatic support in managing the project and guided aspects of the technical performance.

Contents

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This report contains four sections after the tasks are described briefly. Sketches illustrate what options were considered. Results for each task are presented. Additional effort results are presented. Finally, the results are discussed, some recommendations are presented, and concluding remarks are listed.

This activity builds upon prior work on Contract NAS8-98244, which defined a Sun Tower Space Solar Power (SSP) Satellite System. In-space transportation for deployment of such huge SSP satellites presumes that relatively small segments (<50,000 kg) are individually transported from low Earth orbit (LEO) to geostationary Earth orbit (GEO), where they are assembled into the large satellite, probably robotically.

Contract Tasks

- 1.0 In-Space Transportation System Requirements
  - 1.1 In-Space Transportation Delta Velocity Requirements
  - 1.2 Other In-Space Transportation System Requirements
- 2.0 In-Space Transportation System Concept Design
  - 2.1 Autonomous Transportation
  - 2.2 Reusable Orbital Transfer Vehicle Transportation
  - 2.3 Tether
  - 2.4 Sub-Orbital Launch
- 3.0 In-Space Transportation System Comparison
- 4.0 Additional Applications of In-Space Transportation System

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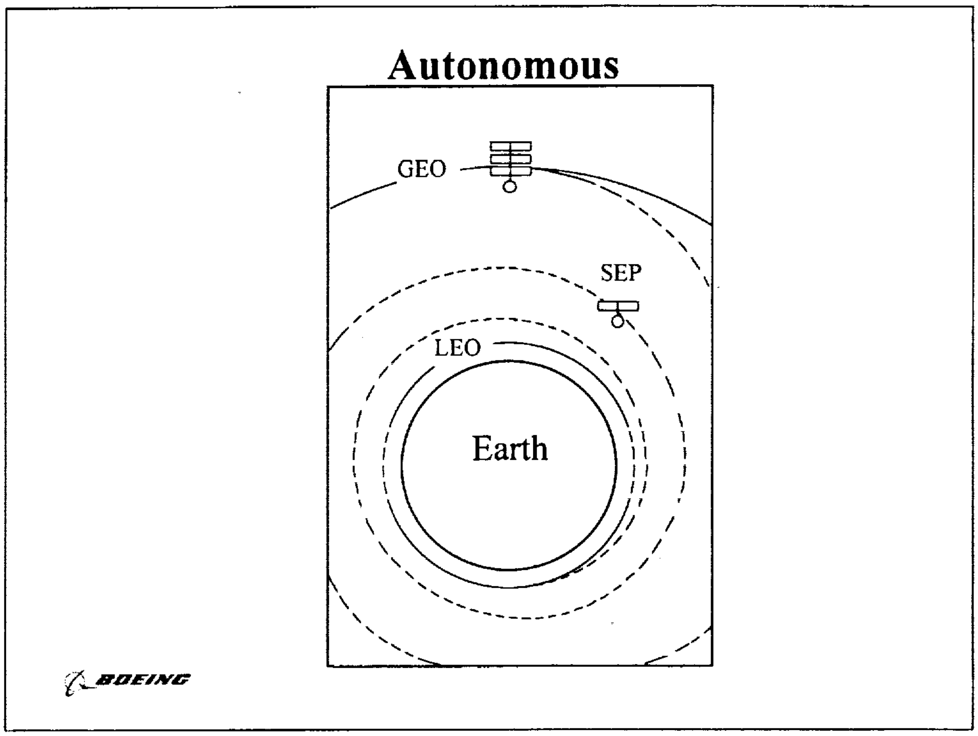
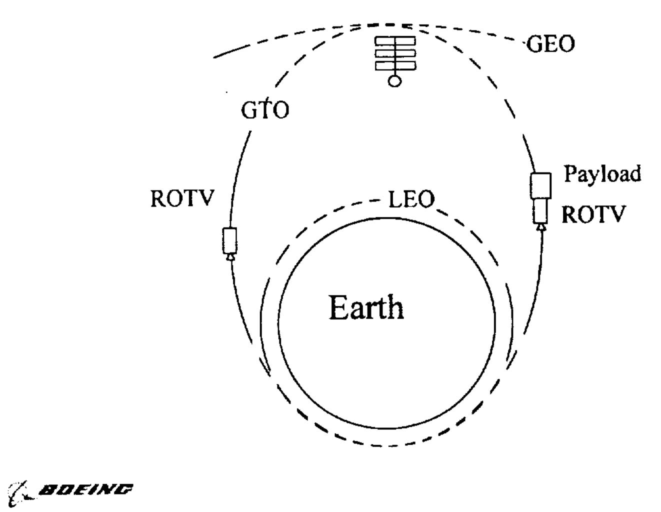
The four tasks with six subtasks are listed here. The results are organized by task.

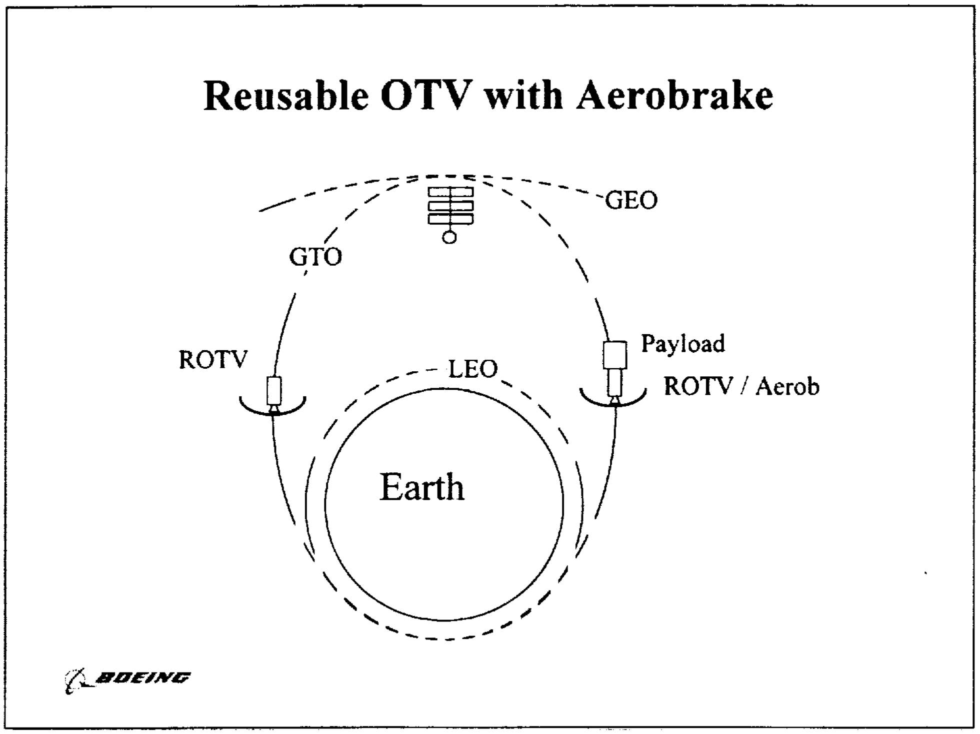


The point of departure for this study was an in-space transportation option which uses spacecraft segment solar electric propulsion for autonomous transfer. In this option, the launch vehicle places the package in a 300 km equatorial low Earth orbit (LEO). The photovoltaic solar arrays are partly deployed to provide power for transfer. Hall thrusters use krypton propellant to move the package to geostationary Earth orbit (GEO), where the package attaches to the Sun Tower satellite. The thrust arcs are continuous, except during passage through Earth's shadow.

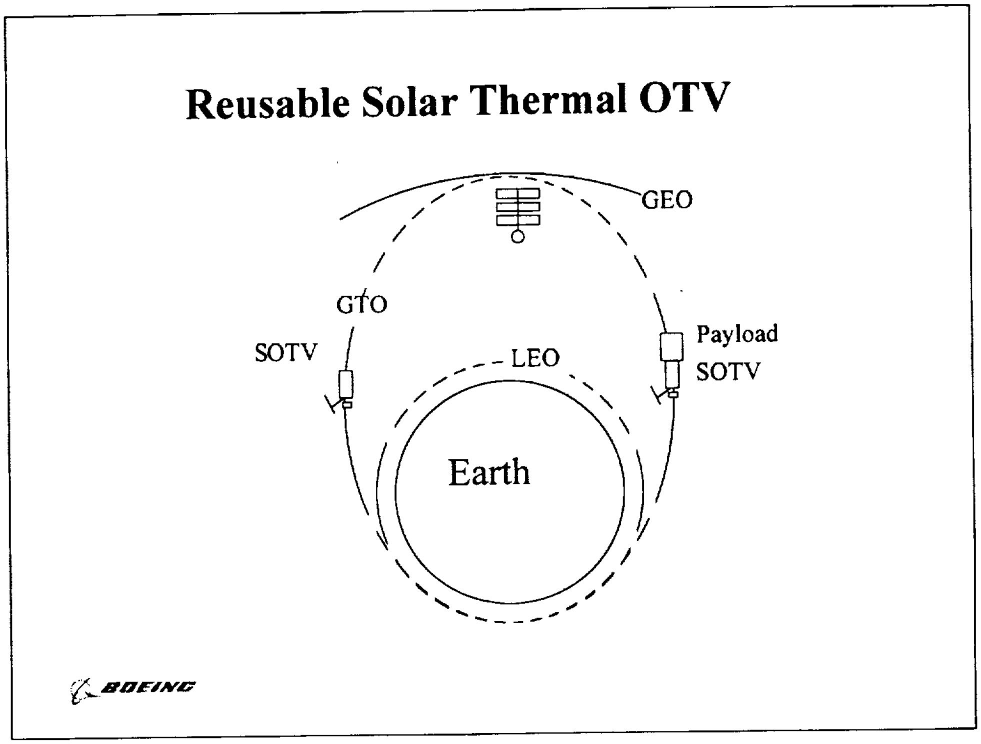
Reusable OTV



The reusable orbit transfer vehicle (ROTV) option delivers the same net payload to GEO as the baseline autonomous option. The ROTV has a lifetime of 200 flights. An oxygen and hydrogen chemical rocket engine is used. The payload, ROTV, and propellant are at a node in LEO prior to departure. After delivering the payload, the ROTV departs from GEO into a geosynchronous transfer orbit (GTO) and returns to LEO using chemical propulsion.

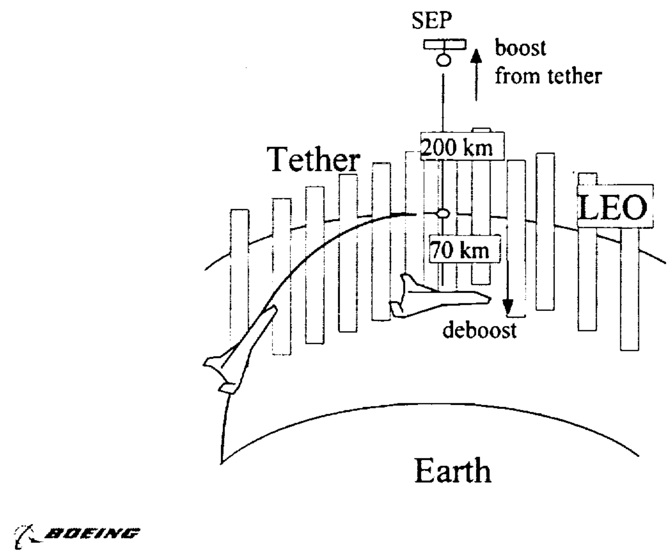


The ROTV option with an aerobrake (ROTV-AB) is similar to the ROTV option except that some of the propellant is not needed because the aerobrake uses the atmosphere of the Earth to brake out of the elliptical transfer orbit. Only a small circularization burn is needed to return to LEO.



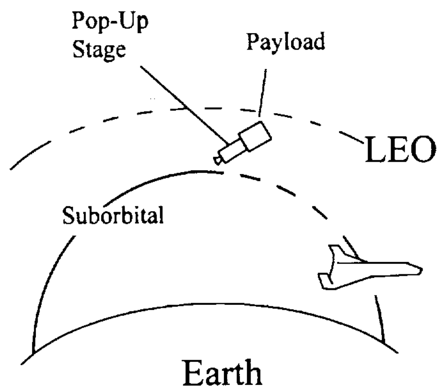
A reusable OTV using solar thermal propulsion (SOTV) was also considered. Because the thrust of the STOV is low, the trajectory actually has many thrust periods. The actual trajectory is more complicated than the one shown and employs multiple perigee and apogee thrust impulses. The thrust arcs are optimized to minimize finite thrust losses (delta-V) while providing a reasonable transfer time. The intermediate trajectory arcs are not depicted above.

Tether Assisted Autonomous



The use of a tether was also considered. After the launch vehicle delivers the package to the LEO node, the launch vehicle is lowered on a tether while the package is raised up a tether. The center of gravity is kept at the LEO node. When the launch vehicle is released, it reenters without the need for propulsion. When the package is released, it moves outward without propulsion, getting a velocity boost equivalent to more than 450 m/s. Autonomous propulsion is used to complete the transfer.

Suborbital Launch Autonomous



The sub-orbital launch option has a launch vehicle that does not reach orbit. The launch vehicle releases a pop-up stage and payload package, then glides to a landing site about halfway around the equator. Two launch sites would be required. The small expendable pop-up stage, which could be an integral chemical propulsion system on the SSP satellite segment, propels the payload to LEO. Although this option could be used with several transfer options, only the autonomous option was considered.

Task 1.1 In-Space Transportation Delta-V Requirements

Configuration	Initial Orbit km x km/deg	Final Orbit km x km/deg	Delta-V km/s
Auto., imp., sb-o.	300 x 300/0	35,786 x 35,786/0	4.757
Autonomous	300 x 300/28.5	35,786 x 35,786/0	5.899
Autonomous imp.	300 x 300/28.5	35,786 x 35,786/0	5.894
ROTV, ROTV-AB	300 x 300/0	35,786 x 35,786/0	3.903
ROTV, SOTV	35,786 x 35,786/0	300 x 300/0	3.903
ROTV-AB	35,786 x 35,786/0	90 x 35,786/0	1.493
ROTV-AB	90 x 300/0	300 x 300/0	.062
SOTV	300 x 300/0	35,786 x 35,786/0	4.215
Tether	500 x 1,917/0	35,786 x 35,786/0	4.306
Sub-orbital	Sub-orbital/0	300 x 300/0	.300

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Task 1.0 was to provide the requirements for the in-space transfer. Task 1.1 was to provide the velocity increments for the various maneuvers and options. This chart summarizes the results of task 1.1. The velocity requirements were developed using trajectory analyses. The trajectory program SECKSPOT was employed for the autonomous (electric propulsion) analysis. The program POST-3D was used for the ROTV (chemical propulsion) analysis. The programs MultiBurn and POST-3D were employed for the SOTV (solar thermal propulsion) analysis. The value given for the tether option was based on the analysis of the length of tether allowed with momentum recovery. The transfer from the tether to GEO requires the 4.306 km/s shown. The initial analysis of the sub-orbital option was analyzed based on an assumption for the velocity. An OTIS analysis of the glide of the launch vehicle was initiated but was not completed during this effort.

The 51.6 LEO orbits were considered for compatibility with space station, but velocity increments were not resoved. Autonomous low thrust transfer with such a large plane change is difficult for spiral trajectories as the plane change must occur during the short equatorial crossings. Optimal high-thrust trajectories are complicated, using three burns. Lunar gravity assist, recently demonstrated on a commercial satellite, may be desirable if transportation from high inclination is required. Detailed analysis of such complicated trajectories was considered outside the scope of this study.

Velocity Increments Split by Thrust Segments

Maneuver dV's in m/s	Autonomous SEP	Autonomous SEP with Tether Assist	SOTV	ROTV with A/B	ROTV w/out A/B
LEO - GTO	4757	4306	4215	2450	2450
GTO - GEO				1453	1453
GEO - GTO	n/a	n/a	3903	1493	1453
GTO - LEO				n/a	2450
Post capture transfer	n/a	n/a	n/a	62	n/a

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The velocity increments on this chart provide the split between the LEO-GTO thrust segment and the GTO-GEO thrust segment, which were not shown on the previous chart. All values shown here are in m/s.

Task 1.1 In-Space Transportation Delta-V Requirements, Additional Applications

Configuration	Initial Orbit km x km/body	Final Orbit km x km/body	Delta-V km/s
Solar electric	300 x 300/Earth	250 x 1 sol/Mars	13.0
Solar electric	300 x 300/Earth	100 x 100/Lunar	8.57
Chemical	300 x 300/Earth	Trans/Mars	3.628
Chemical	Trans/Mars	250 x 1 sol/Mars	1.615
Chemical	250 x 1 sol/Mars	Trans/Earth	1.615
Chemical	300 x 300/Earth	Trans/Lunar	3.200
Chemical	Trans/Lunar	100 x 100/Lunar	0.900
Chemical	100 x 100/Lunar	Trans/Earth	0.915

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Some of the possible velocity requirements for human exploration missions are shown here. Many options for missions could use the transfer systems that might be developed for SSP. Some of the most interesting have been selected for consideration in Task 4.

Task 1.2 Other In-Space Transportation System Requirements - Autonomous

Inclination	deg	0	28.5
Loaded propellant	kg	5,988	7,223
Initial power (lift thrust)	kW	650	730
Final power (lift thrust)	kW	312	409
Initial thrust	N	29.2	32.8
Final thrust	N	14.0	18.4
Initial propellant flow rate	kg/s	0.001487	0.001670
Flight Time	days	91.9	90.7

- Momentum wheels of payload are used for attitude control
- Includes effects of Earth shadow
- Includes effects of power degradation due to radiation exposure

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Additional requirements for transfer options are provided in this and the following charts. The results provided are based on the designs shown under Task 2.0.

The most notable requirement for autonomous in-space transportation using solar electric propulsion is the trip time, which is assumed to be approximately 3 months. This time is considered to be a nearly optimal compromise between short trip times, which reduce radiation damage to solar cells and investment costs, and long trip times, which reduce propulsion system mass and cost and allow higher specific impulse and reduced propellant mass.

Task 1.2 Other In-Space Transportation System Requirements - ROTV

Inclination	deg	0
Loaded propellant	kg	56,960
Initial thrust	N	323,912
Initial veh thrust to weight	N/N	0.4
Initial propellant flow rate	kg/s	70.3
Flight Time	days	1.5
Attitude control	kg	243

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The key ROTV requirement involves engine parameters. Based on prior studies, an initial thrust-to-weight ratio (thrust divided by mass times reference Earth gravity) of 0.4 was assumed. This selection determined engine mass and effected propellant mass.

Task 1.2 Other In-Space Transportation System Requirements - ROTV-AB

Inclination	deg	0
Loaded propellant	kg	37,676
Initial thrust	N	242,367
Initial veh thrust to weight	N/N	0.4
Initial propellant flow rate	kg/s	52.6
Flight Time	days	1.5
Attitude control	kg	176

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Aerobraking can reduce ROTV propellant mass. The resulting reduction in initial mass lowers initial thrust and therefor engine mass.

Task 1.2 Other In-Space Transportation System Requirements - SOTV

Inclination	deg	0
Loaded propellant	kg	25870
Initial thrust	N	83
Initial veh thrust to weight	N/N	.00016
Initial propellant flow rate	kg/s	0.011
Flight Time	days	85.7
Attitude control	kg	152

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Solar thermal orbit transfer optimizes with an initial thrust-to-weight ratio considerably lower than chemical rockets. The ascent flight time is presumed to be about 3 months, as with the autonomous transfer option.

Other SOTV designs will have different requirements. In particular, and SOTV with an aerobraking capability would need less propellant. An SOTV used for only an initial part of the transfer could also have a reduced propellant mass. Such options were considered outside the scope of the study.

Task 1.2 Other In-Space Transportation System Requirements - Tether/Autonomous

Inclination	deg	0
Loaded propellant	kg	5,287
Initial power (lift thrust)	kW	611
Final power (lift thrust)	kW	289
Initial thrust	N	27.4
Final thrust	N	13.0
Initial propellant flow rate	kg/s	0.001397
Flight Time	days	89.9

- Momentum wheels of payload are used for attitude control
- Includes effects of Earth shadow
- Includes effects of power degradation due to radiation exposure

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Tether-assist can provide an initial velocity increment which reduces the increment required from other systems. For this study, a vertical tether was considered rather than a more challenging rotating tether. The downward de-orbiting of the launch vehicle balanced the upward momentum imparted to the in-space transportation system. For this study, tether assist was analyzed only with the autonomous in-space transfer option. The conservative tether system considered provided only a modest velocity increment, about 450 m/s. More ambitious tether options may be worth considering in future studies.

Task 1.2 Other In-Space Transportation System Requirements - Sub-Orbital/Autonomous

Note: data for autonomous apply for transfer from Leo to GEO

Inclination	deg	0
Loaded propellant	kg	2,700
Initial thrust	N	180,000
Initial veh thrust to weight	N/N	0.6
Initial propellant flow rate	kg/s	5,400
Flight Time	days	0.0
Attitude control	kg	8

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The data shown on this chart refers to the small pop-up stage to transfer the payload from the sub-orbital release point to LEO. The transfer from LEO to GEO is the same as in the autonomous transfer option. The propellant required for circularization of the relatively light payload and propulsion system, 2,700 kg, is significantly less than the propellant required for the launch vehicle circularization and subsequent de-orbit.

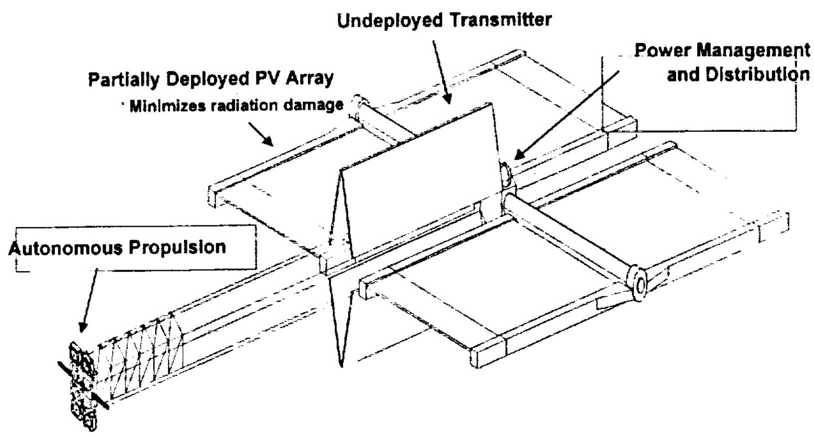
Task 1.2 Other In-Space Transportation System Requirements - Net Mass Transport

- All options designed to deliver the same payload as the autonomous case, 17,149 kg
- Flight rates for in-space transfer depend on required assembly rate for satellites and do not vary with option; current assumptions are launch of one in-space transfer per day
- Flight rates for launch are proportional to initial mass in LEO for each option, shown on comparisons

BOEING

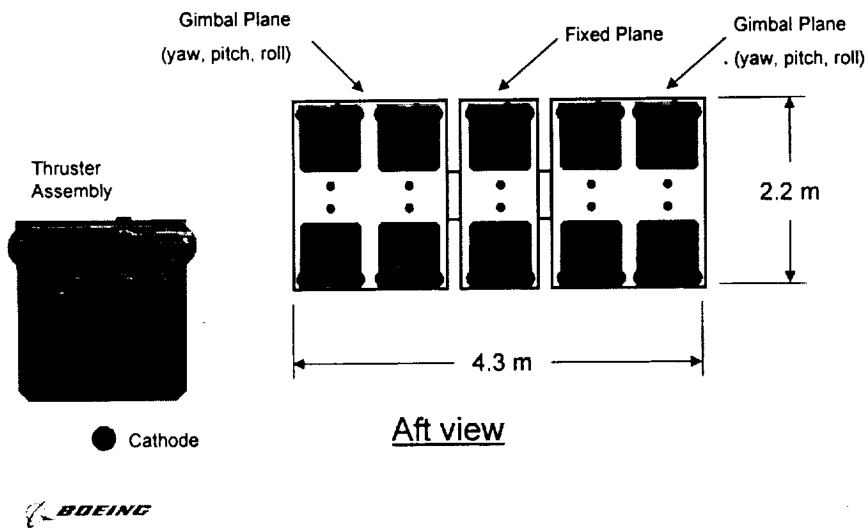
The net mass transport requirements apply to all options. For consistency in evaluating the options, an effective delivered payload mass was assumed to be 17,149 kg. This value excludes the mass of the in-space transportation system.

Task 2.1 Autonomous Transfer: Transfer Configuration



This chart shows the design of the element configuration during the autonomous transfer. Photovoltaic (PV) arrays are only partially deployed to minimize radiation damage.

Task 2.1 Autonomous Transfer: Representative Propulsion Module



This chart shows the design of the Hall thruster system used in the autonomous transfer option. Ten thruster assemblies of 50 kW each are needed to provide transfer in 90 days.

Task 2.1 Autonomous Transfer (Baseline)

- Equatorial launch
- Partial array deployment for transfer
- No PPU, direct drive from high-voltage array

• Specific impulse	2 000 s
• Initial mass in LEO	27 000 kg
• Useable propellant	5 814 kg
• Mass in GEO	21 186 kg
• Residual propellant	174 kg
• Propulsion inert	2 810 kg
• Solar array degraded	1 053 kg
• Useful payload	17 149 kg

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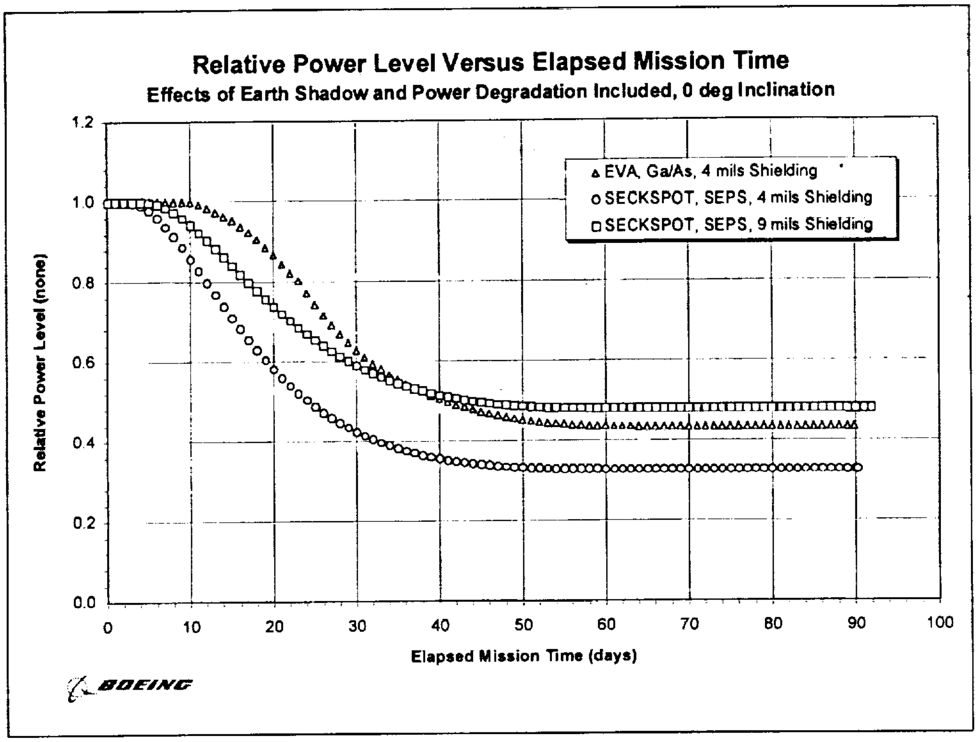
The autonomous transfer was selected as the baseline option because it was expected to provide economical transport. An initial mass in low Earth orbit of 27,000 kg was selected as compatible with likely launch vehicles. The low Earth orbit selected was an altitude of 300 km, circular, and equatorial. Hall thrusters with direct drive were selected to avoid the need for power processing units. The photovoltaic solar arrays were partly deployed to minimize radiation damage. The portion of the solar array that was deployed during transfer suffered some degradation, and the useful payload was calculated to represent the array size to provide equivalent power with no degradation. The additional array mass is shown as the degraded array mass, 1053 kg. The useful payload resulted from the initial mass in LEO less the propellant, propulsion inert mass, and degraded array. Other options were designed to transfer the same useful payload. Each payload represents a segment of the Sun Tower satellite with part of the transmitting antenna.

The specific impulse value of 2,000 s was based on recent work on Hall thrusters. Higher values may be possible but would require more thrusters and more power or longer trip times.



The analysis of the autonomous transfer included the effects of degradation of the solar arrays as they passed through the van Allen radiation belts. Details of that degradation are presented here, showing that the power drops more than 50%. Most of the degradation occurs while the system is passing through the high-dose region at an altitude of about half an Earth radius.

The shielding that can be provided must be limited to keep the mass of the array reasonable. If a concentrator is used, the much smaller arrays can be shielded much better.

Task 2.1 Autonomous Transfer: Effect of Launch Inclination

LEO Inclination degrees	Launch Latitude degrees	Useful Payload kg
0	0	17,149
28.5	28.5	15,571
51.6	28.5	not available

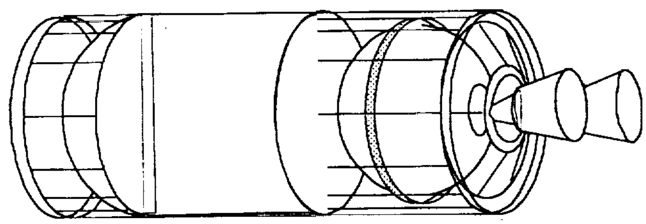
- Initial mass in LEO of 27 000 kg
- Equatorial launch also reduces launch vehicle size

Conclusion: equatorial launch is best

BOEING

One of the considerations was what launch inclination and launch latitude should be selected. Using Kennedy Space Center as the launch site and a due-East launch, the useful payload was reduced with the same initial mass in LEO. Analysis of the launch vehicle showed a 2% increase in dry mass to launch at the higher latitude and some increase in the launch costs. An attempt to analyze the transfer from a space station orbit failed to converge (a solution for a 90 day transfer from a LEO, 51.6 degree inclination to GEO could not be found by the SECKSPOT and may not be feasible), but the results for 28.5 degrees inclination indicate that the higher inclination would not be of interest. One of the advantages of a higher inclination is that the van Allen belt radiation damage is reduced. The results, however, indicate that the difference does not overcome the orbital mechanics advantage of equatorial transfer.

Task 2.2 Reusable OTV, No Aerobrake



This sketch shows the overall design concept of the reusable OTV. It uses liquid oxygen and hydrogen as propellants, with two main engines to provide engine-out abort capability.

Task 2.2 Reusable OTV, No Aerobrake

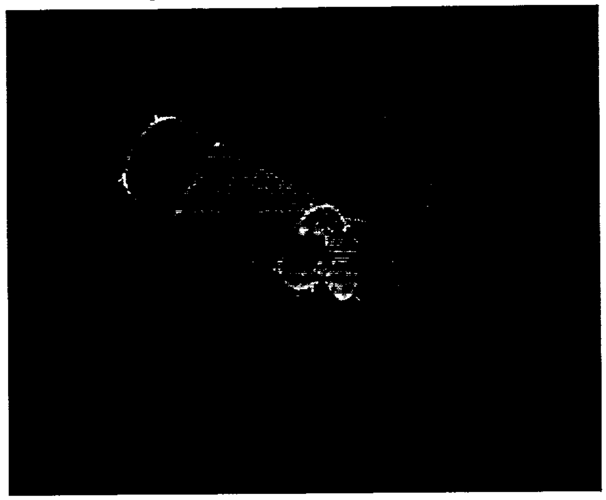
• Equatorial launch	
• Initial T/W	0.4
• Initial mass in LEO	86 053 kg
• Useful propellant	60 009 kg
• Reserves, resid., RCS	1 143 kg
• Stage inert	7 752 kg
• Payload	17 149 kg
• Specific impulse	470 s

BOEING

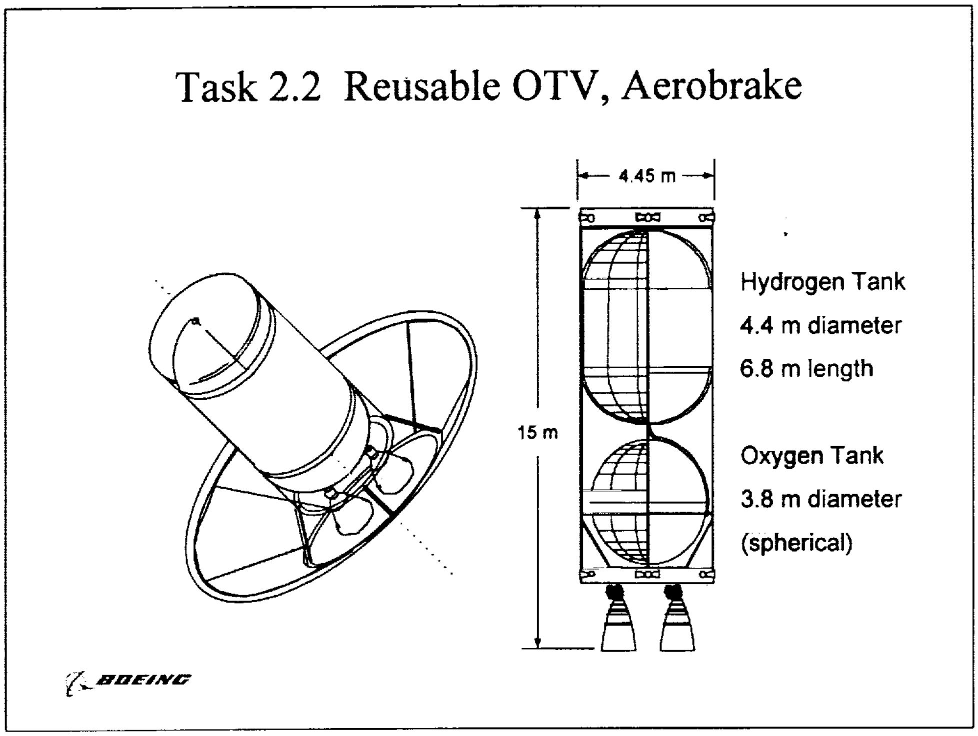
Results of the analyses of the ROTV option are presented here. The requirement for return of the ROTV to LEO as well as the lower specific impulse than the autonomous option leads to a much higher initial mass in LEO.

Task 2.2 Reusable OTV, Aerobrake

Concept Showing Aerobrake
Engines Fire Through Doors



The ROTV-AB is shown here with the aerobrake ready for slowing the vehicle into LEO. When the engines fire, doors on the aerobrake are opened.



These sketches show the design concept of the ROTV-AB option.

Task 2.2 Reusable OTV, Aerobrake

• Equatorial launch	
• Initial T/W	0.4
• Initial mass in LEO	62 274 kg
• Useful propellant	38 113 kg
• Reserves, resid., RCS	748 kg
• Stage inert (less aeroshell)	5 312 kg
• Aeroshell	952 kg
• Payload	17 149 kg
• Specific impulse	470 s

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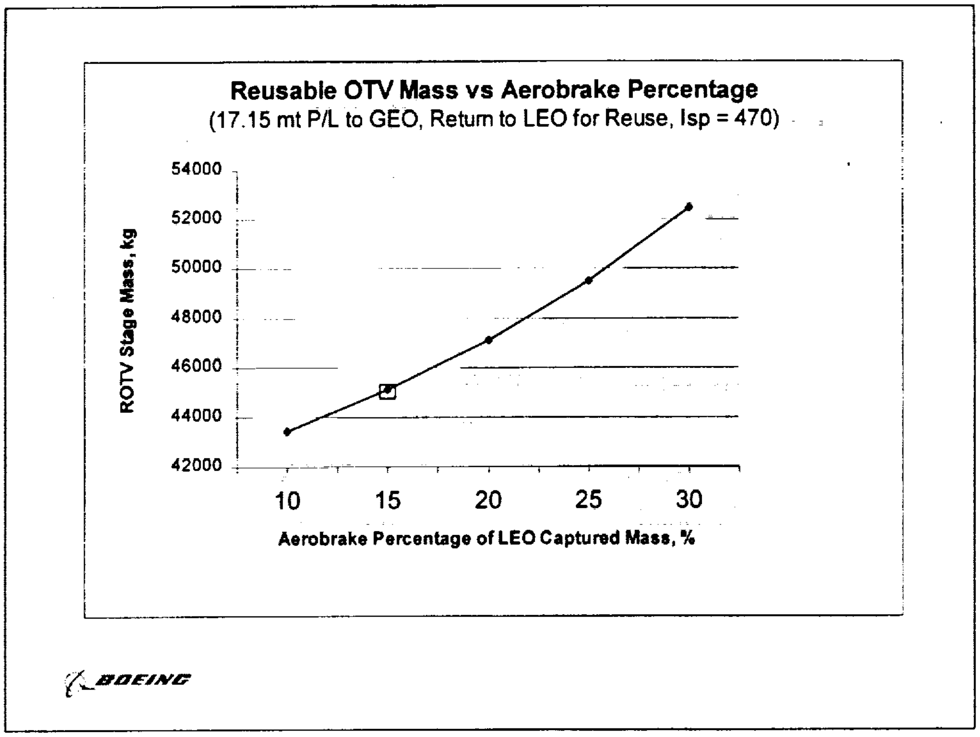
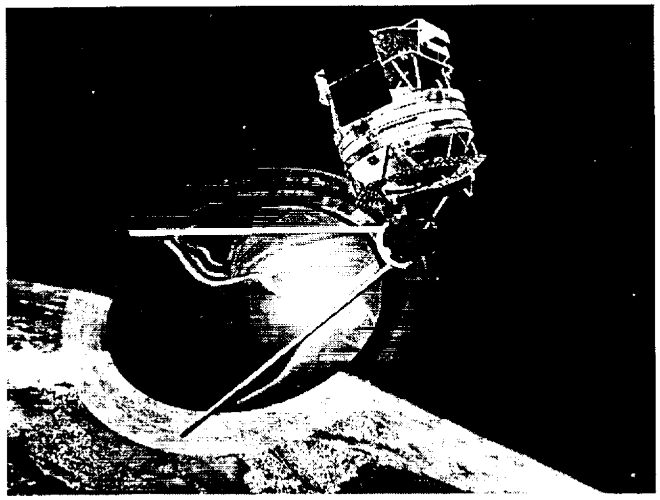
The analysis results for the ROTV-AB show that the initial mass in LEO is less than for the ROTV without the aerobrake but still higher than the autonomous option.



The ROTV-AB option analysis requires an estimate of the mass of the aerobrake. The analysis assumed a mass of 15% of the mass of the vehicle when the aerobrake is employed. While changes in the aerobrake mass assumption would change the ROTV mass estimate, the sensitivity of the aerobrake to the ROTV mass would not likely to effect the relative ranking of the options.

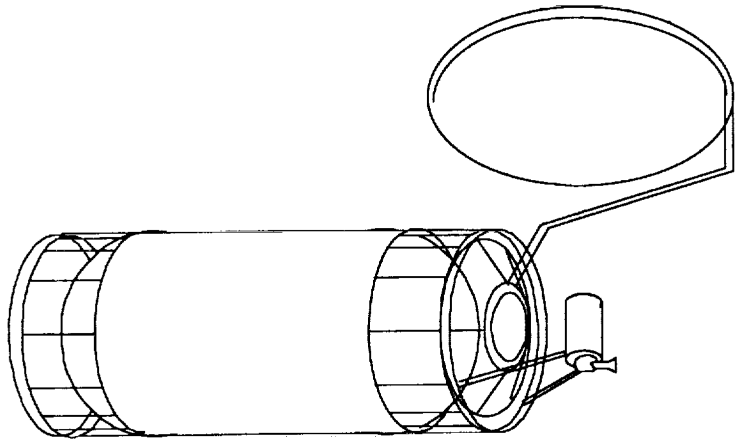
Task 2.2 Reusable OTV, Solar Thermal



BOEING

The reusable solar thermal orbital transfer (SOTV) is illustrated here as it might appear in an early technology demonstration mission.

Task 2.2 Reusable OTV, Solar Thermal



BOEING

The design concept of the SOTV option is shown here.

Task 2.2 Reusable OTV, Solar Thermal

• Equatorial launch	
• Initial T/W	0.00016
• Initial mass in LEO	52 877 kg
• Useful propellant	25 634 kg
• Resv., resid., b/o, RCS	797 kg
• Stage inert	9 297 kg
• Payload	17 149 kg
• Specific impulse	881 s

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The results of the SOTV analyses show that the mass is significantly lower than the chemical ROTV options but still higher than the autonomous option

Task 2.3 Tether

- Launch vehicle docks at LEO node
- Launch vehicle is lowered on tether as payload is raised on tether
- Center of gravity stays at LEO orbit of 300 x 300 km
- Simultaneous release of launch vehicle and payload
- No disturbance of LEO node orbit
- Tether length down limited to avoid heating and drag
- Tether length up limited to 200 km

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The tether-assisted in-space transportation option considered is described here. This tether option is considered to be relatively conservative. Other tether options would provide a greater benefits but would require significant advances in tether technologies such as electromagnetically propelled tethers, rotating tethers, and docking of launch vehicles at less than orbital velocity.

Task 2.3 Tether

- Equatorial launch
- Partial array deployment for transfer
- No PPU

• Specific impulse	2 000 s
• Initial mass in LEO	26 044 kg
• Useable propellant	5 133 kg
• Mass in GEO	20 911 kg
• Residual propellant	154 kg
• Propulsion inert	2 605 kg
• Solar array degraded	1 003 kg
• Useful payload	17 149 kg

BOEING

The results of the tether analyses are shown here. The initial mass in LEO of the package that is autonomously propelled to GEO is reduced about 3.5% from the value without a tether. That reduction reduces the launch costs, and the costs of the transfer are also reduced.

Tether-assisted in-space transportation also improves launch system performance in other ways. To balance net momentum, the launch vehicle is deployed downward and given the necessary velocity increment for reentry when released from the tether. This approach reduces the launch vehicle maneuvering system mass, with a corresponding increase in payload mass fraction.

While the reduction in mass and cost provided by the tether is not large, it is an indication of the potential of tethers. Further reductions from more aggressive tether options should be considered but were beyond the scope of the current effort.

Task 2.4 Sub-Orbital Launch

- Autonomous transfer LEO to GEO
- Launch vehicle releases payload ~300 m/s short of LEO
- Smaller launch vehicle, semi-global glide
- Expendable "pop-up" stage propels payload to LEO
- Oxygen/kerosene propulsion
- Payload to LEO 27 000 kg
- Stage gross mass 3 000 kg
- Stage propellant 2 700 kg
- Stage inert 300 kg

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The sub-orbital launch option was considered only with the autonomous transfer option. The analysis to determine the optimum sub-orbital drop-off node (velocity and altitude of the launch vehicle) requires an extensive trajectory analysis of the glide to the downrange landing site. A glide of about half the distance around the equator allows two launch sites to serve the launch function with no ferry. The scope of the current effort did not allow completion of this trajectory analysis. A reduction of 300 m/s in the launch vehicle velocity was estimated. The pop-up stage masses were estimated based on a specific impulse of 330 s, which represents economical propellants such as oxygen and kerosene, solids, or hybrids. Using oxygen and hydrogen would decrease the mass of the pop-up stage but would increase the costs considerably.

The analysis of the sub-orbital option was based on the assumption of a two-stage fully-reusable rocket launch vehicle. Such a vehicle is quite likely to be a good selection for space solar power launches. The sub-orbital option would increase the potential of single-stage launch vehicles, which could potentially reduce costs.

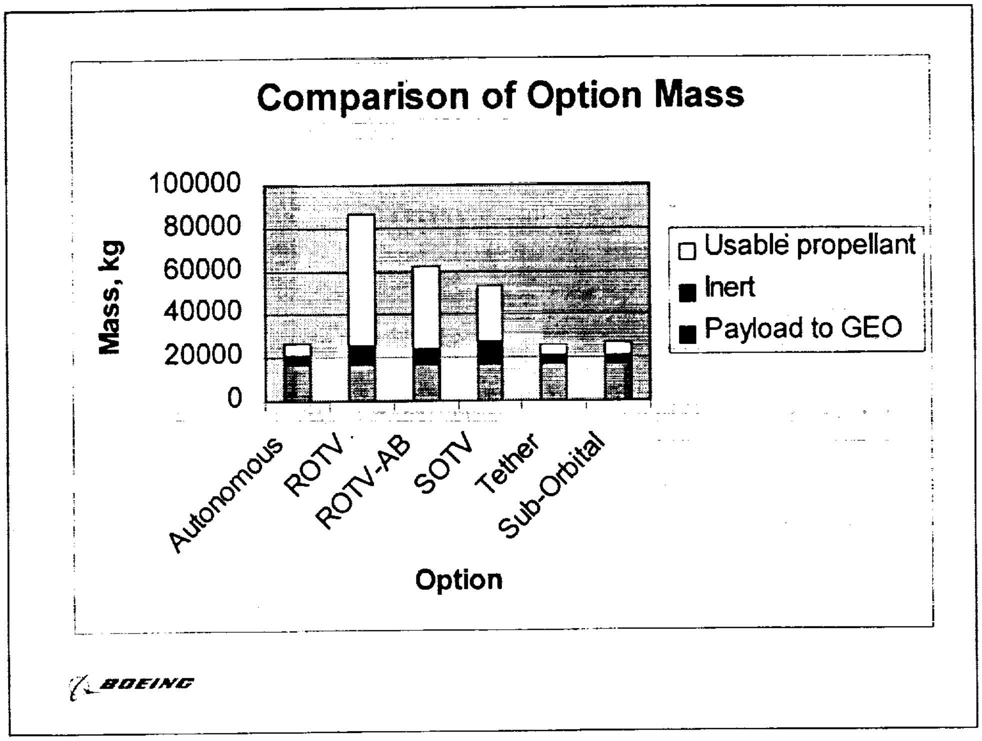
The semi-global glide required for the sub-orbital launch option could also be used for a tether option. The launch vehicle could meet the bottom of a tether at less than orbital velocity.

Task 3 Comparisons

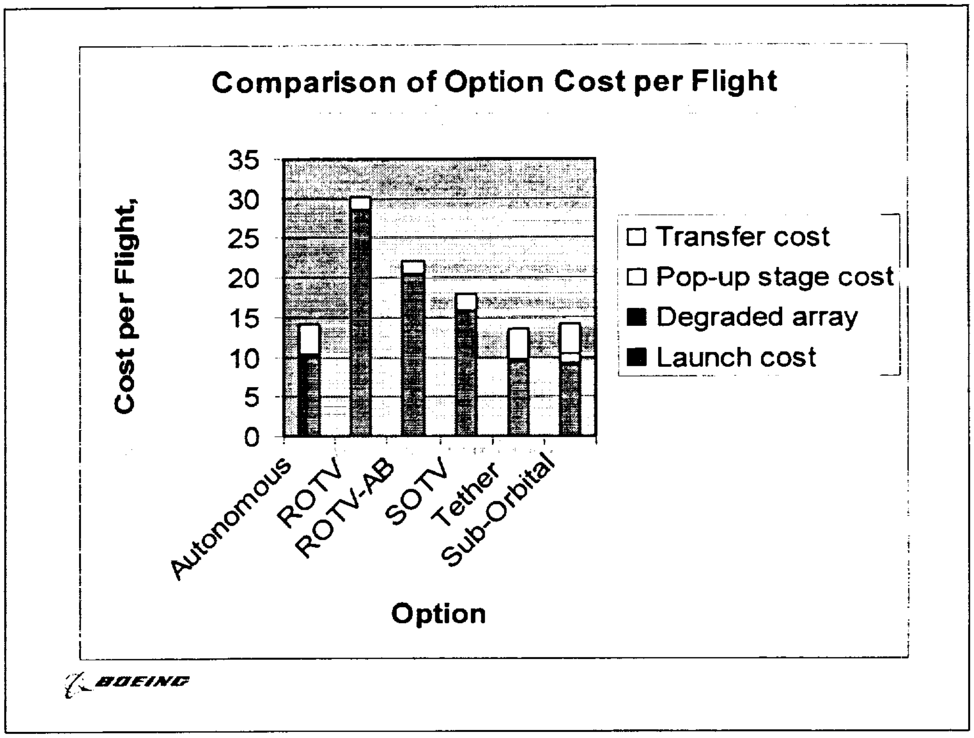
Option	Payload kg	Initial Mass kg	Inert kg	Trip Time days
Autonomous	17149	27000	4037	90
Auto., improved	17149	24632	4686	90
ROTV	17149	86053	8895	2
ROTV-AB	17149	62274	7011	2
SOTV	17149	52877	9297	90
Tether	17149	26044	3762	90
Sub-Orbital	17149	27000	4037	90

BOEING

This chart shows some of the results of the various options. The reduced trip times of the ROTV and ROTV-AB probably do not compensate for the large initial mass. The SOTV option has a large inert mass, but the stage is reusable. The benefits of tether assist indicate that this option should be given more consideration in the future.



The mass results of the various options are compared graphically here. The large propellant requirements of options without electric propulsion are obvious. The inert masses of the reusable transfer vehicles (ROTV, ROTV-AB, and SOTV) are only required to be launched once for every 200 flights to GEO.



The recurring cost per flight is estimated here for each option. These costs do not include any amortization of the development costs or the hardware production costs of the reusable vehicles. The lower specific impulse of the reusable vehicles leads to higher launch costs, which overwhelm all other costs. The tether option has the lowest costs by a small margin, but no costs were included for deployment or maintenance of the tether. The sub-orbital launch option reduces launch costs, but the cost of the expendable "pop-up" prpulsion system balances that reduction so that there is little or no net saving. The tether option provides the lowest cost, but the reduction is small for the conservative tether considered.

The results for the autonomous option indicate that the magnitude of the cost per flight is about $14 million for a net payload of 17,149 kg, or about $820/kg. This cost is split roughly equally between the cost to launch the payload ($370/kg) and costs related to the transfer ($450/kg). These costs do not include costs related to development and production of the initial fleet of the launch vehicle, including return on the investment in these costs.

Task 4 Additional Applications

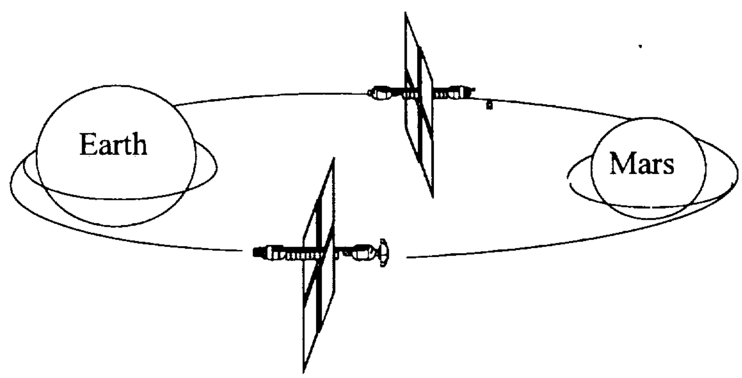
- Mars transfer vehicle
- Lunar transfer vehicle
- Propulsion system for a Mars/Earth Cycling Station
- Propulsion system for a Moon/Earth Cycling Station
- Orbit transfer from LEO to GEO commercial satellites

BOEING

The autonomous transfer design was selected for the consideration of additional applications. The same transfer design used for solar power satellite transfers can be applied to the additional applications considered.

Tethers were not considered for the additional applications. They could provide additional benefits to all applications, but more work would be needed to select a final tether concept. The benefit of the conservative tether concept analyzed may not justify the complexity of adding the tether to the transportation system. Other tether momentum exchange options with higher velocity increment capabilities would be preferred for such additional applications such as a lunar transfer system.

Human Mars Transfer Trajectory Using Solar Electric Propulsion



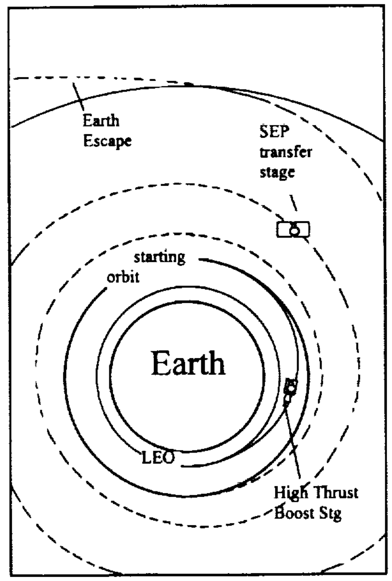
BOEING

This conceptual trajectory sketch shows how a solar-electric propulsion stage could transfer to Mars and return. The crew would transfer to the surface and return while the main vehicle stays in Mars orbit. The crew would return to Earth in a small capsule while the main vehicle continues in orbit for reuse.

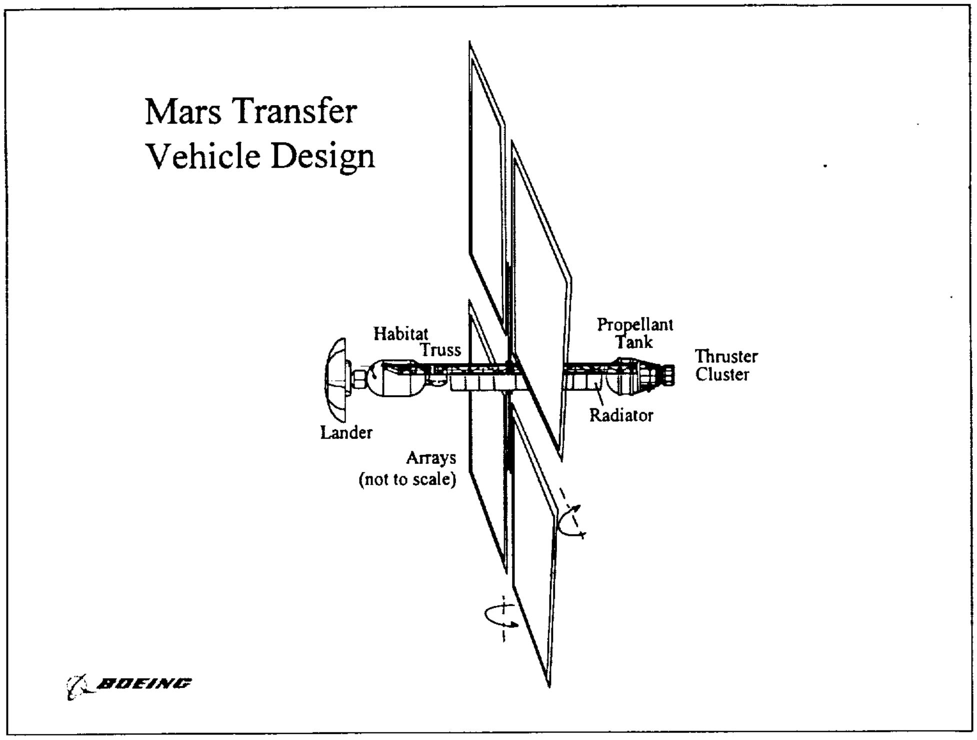
The mission design shown here and in the following pages is an approach to a Mars mission that has been developed as an example of a mission using solar-electric propulsion and has not been optimized or compared to other mission designs. The intent was to find a mission design that would allow reuse of a significant part of the high-cost equipment.

A conjunction class mission is assumed. The crew would stay at Mars about 500 days. The Mars orbit is 250 km x 1 sol (33,000 km). Propulsive capture into the Mars orbit is used. Only a small lander would enter the Mars atmosphere. Return to the 13,000 km Earth orbit is propulsive.

Human Mars Mission Leaving Earth



The human Mars mission would start with the main vehicle transferred from LEO to a parking orbit at high altitude. The solar arrays are not fully deployed in the high radiation belts, minimizing degradation. The solar-electric propulsion would then be used to spiral out from Earth orbit. The solar-electric system could be returned to the starting orbit for reuse. On subsequent flights, the crew, propellant, consumables, Mars lander, and Earth-return capsule would be lifted to the starting orbit by a high-thrust system.



This sketch shows one way that a solar-electric propulsion system, designed for the autonomous transfer of a space solar power satellite element, could be used for a human Mars mission. The lander and Habitat are located together so that both volumes can be used during the outbound transit. The habitat and lander are separated from the thrusters by a lightweight truss, and the radiator is mounted to the truss.

Propulsion system assumptions were based on the SSP transfer design. The specific impulse is 2,000 s. The mass of the collector power system is 8.4 kg/kW, and the mass of the thruster is 3.4 kg/kW. At Earth departure, the T/W, based on local gravity, is 0.0004, which gives a thrust of 236 N and a jet power of 2.3 MW. At an efficiency of 44 percent, the solar arrays must produce 5.25 MW. The low T/W minimizes structural loads.

2018 Piloted Mars SEP Vehicle

120 day Spiral out from 13000km Circ Orbit, 230 day Outbound,
500 day Mars Stay Time, 181 day Inbound Return Trip Time
Vehicle Recaptures into 13000km Circ Earth Orbit for Reuse

	kg	lb
PAYLOADS		
Crew Transfer Habitat	39,000	85,979
Crew Return Capsule	4,000	8,818
Mars Lander System	60,000	132,276
Total Payloads Wt	103,000	227,074
INERTS		
Power System (alpha = 8.4)	44,134	97,297
Thruster Sys (alpha = 3.4)	7,860	17,328
Tankage	42,373	93,415
Structure	6,030	13,294
RCS Propellant	2,417	5,328
Stage Dry Weight	102,813	226,662
PROPELLANTS		
Earth Spiral Out	124,527	274,533
Heliocentric transfer to Mars	240,178	529,497
Heliocentric transfer to Earth	87,583	193,085
Earth Capture	18,520	40,829
Total Propellant Weight	470,808	1,037,944
TOTAL WT	676,622	1,491,680

BOEING

The mass statement for a Mars mission in 2018 is given here. The crew transfer habitat and the inerts, except for RCS propellant, are reused.

Mars Missions Designed for Various Launch Years

3000 km piral Out Delta-V km/s	Spiral Flight Time days	Earth-Mars Helio. Departure Date	Outbound Helio. Delta-V km/s	Outb Flight Time days	Mars Arrivel Date none	Mars Departure Date none	Inb. Helio. dV km/s	Inb. Flight time days	Earth Arrivel Date none
3.936	120.0	12 Jan 2014	14.850	209.5	10 Aug 2014	15 Jan 2016	12.375	209.9	12 Aug 2016
3.936	120.0	05 Mar 2916	13.552	189.0	10 Sep 2016	11 Apr 2019	11.205	177.2	05 Oct 2018
3.936	120.0	13 May 2018	11.804	169.2	29 Oct 2018	17 Jun 2020	11.539	181.1	15 Dec 2020
3.936	120.0	17 Jul 2020	12.165	180.7	14 Jan 2021	22 Jul 2022	12.740	209.5	16 Feb 2023
3.936	120.0	03 Sep 2022	13.777	208.4	30 May 2023	14 Aug 2024	13.360	233.3	04 Apr 2025
3.936	120.0	11 Oct 2024	14.852	223.5	23 May 2024	17 Sep 2026	14.243	235.0	10 May 2027
3.936	120.0	17 Nov 2026	15.297	224.9	30 Jun 2027	24 Oct 2028	13.928	233.9	15 Jun 2029
3.936	120.0	26 Dec 2028	15.118	215.6	30 Jul 2029	16 Dec 2030	12.736	222.9	27 Jul 2031
3.936	120.0	12 Feb 2031	14.170	197.7	28 Aug 2031	22 Feb 2033	10.356	210.1	20 Sep 2033
3.936	120.0	14 Apr 2033	12.416	175.4	07 Oct 2033	06 May 2035	9.155	208.7	01 Dec 2035

BOEING

The values on this chart show the influence of the year on the Mars mission. One of the advantages of the solar-electric propulsion is that the performance is less sensitive to the year.

Effects of Reuse and Specific Impulse

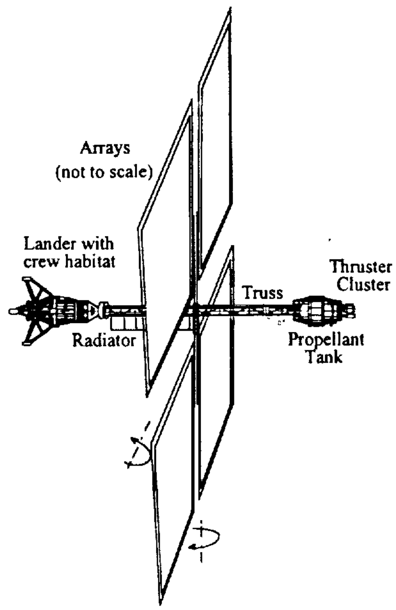
2018 Mars SEP Vehicle Departs 13000 km Circular Orbit	Isp = 2000 sec		Isp = 4000 sec	
	Expended	Reuse	Expended	Reuse
Total Vehicle Mass	542.4 mt	676.6 mt	287.9 mt	316.8 mt
Propellant Mass	357.7 mt	470.8 mt	122.3 mt	144.5 mt
Inert Mass	81.7 mt	102.8 mt*	62.5 mt	69.3 mt*
Mars Lander P/L	60.0 mt	60.0 mt	60.0 mt	60.0 mt
Transfer Habitat and Crew Capsule	43.0 mt	43.0 mt*	43.0 mt	43.0 mt*

* **Returned to 13000 km Circular Orbit for Reuse**

BOEING

This chart shows how the design of a human Mars mission can change when the transfer vehicle is reused and when the specific impulse is increased. As expected, the mass increases for a reusable system. Also, the higher specific impulse reduces the propellant mass considerably.

Lunar Transfer Vehicle Design



A lunar transfer vehicle concept is shown here using propulsion based on the autonomous solar power transfer option. The lunar vehicle is designed to operate from LEO.

Reusable Piloted Lunar SEP Vehicle

94 day Spiral Out, 40 day Return

Vehicle departs from, and Returns to, 300 km circular LEO

	kg	b
PAYLOADS		
Crew Transfer Cab	6,850	15,102
Lunar Lander System	22,844	50,362
Surface Habitat Payload	0	0
Total Payloads Wt	29,694	65,463
INERTS		
Power Generation (alpha = 8.4)	26,557	58,549
Thruster System (alpha = 3.4)	4,730	10,427
Tankage	8,436	18,599
Structure	3,594	7,923
RCS Propellant	1,476	3,255
Stage Dry Weight	44,794	98,753
PROPELLANTS		
Earth Spiral Out to Lunar Capture	58,643	129,285
Trans Earth Injection	6,889	15,187
Earth Capture (LEO)	18,830	41,514
Total Propellant Weight	84,362	185,986
TOTAL WT	158,850	350,201

The mass statement for the lunar transfer vehicle concept is shown here.

Lunar Solar-Electric-Propulsion Vehicle

Isp = 2000 s Maneuver	Low Thrust dV (m/s)	High Thrust dV (m/s)	T/W* of Vehicle (N/N)
LEO - Spiral Out Trans Lunar Injection	7093	3084	0.00005
Lunar Orbit Insertion	2070	900	0.0002
Trans Earth Injection	2013	915	0.0003
Earth Capture to LEO	6785	3084	0.0010

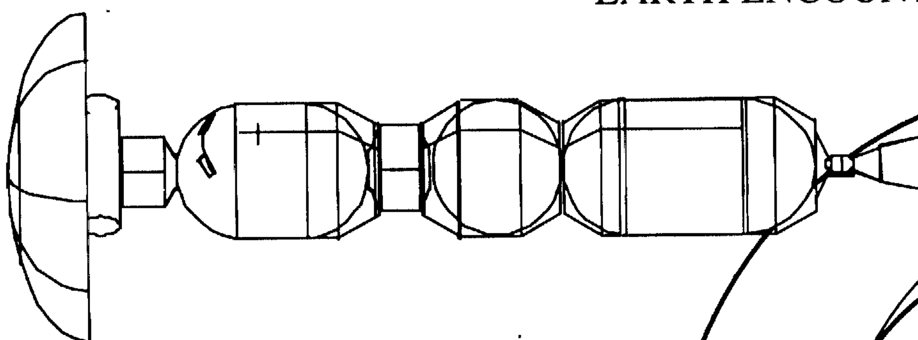
* Thrust divided by mass times gravity of nearest body

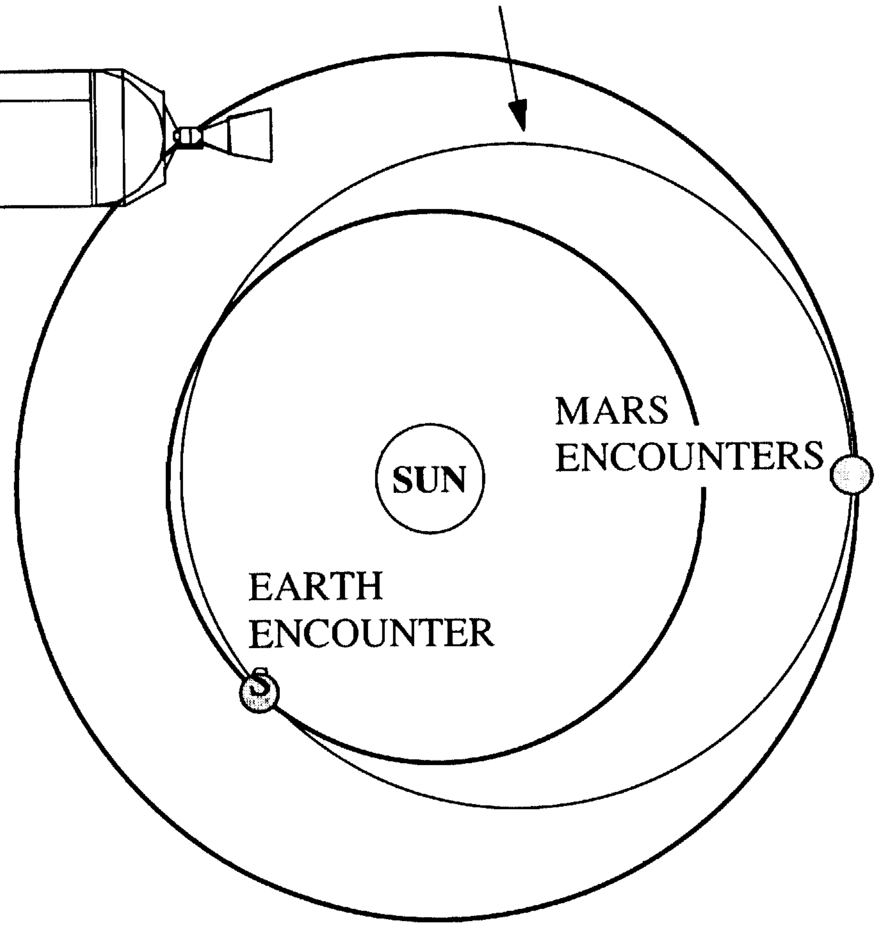
BOEING

This chart provides design some design parameters for the lunar transfer vehicle concept.

Mars Cycler Concept

SINGLE CYCLER ORBIT PERIOD 1.25 YEARS
EARTH ENCOUNTERS EVERY 5.0 YEARS
MARS ENCOUNTERS EVERY 3.25 YEARS.
DUAL CYCLER ARCHITECTURE PROVIDES
EARTH ENCOUNTER EVERY 2.5 YEARS





REPRESENTATIVE HIGH THRUST OTV STAGE (CHEMICAL O2 / H2 or NTP H2) for RENDEZVOUS WITH SEP EARTH - MARS CYCLER. DEPARTS LEO for HYPER-BOLIC TRAJECTORY. STAGE SHOWN WITH MARS LANDER

Additional In-Space Transportation Systems

- Improved Hall Effect Thruster
  - 4000 second Isp
- Hybrid System
  - Reusable Aerobraked OTV
    - Supplies part of the initial delta V for transfer
  - Autonomous transfer from intermediate trajectory to GEO

BOEING

Additional Effort: Effect of Improved Thruster

- Equatorial launch
- Partial array deployment for transfer
- No PPU, direct drive from high-voltage array

Specific impulse	2 000 s	4 000 s
Initial mass in LEO	27 000 kg	24 632 kg
Useable propellant	5 814 kg	2 797 kg
Mass in GEO	21 186 kg	21 835 kg
Residual propellant	174 kg	84 kg
Propulsion inert	2 810 kg	3 216 kg
Solar array degraded	1 053 kg	1 386 kg
Useful payload	17 149 kg	17 149 kg

BOEING

The data package used for the baseline autonomous transfer option came from published information considered reasonable at Glenn Research Center. Consideration of the possibilities in the time frame of interest resulted in postulation of an improved thruster which has a 4000 s specific impulse. The results of the vehicle analysis with the improved thruster are shown to reduce the launch mass nearly 9%, but a larger propulsion inert mass and degraded array mass are needed because the higher specific impulse requires more power. Another option that was discussed is to use a higher specific impulse later in the flight. This is an area worthy of further consideration.

Addition Effort: Other In-Space Transportation Requirements - Autonomous (Improved)

Inclination	deg	0	28.5
Loaded propellant	kg	2,881	4,138
Initial power (lift thrust)	kW	861	947
Final power (lift thrust)	kW	417	533
Initial thrust	N	28.5	31.4
Final thrust	N	13.8	17.7
Initial propellant flow rate	kg/s	0.000728	0.000800
Flight Time	days	90.0	90.1

- Momentum wheels of payload are used for attitude control
- Includes effects of Earth shadow
- Includes effects of power degradation due to radiation exposure

BOEING

Other parameters for the autonomous transfer case with the improved thruster are shown on this chart.

Additional Effort: Hybrid ROTV-AB and Autonomous Transfer

- Equatorial launch
- ROTV-AB provides transfer of 2 payloads to 300 km x 9000 km orbit
- Autonomous transfer of each payload to GEO
- ROTV aerobrakes directly from the 300 km x 9000 km orbit

BOEING

An intriguing possibility is to combine two or more of the basic transfer modes. An analysis was started to consider combining the ROTV-AB and the autonomous options. The launch and transfer would all be equatorial. The ROTV-AB would provide the initial transfer to an intermediate orbit. An orbit of 300 km x 9000 km was selected based on a few representative trajectory runs. In order to have an ROTV near the same size as the other reusable stages in this study, 2 elements with their autonomous propulsion were transferred to the intermediate orbit on each ROTV-AB flight. Autonomous transfer was then used to complete the transfer to GEO.

Additional Effort: Hybrid ROTV-AB and Autonomous Transfer; Autonomous Data

• Partial array deployment for transfer	
• No PPU, direct drive from high-voltage array	
• Specific impulse	2 000 s
• Init. mass at 300 x 90000	23 138 kg
• Useable propellant	3 643 kg
• Mass in GEO	19 495 kg
• Residual propellant	109 kg
• Propulsion inert	1 684 kg
• Solar array degraded	552 kg
• Useful payload	17 149 kg

BOEING

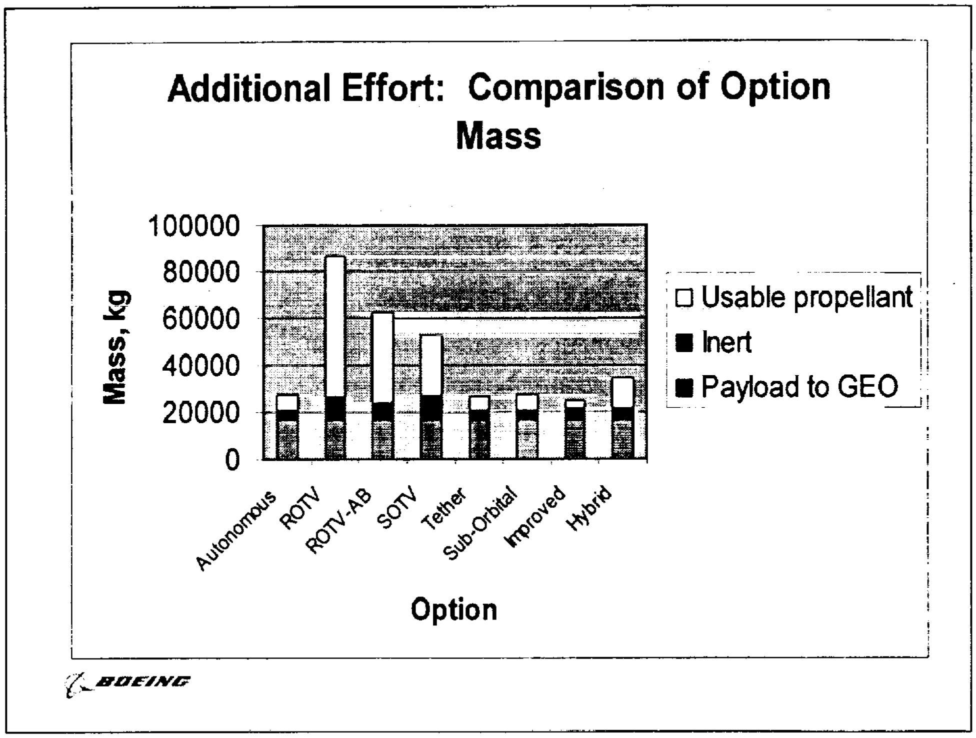
The hybrid case of a reusable, aerobraked OTV and autonomous transfer analysis provided the results shown. The autonomous transfer part of the results are presented on this chart. Compared to the baseline autonomous transfer, this portion of the transfer requires 3643 kg of propellant compared to 5814 kg, 1684 kg of propulsion inerts compared to 2810 kg, and 552 kg of degraded solar array compared to 1053 kg.

Additional Effort: Hybrid ROTV-AB and Autonomous Transfer; ROTV-AB Data

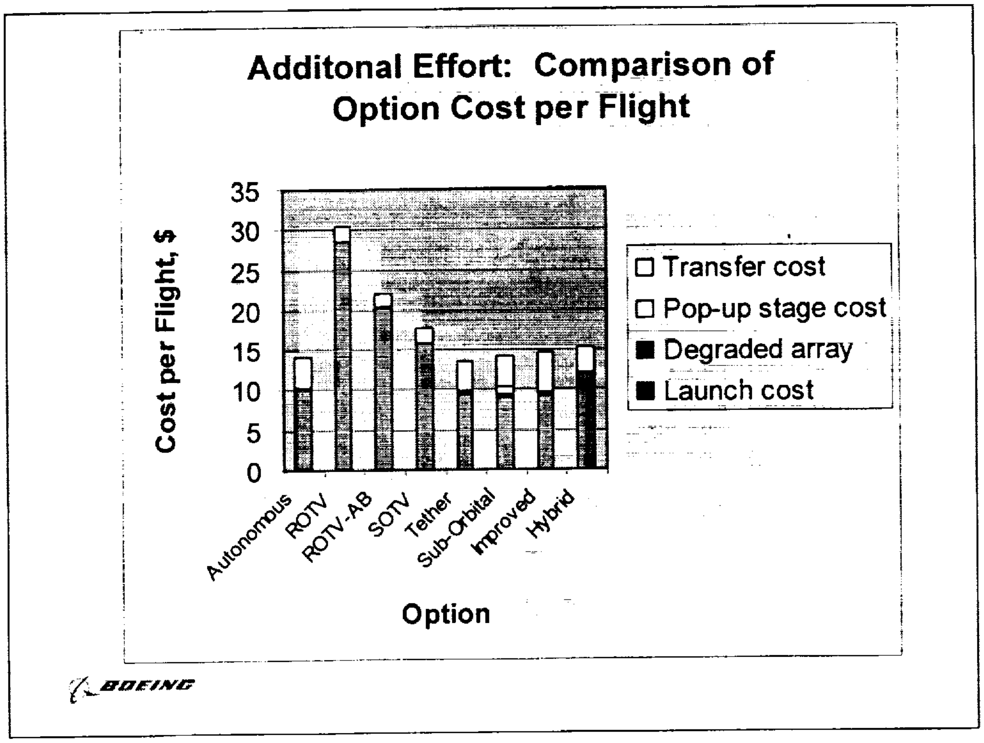
- Initial T/W 0.4
- Initial mass in LEO 68 658 kg
- Useful propellant 18 103 kg
- Reserves, resid., RCS 392 kg
- Stage inert (less aeroshell) 3 296 kg
- Aeroshell 591 kg
- Specific impulse 470 s
- Data for 2 payloads

BOEING

The analysis results for the ROTV-AB part of the hybrid case provided the results shown. Note that the ROTV-AB carries 2 elements each flight. The stage inert of 3296 kg compares very well to the inert mass of 5312 kg for the basic ROTV-AB design. The initial mass in LEO of 68,658 kg for 2 elements is much better than the ROTV-AB mass of 62,274 kg for 1 element but higher than the 27,000 kg for 1 element using only autonomous transfer.



Launch masses for the improved thruster case and the hybrid case are shown here compared to the basic cases. The improved thruster offers a modest, but significant, improvement in launch mass.



Recurring cost estimates for the improved thruster case and the hybrid case are shown here compared to the basic cases. Although the improved thruster case results in reduced initial mass in LEO and lower launch costs, it does not result in reduced cost per flight. The higher specific impulse requires more power and more propulsion inerts, which increase the estimates of the transfer stage costs and the degraded array costs.

The hybrid case does not result in lower costs than the fully autonomous case, but the results are very close. The hybrid option is attractive because the fully autonomous case requires a slow spiral from Earth's gravity well. When some transfer is provided by another means, the autonomous completion of the transfer is much easier. These results indicate that hybrid cases need further study. The hybrid case examined was rather arbitrary, and optimization of the drop-of orbit should improve the results. The relative merits of the ROTV-AB and SOTV cases indicate that a hybrid case with the SOTV and autonomous transfer should also be considered.

Additional Effort: Aerobraked SOTV

- SOTV has a large collector area
- Normal operation of the SOTV requires many thrust arcs to return SOTV to LEO
- The thrust arcs near LEO, to slow down SOTV and reduce apogee to LEO altitude, could be replaced with aerobraking using the solar collector as the aerobrake
- If the altitude is sufficiently high, the forces and thermal load on the solar collector do not exceed capability
- Velocity increment is therefore almost "free"
- Reduction of SOTV launch mass could be significant

BOEING

The SOTV results presented in the Task 2.2 section were based on the standard design of an SOTV, using propulsion for the four major velocity increment: LEO to GTO, GTO to GEO, GEO to GTO, and GTO to LEO. The ROTV-AB option replaces the last velocity increment with braking and a small circularization velocity increment. The same approach could be used with SOTV. Normally, adding an aerobrake to an ROTV is only worthwhile if the propulsive specific impulse is like that of a chemical rocket. With the higher specific impulse of the SOTV, an aerobrake would not be worthwhile in the usual sense of 1-5 passes into the upper atmosphere of the Earth. A special option is available with the SOTV, however. The solar collector could be moved to a position to allow it to serve as an aerobrake. Normally, the solar collector would not be designed for the forces or thermal load of an aerobrake, but accurate navigation could allow the SOTV to graze the upper atmosphere so gently that the loads would stay within the design parameters of the solar collector.

If the SOTV can use the solar concentrator as an aerobrake with no almost mass increase, the final velocity increment (GTO to LEO) is essentially eliminated. The reduction of the mass required at GTO on return could be significant, reducing the launch mass and the costs of the SOTV option. Analysis of this option was not possible in the study but should be considered in the future. Combining such an SOTV with autonomous transfer in a hybrid option should also be considered.

Discussion of Results

- Autonomous transfer attractive for "Power Tower"
- Alternates not presented may be attractive
- Cost results depend on launch costs

BOEING

The results that appear on previous charts are the output of an analysis for transfer of elements of a "Power Tower" design for a space solar power satellite. With the power tower concept, each element can be transported to GEO and added to the top of the satellite. With the power tower design, autonomous transport appears to be attractive. With other satellite designs, a similar transport option may not be available.

There are options which have not been analyzed in this study and which might be attractive. Some effort was initiated to consider hybrid transfer. Trajectory results were generated for using the ROTV-AB for part of the transfer followed by autonomous completion of the transfer. The results indicated that the propellant mass was greater for the hybrid than for the autonomous, as expected. Completion of the hybrid analysis was not possible before the end of the effort. The relatively low costs of the SOTV option indicate that a hybrid should be considered with SOTV and autonomous transfer.

The comparisons of the costs of the options were made with launch costs estimated for recurring costs only. Changes in the launch costs could change the order of the options. In particular, if a launch concept incorporating a tether reduced launch costs significantly, the SOTV option could become more attractive than the autonomous option. On the other hand, including all launch costs rather than just recurring costs would favor the autonomous transfer.

Discussion of Results (Continued)

- Limited effort and multiple required results
- Many options for further study

BOEING

The results of this study were limited to what could be accomplished in the scope provided. In order to examine multiple concepts and additional applications, all options for transfer could not be examined.

The results are believed to be an indication of the relative merit of some of the better concepts, but opportunities exist for improving these results. Numerous tether options could be considered. Additional hybrid effort appears warranted. Options with more than one specific impulse for the autonomous transfer could prove worthwhile.

Recommendations

- Continue to assume a solar-electric autonomous baseline
- Study tether, solar thermal, and combined options further
- Consider concentrators to reduce the photovoltaic array area, allowing additional radiation protection
- Compare options using the total transportation costs including development and financing costs

BOEING

Several recommendations resulted from this study. The autonomous in-space transportation option using solar-electric propulsion continues to be a reasonable choice for a baseline design. No clear reason to switch has been found. There are, however, several options that have sufficient promise to justify further study. More aggressive tether options could be very attractive if they prove feasible under further study and technology development. Solar thermal propulsion with aerobraking should not be dismissed without some analysis. Combining options, for example, to get the payloads to a higher altitude before starting the autonomous propulsion, should be studied further.

Concentrating solar arrays, perhaps using Fresnel lens concentrators, should be considered from a viewpoint of the entire satellite. If concentrator arrays work well for the operational period, they could be attractive for the transfer by allowing additional radiation protection. The partial deployment of the solar arrays might be eliminated.

The recurring cost analysis performed for this study provided a top-level estimate for initial comparison purposes. More detailed and complete cost calculations could show whether the inclusion of development and finance costs would change the selection.

Concluding Remarks

- The autonomous transfer with solar-electric propulsion appears more cost-effective than reusable OTV options
  - Aerobraked SOTV for partial transfer may still be viable
- Solar array degradation during transfer is a major concern
  - Partial deployment of the array significantly reduces damage
  - Designs are needed to assure completion of deployment
- Equatorial launch is more cost-effective than KSC launch into 28.5 deg. orbit
- Tether transfer provides a modest improvement
  - The tether length is limited by the momentum recovery from the launch vehicle
  - Increasing the node altitude should be considered
  - Electromagnetic propulsion of the node should be considered

BOEING

The results indicate that autonomous transfer with solar-electric propulsion, using the arrays that will be used by the power tower, is a more cost-effective option than the reusable orbit-transfer vehicles. Solar array degradation during the transfer can reduce the output of portions of the array by over 50%. To minimize the effect of that damage, the arrays could be partly deployed for the transfer, then fully deployed at GEO. No effort was possible in this study to design arrays that could be partly deployed; designs that would provide reliable completion of the deployment are needed. This study did not consider arrays with concentration, which would allow increased protection of the photovoltaic cells from radiation damage.

An alternative was suggested for an SOTV which provides part of the velocity, then aerobrakes using multiple passes to return to LEO. This option might still be viable but was considered outside the current scope.

There appears to be a significant advantage to an equatorial launch site. Keeping the orbit inclination low reduces the transfer costs. With a program of the magnitude expected for solar power, a new launch site would probably be required.

Tethers offer potential cost reductions. The conservative option analyzed offers minimal benefits but suggests the need for further analysis of more ambitious tether momentum-exchange options.

REPORT DOCUMENTATION PAGE	*Form Approved* *OMB No. 0704-0188*

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1. REPORT DATE *(DD-MM-YYYY)*	2. REPORT TYPE	3. DATES COVERED *(From - To)*
22-12-1999	Final Technical Report	22-09-99 through 22-12-99

4. TITLE AND SUBTITLE
Final Report
In-Space Transportation for GEO Space Solar Power Satellites

5a. CONTRACT NUMBER NAS8-98244, Mod 3
5b. GRANT NUMBER
5c. PROGRAM ELEMENT NUMBER

6. AUTHOR(S)
Martin, James A.

Donnahue, Benjamin B

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5d. PROJECT NUMBER
5e. TASK NUMBER
5f. WORK UNIT NUMBER

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
The Boeing Company, Reusable Space Systems (RSS)
5301 Bolsa Ave.
Huntington Beach, CA 92647-2099

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

NASA
Marshall Space Flight Center (MSFC)
Huntsville, AL 35812

10. SPONSOR/MONITOR'S ACRONYM(S)

11. SPONSOR/MONITOR'S REPORT NUMBER(S)

12. DISTRIBUTION/AVAILABILITY STATEMENT

13. SUPPLEMENTARY NOTES

14. ABSTRACT

This report summarizes results of study tasks to evaluate design options for in-space transportation of geostationary Space Solar Power Satellites. Referring to the end-to-end architecture studies performed in 1988, this current activity focuses on transportation of Sun Tower satellite segments from an initial low Earth orbit altitude to a final position in geostationary orbit (GEO; i.e., 35,786 km altitude.circular, equatorial orbit). This report encompasses study activity for In-Space Transportation of GEO Space Solar Power (SSP) Satellites including: 1) assessment of requirements, 2) design of systme concepts, 3) comparison of alternative system options, and 4) assessment of potential derivatives.

15. SUBJECT TERMS
All applicable Terms & Conditions contained in Contract NAS8-98244, Mod 3

16. SECURITY CLASSIFICATION OF:			17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON
a. REPORT	b. ABSTRACT	c. THIS PAGE			Mark Henley
					19b. TELEPHONE NUMBER *(Include area code)* (714) 934-0616